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PROCEEDINGS OF THE 1987 AIRCRAFT/ENGINE STRUCTURAL INTEGRITY PROGRAM (ASIP/ENSIP) CONFERENCE

AF WRIGHT AERONAUTICAL LABORATORIES

John W. Lincoln
ASD/Deputy for Engineering

Thomas D. Cooper
Materials Integrity Branch
Systems Support Division

June 1988

Final Report for Period December 1987 - March 1988

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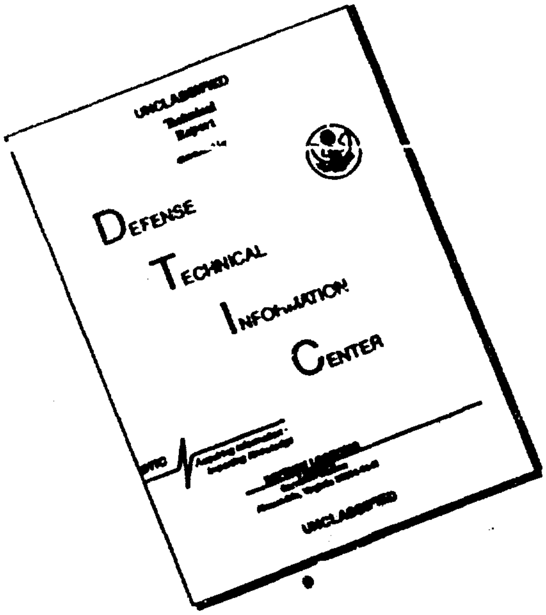

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THOMAS D. COOPER, Chief
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FOR THE COMMANDER

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This report is a compilation of the papers presented at the 1987 Aircraft/Engine Structural Integrity Program (ASIP/ENSIP) Conference held at the Hilton Palacio del Rio Hotel, San Antonio, Texas on 1-3 December 1987.

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FOREWORD

This report was compiled by the Materials Integrity Branch, Systems Support Division, Materials Laboratory, Air Force Wright Aeronautical Laboratories, Wright-Patterson AFB, Ohio. It was initiated under Task 24180704 "Corrosion Control & Failure Analysis" with Thomas D. Cooper as the Project Engineer.

This technical report was submitted by the editors.

The purpose of this 1987 Conference was to bring together technical personnel in DOD and the aerospace industry who are involved in the various technologies required to insure the structural integrity of aircraft gas turbine engines and airframes. It provided a forum to exchange ideas and share new information relating to the critical aspects of durability and damage tolerance technology for aircraft systems. The Conference was sponsored by the Aeronautical Systems Division Deputy for Engineering and Materials Laboratory of the Air Force Wright Aeronautical Laboratories, Wright-Patterson AFB, Ohio. It was hosted by the Air Force Logistics Command's San Antonio Air Logistics Center.

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1-3 December 1987

Hilton Palacio Del Rio Hotel
San Antonio, Texas

SPONSORED BY:

ASD/Deputy for Engineering

AFWAL/Materials Laboratory

HOSTED BY:

San Antonio Air Logistics Center
Material Management Directorate

Fighter/Tactical/Trainer
Systems Program Management
Division (SA-ALC/MMS)

AGENDA

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INTRODUCTION

This report contains the proceedings of the 1987 Aircraft/Engine (ASIP/ENSIP) Conference held at the Hilton Palacio Del Rio Hotel in San Antonio, Texas ~~from the first through the~~ third of December of 198?. The conference, which was sponsored by the ASD Deputy for Engineering and the AFWAL Materials Laboratory, was hosted by the San Antonio Air Logistics Center Material Management Directorate Fighter/Tactical/Management Division (SA-ALC/MMS).

The program for this year followed the format of the 1986 conference by including papers from integrity programs for aircraft and engines. The papers given on engine nondestructive inspections were particularly relevant to the work in this field at the SA-ALC. Also included in this years program was an update of the work in mechanical subsystems integrity (MECSIP) that was introduced at the 1985 ASIP Conference. The high quality of papers given at the conference attests to the good work in progress to maintain the structural reliability of aircraft.

The sponsors are indebted to their hosts for their support of the conference which included arranging for speakers and tours. They made the arrangements for Major General L. G. Curtis to get the meeting off to a good start with some very timely remarks. They also arranged for tours of the San Antonio Air Logistic Center and the Southwest Research Institute. The sponsors are appreciative of the efforts made by these two organizations to make the these tour events significant contributions to the success of the conference. They are also indebted to the conference speakers for their excellent technical presentations and to the attendees for their support of the conference. Much of the success of this conference is due to the efforts of Jill Jennewine and David Bell from the Universal Technology Corporation. Their contribution is appreciated.

John W. Lincoln
ASD/ENFS

Thomas D. Cooper
AFWAL/MLSA

SESSION I: COMPOSITE STRUCTURES

Keywords: Composite structures, composite
materials, aircraft engines, aircraft
gas turbine engines. airframes, (SWD)

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DECEMBER 1987

LESSONS LEARNED FOR COMPOSITE AIRCRAFT STRUCTURES QUALIFICATION

By

R. WHITEHEAD

Presented to

1987 AIRCRAFT/ENGINE (ASIP/ENSIP) CONFERENCE
SAN ANTONIO, TEXAS
DECEMBER 1-3, 1987

NORTHROP

LESSONS LEARNED FOR COMPOSITE AIRCRAFT STRUCTURES QUALIFICATION

ABSTRACT

An overview of the extensive experience, lessons learned, and recommended certification procedures from two major USAF composite R&D programs is presented. Subject areas discussed in detail are static strength, fatigue/durability, and damage tolerance.

INTRODUCTION

The increased application of advanced composite materials in aircraft structures requires a critical assessment of the adequacy and applicability of existing metallic oriented certification specifications to this emerging class of materials. To do this, it is necessary to recognize the inherent differences between metals and composites. These inherent property differences led to an ad hoc qualification approach for early production hardware. This individual requirement development, or pay-as-you-go approach, while satisfying the immediate need at a "single copy" price, limited generic application and prolonged airframe development. Thus, in the long run, this approach was more expensive and time-consuming than a subscription price approach, which repeatedly uses established standardized specifications. A need exists, therefore, for an orderly, unified, consistent, and verified approach for designing, certifying, and force managing composite structures. This need has been addressed in two Air Force sponsored R&D programs. The purpose of these programs was to develop an extensive test data base on specimens ranging in complexity from coupons through elements, element combinations, subcomponents, and full-scale wing and fuselage structures. This data base was then used to develop draft certification specifications for static strength, durability, and damage tolerance. In addition to the specifications, certification compliance procedures were also developed. Details of this work were

presented previously in the open literature in References 1-8. This paper discusses experience, lessons learned, and recommended certification procedures for static strength, fatigue/durability, and damage tolerance of composite structures.

STATIC STRENGTH

Experience

Static strength testing of composites has shown that several inherent differences exist between composites and metals. These inherent differences are summarized in Figures 1-4.

Figure 1 compares the static strength notch sensitivity of composites and metals to stress concentrations such as fastener holes. The notched strength of metals follows the net section strength reduction line. In contrast, composites are very notch sensitive to fastener holes under both tension and compression loading. In fact, this behavior is similar to the

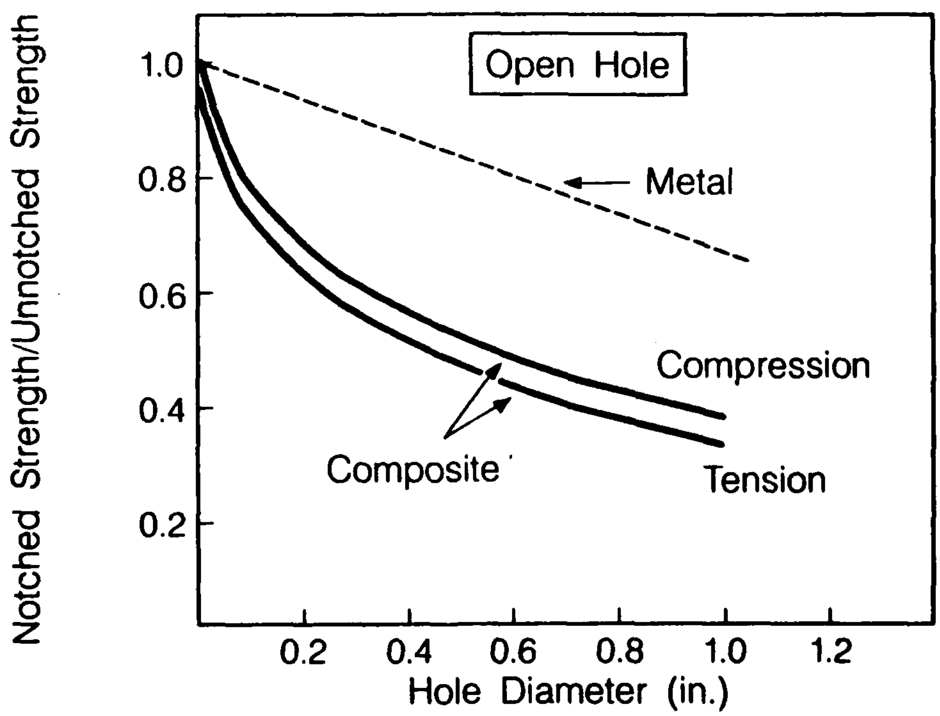


FIGURE 1. STATIC NOTCH SENSITIVITY COMPARISON OF GRAPHITE/EPOXY AND ALUMINUM TO FASTENER HOLES

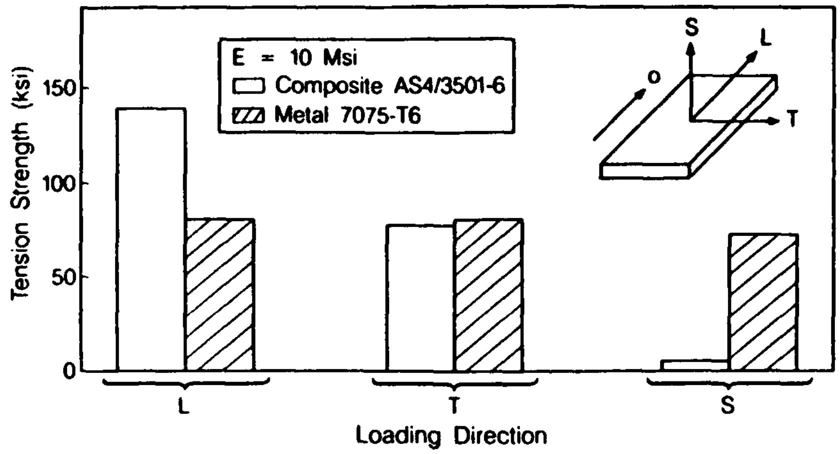


FIGURE 2. INFLUENCE OF LOADING DIRECTION ON GRAPHITE/EPOXY AND ALUMINUM STATIC STRENGTH

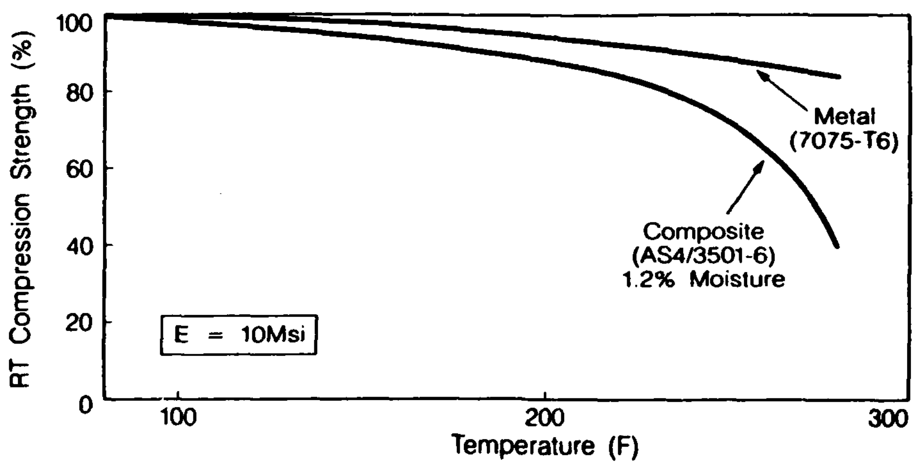


FIGURE 3. INFLUENCE OF ENVIRONMENT ON COMPRESSION STATIC STRENGTH RETENTION

Material	Static Strength Variability		Design Allowable/ Mean
	α	C.V. (%)	
Aluminum	35	3.5	0.95
Composite – Laminate	20	6.5	0.89
Composite – Bonded/Cocured	12	10.0	0.84

FIGURE 4. COMPARISON OF STATIC STRENGTH VARIABILITY OF COMPOSITES AND METALS

linear elastic response of metals in the presence of fatigue cracks. The static strength notch sensitivity of composites is caused by their essentially linear elastic load-strain response. The sensitivity of static strength to loading direction is also different for composites and metals. Figure 2 shows this comparison for longitudinal (L), transverse (T), and out-of-plane (S) tension loading. Aluminum static strength is relatively insensitive to loading direction. In contrast, graphite/epoxy static strength is significantly influenced by loading direction. This is caused by the anisotropic characteristics of composites. The differences in L and T direction strength are simply a function of the percent 0°, ±45° and 90° plies used in the layup. However, strength in the S direction is controlled by the interlaminar tension strength between the plies, which is very low and is in the 3-4 ksi range.

Composites, which exhibit matrix controlled failure modes (e.g., compression), are sensitive to the aircraft hygrothermal environment. In particular, the effects of temperature and moisture have a synergistic effect. Therefore, the strength degradation of composites in hot/wet environments controls their maximum service usage temperature. Figure 3 shows the influence of temperature and moisture content on composite compression static strength. These data are for the 350°F cure system AS4/3501-6. Figure 3 shows that as the test temperature is increased above 220°F, strength loss (relative to ambient) is 15 percent, while at 250° strength loss is more than doubled to 33 percent. This large strength loss is due to rapid degeneration in resin properties (e.g., shear stiffness), which is caused by the resin approaching its glass transition temperature. In contrast, Figure 3 shows that aluminum strength is much less sensitive to temperature. Figure 4 compares the static strength variability of composites and metals. Because of their anisotropic heterogeneous characteristics, composites exhibit higher variability for laminate failure modes. For cocured composite-to-composite failure modes, even higher variability (10 percent coefficient of variation) is observed. This causes the ratio of the B-basics design allowable to mean value to be lower for composites compared to metals.

These property differences between composites and metals (notch sensitivity, weak transverse properties, matrix-dominated failures, higher variability, hygrothermal sensitivity) must be addressed in static strength

structural certification. It is emphasized that these properties do not negate the weight efficiency of composite structures, just that different parameters (from metals) are important in composite certification.

The historical approach to design, analysis, and certification of composite structures has been similar throughout the industry. Composite design methods have been tailored to recognize the unique composite properties shown in Figures 1-4. In addition, conservative strain allowables (3,000-5,000 μin/in at design ultimate load) coupled with semi-empirical analysis methods have been used for flight hardware. Design ultimate load had also been maintained at 1.5 times design limit load. Structural certification approaches have mainly been based on metals experience. This overall design, analysis, and certification approach has led to the following composite hardware experience:

1. Significant weight savings compared to metal structure
2. Mixed certification success
3. Successful in-service application.

Problems associated with the static strength certification of composite structures are discussed below.

Figure 5 shows an outer wing box subcomponent tested in Reference 1. The box consists of three bays with cocured intermediate spar to lower skin joints and an upper skin access hole. Fifteen wing boxes were tested as follows: 1) three room temperature ambient (RTA) static tests; 2) three RTA residual static strength tests after two lifetimes of severe fatigue loading; 3) three 250°F/1.3 percent moisture (ETW) static tests and six ETW residual static strength tests after two lifetimes of severe environmental fatigue loading. No fatigue failures occurred. The results are presented in Figure 6. The predicted failure mode for all the RTA tests was a lower skin failure mode, which was observed in five of the six tests. The failure mode in the sixth test was an unanticipated separation of the cocured intermediate spar/lower skin joints. This failure mode was not expected because the joint had a margin of safety of 1.35. Post-test failure analysis led to the following scenario for this failure mode. In addition to the shear flow in the joint, stress analysis of the steel shear clips, which were used to transfer shear load through the pylon rib, showed that the clips had a low torsional

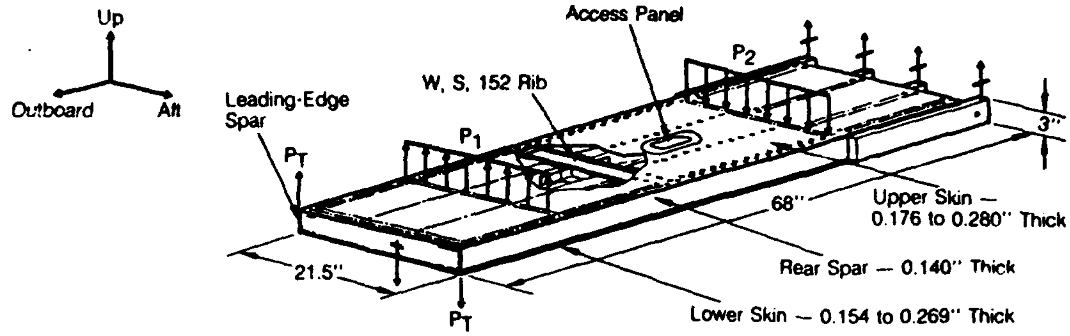


FIGURE 5. WING BOX SUBCOMPONENT TESTED IN REFERENCE 1

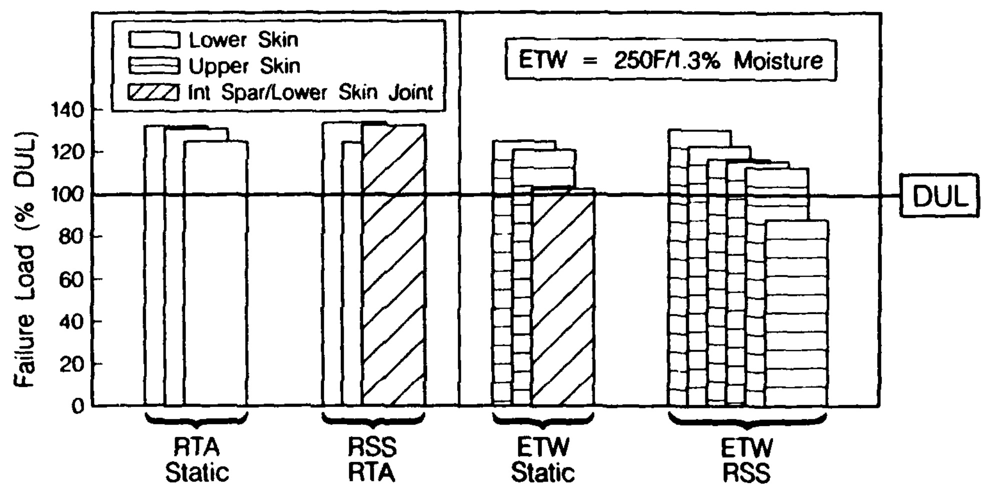


FIGURE 6. SUMMARY OF WING BOX SUBCOMPONENT FAILURE LOADS AND MODES

stiffness. The load carrying capability of the bonded joint was reduced due to bending moments induced by the relatively long length, L, of the shear clip. This is shown in Figure 7. Flexibility of the clip, where it attaches to the rib, caused the moment M to be small relative to VL, with the result that the moment must be reacted through the skin/spar joint. It was this flatwise tension load combined with the shear flow which led to the failure in the cocured joint. To inhibit this failure mode in the full-scale wing box, the shear clip was redesigned with enhanced torsional stiffness.

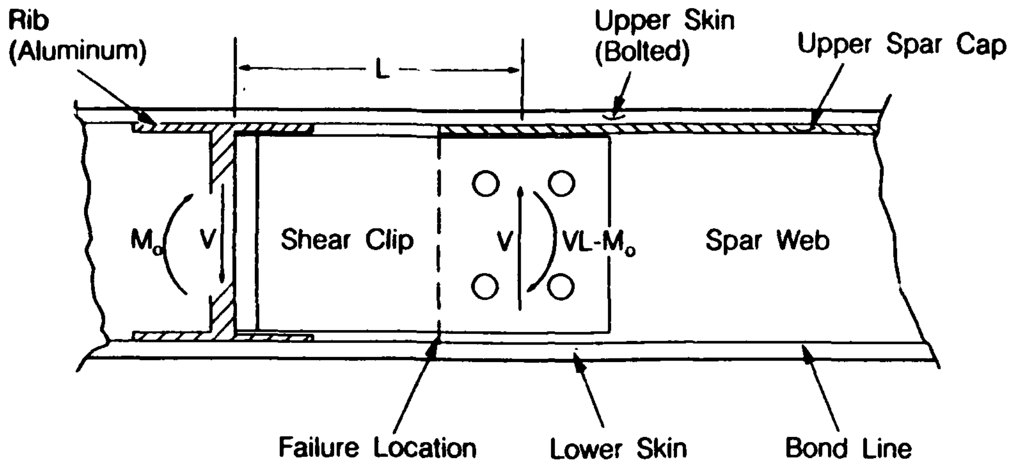


FIGURE 7. WING BOX SUBCOMPONENT LOWER SKIN/INTERMEDIATE SPAR JOINT SHEAR CLIP LOADING

The predicted failure mode for all the ETW tests was upper skin failure at the access hole, which was observed in eight of the nine tests. However, considerable strength scatter was observed (87 percent to 132 percent DUL). The ninth specimen failed by the same unanticipated intermediate spar/lower skin cocured joint failure mode previously observed under RTA conditions.

Following testing of the wing box subcomponents, four full-scale wing boxes were tested. The failure loads and failure modes are summarized in Figure 8. The predicted failure mode for the two RTA tests was lower skin failure. However, both test failures were caused by failure of the ten intermediate spar/lower skin cocured joints, as shown in Figure 9. This was the same unexpected failure mode as that observed in the wing box subcomponent and occurred despite a careful redesign of the spar to rib shear clips. The predicted ETW failure mode was upper skin failure, which was observed in both test articles. The predicted and observed failure mode change between RTA and ETW tests was caused by the static design conditions. For ambient conditions, a subsonic high N_z pull-up maneuver was the most critical design case, whereas a supersonic moderate N_z pull-up maneuver was the most critical design case for ETW conditions.

The occurrence of unexpected failure modes in full-scale static strength tests has occurred in many other hardware development programs.

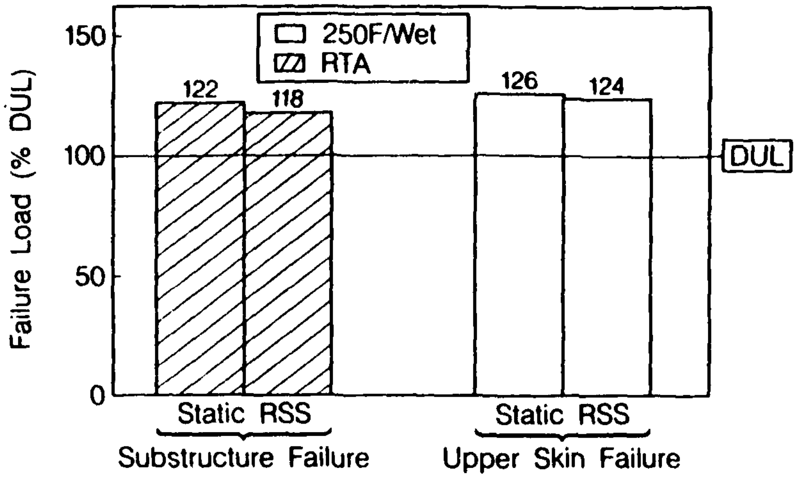


FIGURE 8. FULL-SCALE WING FAILURE MODES AND LOADS

FIGURE 9. FULL-SCALE WING BOX RTA INTERMEDIATE SPAR/LOWER SKIN CO-CURED JOINT STATIC FAILURE MODE

Unfortunately, for obvious reasons, many have not been documented in the open literature. One exception is the NASA ACEE program experience

documented in Reference 9. Unexpected failure modes were observed in three separate full-scale hardware tests. One example for a transport aircraft vertical stabilizer is shown in Figure 10.

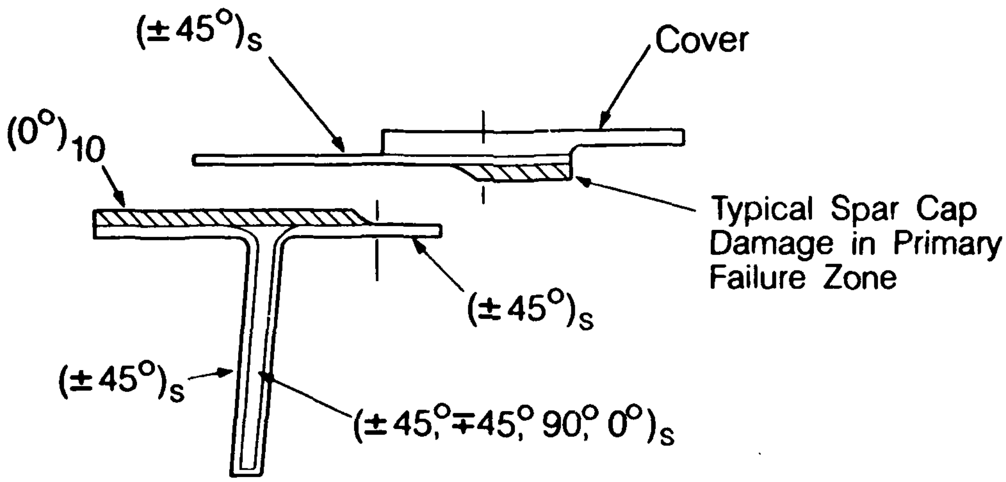


FIGURE 10. FRONT SPAR CAP FAILURE IN A TRANSPORT AIRCRAFT VERTICAL STABILIZER STATIC TEST (REFERENCE 9)

The fin failed at 98 percent of design ultimate load during the planned test to 106 percent of design ultimate load in bending. Failure caused separation of the cover and front spar along the entire length of the spar as well as considerable internal damage to rib structure. After an investigation, the cause of failure was determined to be due to secondary loads, of which the principal contributor probably was local buckling of the cover near the front spar interface. While local buckling beyond limit load was allowed in the design, the influence of loads caused by buckling on the integrity of the structure was unexpected. Interlaminar tension forces caused delamination of the spar cap as shown by the insert in Figure 9 and ultimate separation along the line of fasteners.

Lessons Learned

Some very important lessons have been learned from our static strength certification experience. First, the low interlaminar strength of composites makes them sensitive to out-of-plane loads. Out-of-plane loads can arise directly (e.g., from fuel pressure) or indirectly from in-plane loads. The

most difficult loads to design and analyze for are those loads which arise insidiously in full-scale built-up structures. It is very important, therefore, to recognize all potential sources of out-of-plane loads and design composite structure to maximize interlaminar strength.

An example of this requirement is presented in Figure 11 for a composite torque box with cocured substructure and skins which are allowed to buckle prior to design ultimate load. The skin postbuckling produces out-of-plane flatwise tension, compression, and bending loads in the spars. The amount of postbuckling is limited by the strength of the skins, the skin/substructure interface, and the spar substructure. Figure 11 shows the possible failure modes for the all-composite torque box subjected to postbuckling loads. The cover skin can fail if the local outer fiber compression strength of the laminate is exceeded due to bending, or if the interlaminar shear strength of the laminate is exceeded. The skin/substructure attachment can fail if delamination occurs in the cocured joint, or if the transverse load exceeds the pull-through strength of the fasteners/laminate combination, or the strength of the spar caps or the transverse shear strength of the flange. In addition, the radius portion of the spar flange/web can fail if the interlaminar shear or flatwise tension strength is exceeded. Finally, compression failure of the spar web can occur if the web does not have enough strength to resist the crushing loads induced by buckling of the cover skins and overall stabilizer bending.

Another important conclusion from our static strength certification experience is as follows. The full-scale static strength test identifies structural "hot-spots." This is the reverse of our experience with metal structures, where "hot-spots" are generally identified in the full-scale fatigue test.

Figure 12 summarizes the lessons learned from static strength certification testing of composite structures.

Certification Recommendations

The follow static strength certification recommendations are based on the experience and lessons learned described above.

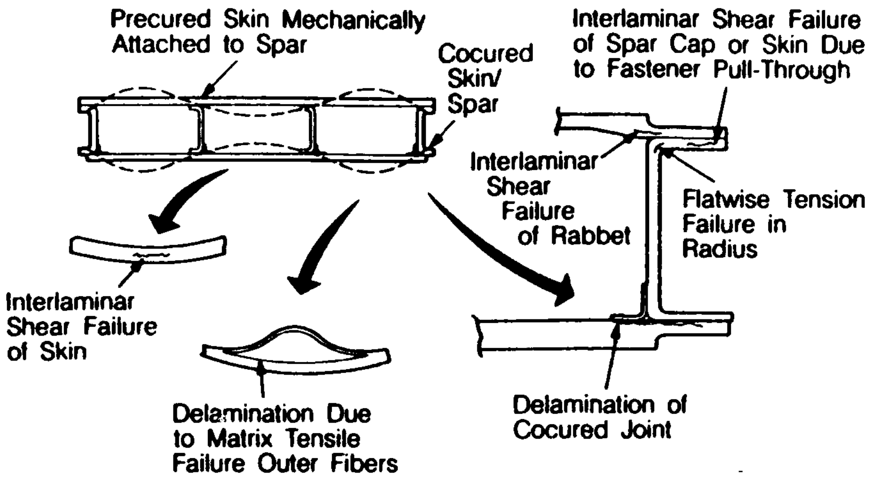


FIGURE 11. POTENTIAL OUT-OF-PLANE FAILURE MODES IN A COMPOSITE TORQUE BOX

- Inherent Property Differences Exist Between Composites and Metals
- Composite Structures Are Sensitive to Out-of-Plane Loads
- Multiplicity of Potential Failure Modes
- Failure Modes of Full-Scale Structures Are Difficult to Predict
- Static-Strength Test Identifies Structural "Hot Spots"

FIGURE 12. SUMMARY OF LESSONS LEARNED FROM STATIC STRENGTH CERTIFICATION TESTING OF COMPOSITE STRUCTURES

Material selection. Material selection is crucial to the successful application of composites in primary aircraft structures. Composite materials have operating limits just as aluminum does. Selection of a composite material should be based on the relationship between the aircraft hygrothermal envelope and the material operating limits (MOL). The material operating limit is reached when the synergistic effect of temperature and moisture causes

severe degradation in resin mechanical properties. Good design practice dictates that composites should not be used in this regime.

The concept of a MOL is shown schematically in Figure 13 for an environmentally sensitive failure mode. The decrease in design allowable strain as temperature increases is shown for a constant moisture level. Catastrophic strength loss coincides with severe degradation in resin properties as the glass transition temperature (Tg) is reached. In order to operate in a safe regime, the MOL should be reduced below the Tg by a safety factor K. This produces the shaded material service envelope shown in Figure 13.

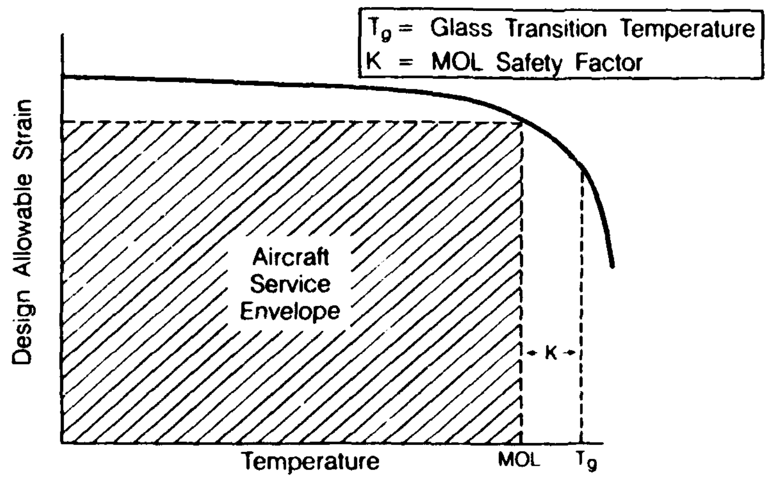


FIGURE 13. MATERIAL SELECTION CRITERION

The discussion above appears to be a statement of the obvious. However, violations of the approach shown in Figure 13 have occurred and have led to disastrous certification experiences. It should also be noted that careful compliance with the requirements in Figure 13 will minimize environmental issues in the subsequent certification test program.

Design verification testing. Design development tests are conducted prior to the full-scale test. The objective of these tests is to validate the design of critical structural features.

A building block approach to design development testing is crucial for the certification of composite structures because of their sensitivity to out-of-plane loads and their multiplicity of potential failure modes. This is

discussed in more detail in Reference 5. The essence of the building block approach for composites is as follows. First, use the design/analysis of the aircraft structure to select critical areas for test verification. Second, determine the most strength-critical failure mode for each design. Third, select the test environment which will produce the strength critical failure mode. Special attention should be given to matrix sensitive failure modes (such as compression and bondline) and potential stress "hot spots" caused by out-of-plane loads. Following selection of the critical failure modes, a series of specimens is designed, each one to simulate a single failure mode. These specimens will generally be low complexity specimens. However, the crux of the building approach is to also design test specimens which simulate progressive design complexity. In this way, multiple potential failure modes are interrogated.

This building block method to design development testing provides a step-by-step approach to composite design development testing, which has several advantages:

1. The influence of the environment on individual failure modes is determined.
2. The interaction of failure modes is established from the known behavior of individual failure modes.
3. Scale-up effect is determined from data on smaller-scale specimens.
4. "Hot spots" induced in complex structures can be analyzed relative to the known behavior of smaller specimens.

Specimen complexity should be a function of the design feature being validated and the predicted failure mode. Special attention should be given to correct failure mode simulation, since failure modes are frequently dependent on the test environment. In particular, the influence of complex loading on the local stress at a given design feature must be evaluated. In composites, out-of-plane stresses can be detrimental to structural integrity and therefore require careful evaluation.

An example of the building block approach for specimen complexity is given in Figure 14, which shows the approach used for the wing structure in Reference 1. Here the wing structure has been broken down into critical

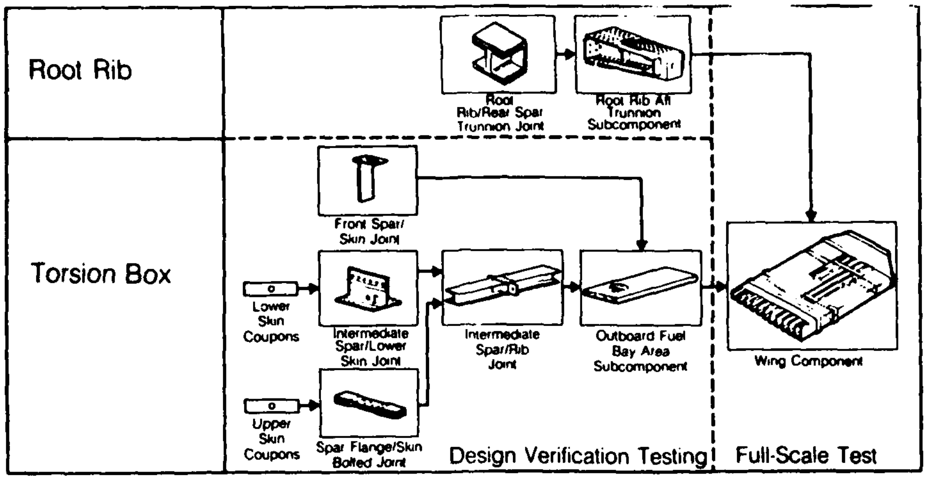


FIGURE 14. BUILDING BLOCK APPROACH USED FOR THE WING STRUCTURE IN REFERENCE 1

areas. Each critical area has been simulated in a test specimen whose complexity is governed by the necessity to simulate all potential failure mode(s). Particular attention was given to matrix critical failure modes. The following recommendations are made for specimen complexity simulation in design development testing:

1. Use the design/analysis of the aircraft structure to select critical areas for test verification.

2. Specimen complexity should be controlled by the requirement to simulate the correct (full-scale structure) failure mode(s) in the specimen.

3. Special attention should be given to matrix sensitive failure modes, such as compression, bondline, and hole wear.

4. Potential "hot spots" caused by out-of-plane loads should be carefully evaluated.

The sensitivity of composite matrix dominated failure modes to the aircraft hygrothermal environment makes environmental test simulation a key issue. Environmental test simulation should be considered separately for static and durability testing. However, the static test philosophy will form

an integral part of the overall test philosophy. The philosophy for static design development tests should be that the test environment used will be the one that produces the failure mode which gives the lowest static strength.

Full-scale test. The full-scale static test is the most crucial qualification test for composite structures for the following reasons. Secondary loads are virtually impossible to eliminate from complex built-up structures. Such loads can be produced by eccentricities, stiffness changes, discontinuities, fuel pressure loading, and loading in the post-buckled range. Some of these sources of secondary loads are represented for the first time in the full-scale structural test article. These loads are not a significant design driver in metallic structures. However, the poor interlaminar strength of composites makes them extremely susceptible to out-of-plane secondary loads. It is very important, therefore, to carefully account for these loads in the design of composite structures. Unfortunately, there is a general state of uncertainty as to the source, magnitude, and effects of secondary loads in complex built-up full-scale composite structures. This has been confirmed by several documented examples of unanticipated secondary loads leading to unexpected failure modes in full-scale composite structural static tests.

In addition, a detailed correlation in terms of measured load and strain distributions, structural analysis data, and environmental effects between the design development and full-scale test data will be necessary to provide assurance of composite static strength. Static test environmental degradation must be accounted for separately either by adverse condition testing, by additional test design factors, or by correlation with environmental design development test data.

Work in Reference 1 has shown that the RT/ambient static test plays the most significant role in revealing unexpected hot spot failures from secondary out-of-plane loads. A room temperature environment is, therefore, recommended for the full-scale static test, which should be conducted to failure. This recommendation is not universally accepted by all certification agencies. Some agencies favor an environmental static test which corresponds to the temperature and absorbed moisture level of the most critical static design condition. This issue is most significant in fighter aircraft where a

hot/wet failure mode is often the most critical design condition. Unfortunately, a full-scale environmental static test is very expensive and time-consuming. A possible solution to this problem, proposed by Dr. Lincoln in Reference 3, is as follows: the design philosophy would not permit any significant change in failure mode due to environment. For example, consider an aircraft structure, where the most critical design load condition is associated with a hot/wet environment. The requirement of this philosophy would be for the structure to be designed to have the same failure mode under both hot/wet and RT/ambient conditions. This approach would eliminate the need for hot/wet qualification testing. If this design requirement had been adopted for the fighter aircraft wing structure in Reference 1, a weight penalty of approximately 6 percent would have been incurred in the main wing box structure.

FATIGUE/DURABILITY

Composites have superb fatigue properties. Figure 15 compares the RTA spectrum fatigue behavior of graphite/epoxy and aluminum under their respectively most sensitive fatigue loading modes. It can be seen that graphite/epoxy fatigue response is vastly superior to aluminum. This has been confirmed by the extensive environmental data base generated in Reference 1 and summarized in Reference 6.

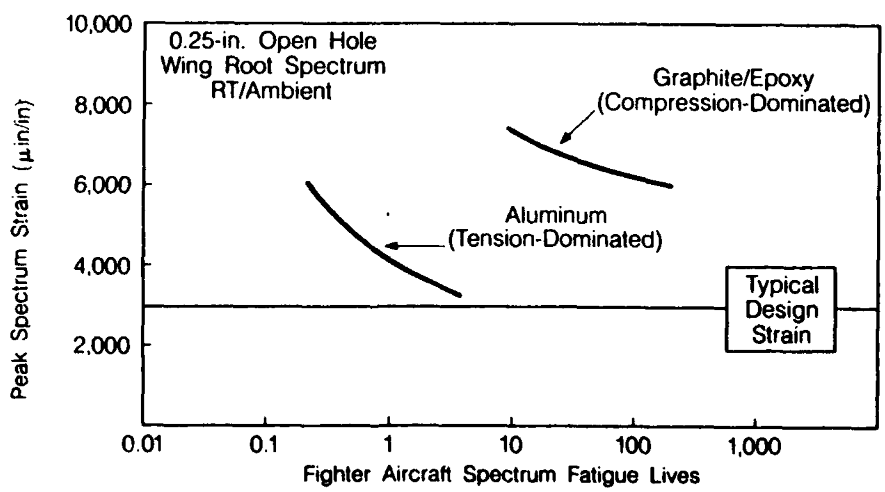


FIGURE 15. COMPARISON OF GRAPHITE/EPOXY AND METAL SPECTRUM FATIGUE BEHAVIOR

In this work, several hundred specimens ranging in complexity from coupons to elements to element combinations to subcomponents to full-scale wing and fuselage structures were tested under very severe environmental and fatigue loading conditions. The fatigue loading was much more severe than the design spectrum. All tests were conducted with the maximum spectrum load set at 72 percent of the previously determined average static failure load, which led to test spectra with significant load enhancement factors compared to design. In addition, severe quasi-real time environmental cycling was imposed on the test articles. This involved continuous thermal cycling, severe thermal spikes, and regular moisture absorption/deabsorption cycles. Representative results are presented in Figure 16. No fatigue failures occurred in the two lifetimes of fighter aircraft fatigue loading and all specimens were residual static strength tested at 250°F/wet environmental conditions. Figure 16 shows some variability in residual strength; however, this was determined to be due to scatter in static strength rather than fatigue degradation. It should be noted that even matrix sensitive failure modes such as compression and out-of-plane flatwise tension were not fatigue sensitive.

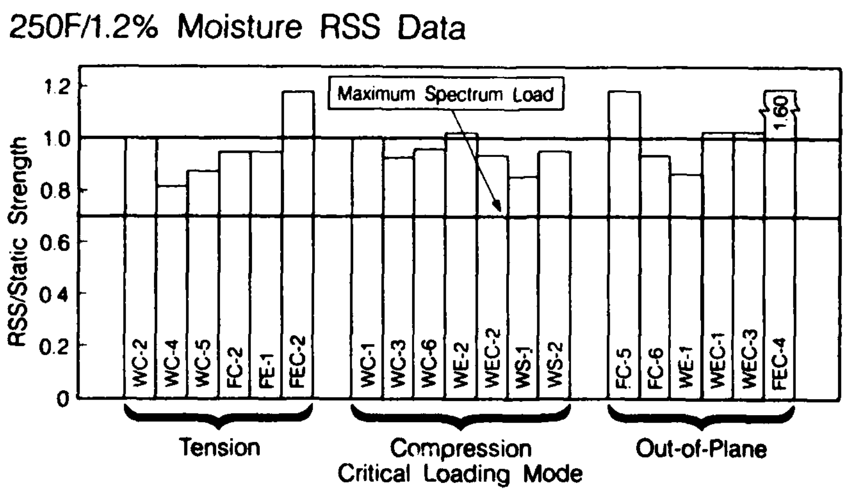


FIGURE 16. ENVIRONMENTAL SPECTRUM FATIGUE AND RESIDUAL STATIC STRENGTH RESPONSE OF COMPOSITE STRUCTURES

Figure 17 shows a comparison of the scatter in spectrum fatigue life observed in composites and aluminum. The scatter in life is inversely proportional to the Weibull shape parameter (α). That is, the higher the value

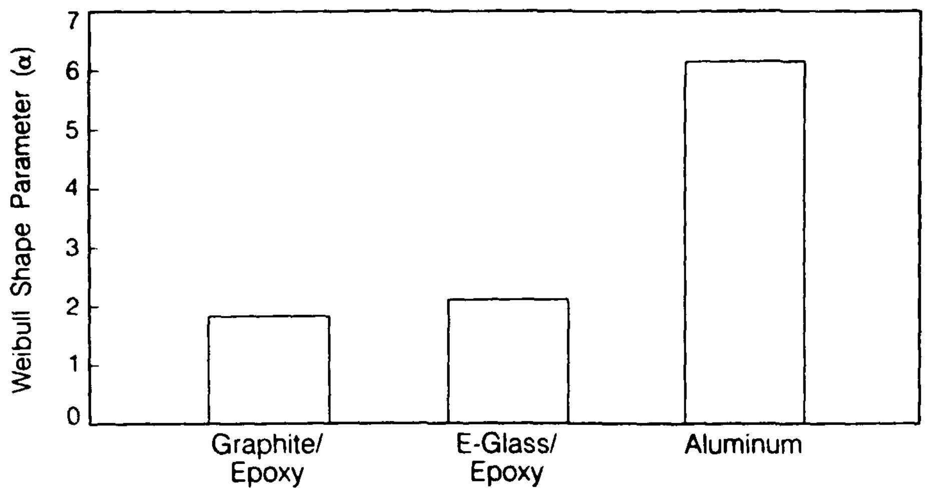


FIGURE 17. COMPARISON OF THE SPECTRUM FATIGUE LIFE DATA SCATTER OF COMPOSITES AND ALUMINUM

of α, the lower the scatter in fatigue life data. Figure 17 shows that composites exhibit significantly higher scatter than aluminum. This is caused by the significantly flatter S-N curves (superior fatigue resistance) observed in composites.

In metallic structures, it has been demonstrated that both fatigue initiation life and crack growth life are a function of load sequence. This load sequence dependence is caused by high loads producing residual compressive stresses, which reduce the fatigue damage accumulation rate. Historically, therefore, considerable attention has been given in metallic fatigue tests to careful simulation of the flight-by-flight loading history. In particular, the number of high loads included in the fatigue test spectrum has been a subject of concern. A common practice is to delete some high loads from the fatigue test spectrum in order to provide a conservative fatigue test, since retardation of crack initiation will be reduced by the omission of the high loads. In composite materials, no significant load sequence effect on fatigue life has been observed. However, studies on load spectrum variations have shown that composites are extremely sensitive to variations in the number of high loads in the fatigue spectrum. In contrast, truncation of low loads does not significantly affect fatigue life. These differences in load spectrum sensitivity may lead to contradictory load history requirements for a

mixed composite/metal fatigue test. For example, removal of some high loads may be prudent for the metallic structure, while their removal may cause significant overestimation of composite fatigue life.

Although composites have outstanding fatigue resistance, they have exhibited some durability sensitivity. Durability is defined by the USAF as a measure of economic life. Adequate structural durability is assured by eliminating functional impairment during the life of the airframe. Functional impairment occurs when excessive repair or part replacement causes unacceptable economic burden. Thus, durability is an economic issue, not a safety issue.

The durability in-service experience with monolithic structures has been excellent. However, durability experience with thin-skinned honeycomb structure has been less satisfactory. The following problems have occurred:

1. Sensitivity to low-level impacts ($<$10 ft-lb), causing visible skin damage, nonvisible skin or core damage, accelerated moisture intrusion, and core corrosion
2. High repair frequency
3. Excessive part replacement.

These problems have caused unacceptable maintainability and supportability costs.

Lessons Learned

Two major lessons have been learned from our composite fatigue/durability experience. These are:

1. Composites have outstanding fatigue resistance. For realistic structural laminates in typical design applications, composite structures can be considered to be fatigue insensitive, if they possess adequate static strength.
2. Maintainability and supportability of thin-skinned honeycomb structures has been poor.

Certification Recommendations

Detailed recommendations, given in Reference 1, are summarized below.

Load Spectrum Simulation. The same general guidelines established for metallic structures should be used. The following recommendations are made for load spectrum simulation in composite fatigue testing:

1. High loads in the fatigue spectrum must be carefully simulated.

2. Low loads ($<$ 30 percent limit load stress) may be truncated to save test time without significantly affecting fatigue life.

Mixed composite/metal structure. Because of the superior fatigue performance of composites, a mixed composite/metal structure fatigue will essentially interrogate adequately only the metal structure. Thus, any potential "hot spots" in the composite structure may not be found.

Because of the potential inadequacy of full-scale tests on mixed composite/metal structures and also the natural reluctance to overdesign metal parts in a full-scale test structure, it will be necessary to validate the composite structure during the design development testing phase. However, the specimen complexity should be adequate to enable the performance of the full-scale structure to be correctly simulated. Validation of the composite structure using subcomponent tests can offer the following advantages:

1. The components may be chosen for test purposes to interrogate the composite structure only.

2. If environmental test conditions are required, it will be easier and cheaper to achieve in a component.

3. It may be possible to test more than one replicate and thus increase confidence in the data base.

4. The results can be utilized in the certification of the full-scale structure.

For component tests to achieve their objective, great care must be taken in getting the boundary conditions correct. In addition, eliminating

metal failure modes by overdesign or replacement must be carefully evaluated so that relative effects such as differential thermal expansion are not masked.

Environmental simulation. The environmental complexity necessary for fatigue design development testing will depend on the aircraft hygrothermal history. Three factors must be considered. These are: structural temperatures for each mission profile, the load/temperature relationship for the aircraft, and the moisture content as a function of aircraft usage and structure thickness. In order to obtain these data, it is necessary to derive real-time load-temperature profiles for each misson in the aircraft's history. These relationships will have a significant influence on the fatigue test environment, and are strongly dependent on the aircraft type, configuration, and mission requirements and must be carefully developed on a case-by-case basis. The structural material should be selected to meet these mission requirements using the criterion in Figure 13. If this is accomplished, hot/wet fatigue testing will be minimized. Material selections which lead to significant environmental fatigue test requirements should be a last resort.

Scatter. The large scatter in composite fatigue life data makes the traditional life factor approach used for metals impractical because equivalent composite life scatter factors are in the 20-70 range. Alternate approaches to account for scatter effects were evaluated in Reference 10.

The first approach was use of the load enhancement factor. The objective of this approach is to increase the applied loads in the fatigue certification tests so that the same level of reliability can be achieved with a shorter test duration. A schematic showing this approach is shown in Figure 18, where the fatigue life scatter represented is typical of that observed in composites. At one fatigue lifetime, a typical residual strength distribution is shown. If the maximum applied load in the fatigue test (P_F) is increased to the mean residual strength at one lifetime (P_T), then the B-basis residual strength of the structure would be equivalent to the design maximum fatigue stress. Thus, a successful fatigue test to one lifetime at applied stress (P_T) or a fatigue test to N_F would both demonstrate B-basis reliability. In addition, combinations of the load enhancement and fatigue life factors could also be used to demonstrate B-basis life. In order to use this approach with

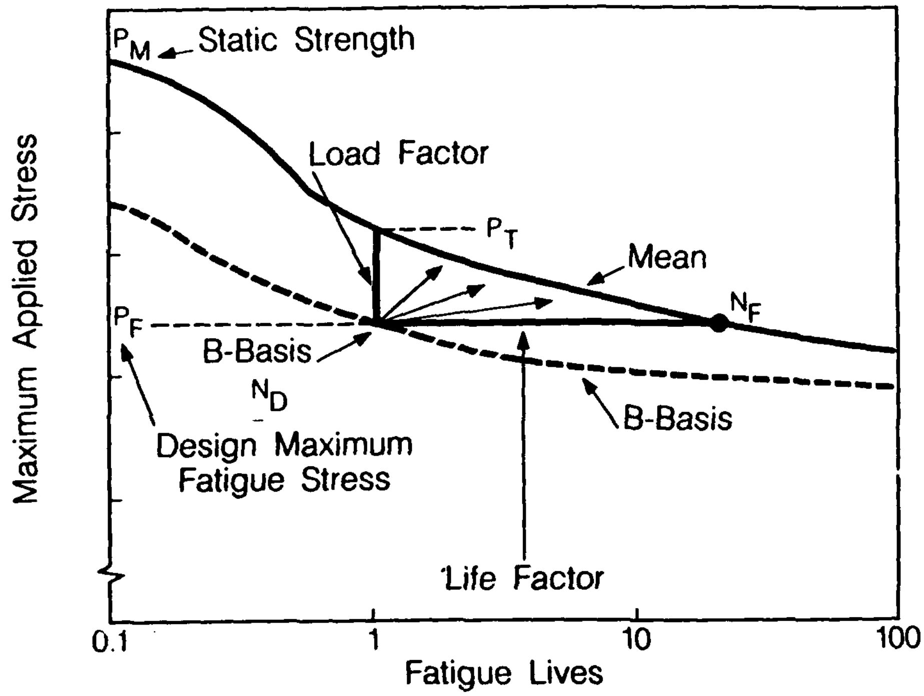


FIGURE 18. LOAD ENHANCEMENT FACTOR APPROACH

confidence in a certification methodology, a formal relationship between the load enhancement factor (LEF) and the life factor is required. This was verified mathematically in Reference 10.

While the evaluation of the enhanced loads approach in Reference 10 has shown that it has a sound theoretical basis and can be used with confidence for certification testing, some practical limits of this approach exist. First, for asymmetric spectra, the degree of load enhancement may be limited because of a requirement not to exceed ultimate load. Second, for mixed structures, the enhanced load approach will provide an excessively severe fatigue test for the metal parts.

A second approach takes advantage of the excellent fatigue response of composites and is summarized in Figure 19. The objective of this approach is to set fatigue stress allowables below the B- (or A-) basis fatigue threshold. This is possible in practice because composites have flat curves where the fatigue threshold is a high proportion of the static strength.

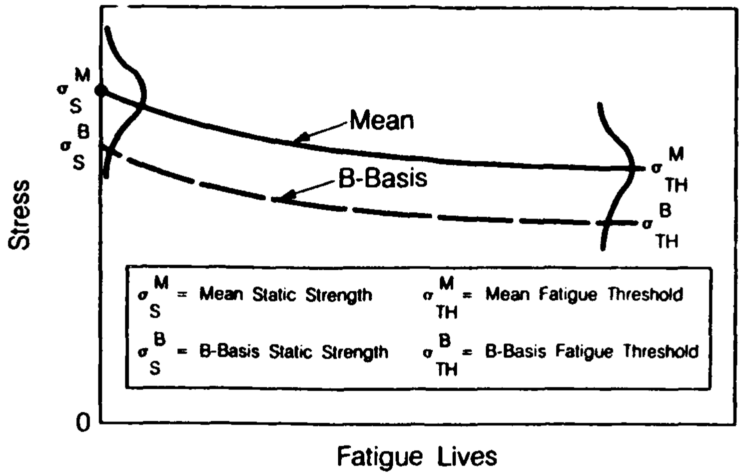


FIGURE 19. FATIGUE LIFE THRESHOLD APPROACH

Durability. The poor service experience with thin-skinned honeycomb composite structures has led the USAF to introduce draft low-level impact design requirements in Reference 3. These are summarized in Figure 20. The object of these requirements is to improve the maintainability of composite structures.

Zone	Damage Source	Damage Level	Requirements
1. High Probability of Impact	• 0.5-in. Diameter Solid Impactor • Low Velocity • Normal to Surface	• Visible • 6 ft-lb Max	• No Functional Impairment or Structural Repair Required for Two Design Lifetimes • No Water Intrusion • No Visible Damage From a Single 4 ft-lb Impact
2. Low Probability of Impact	• Same as Above	• Same as Above	After Field Repair of Visible Damage: • No Functional Impairment After Two Design Lifetimes • No Water Intrusion

FIGURE 20. PROPOSED USAF LOW-LEVEL IMPACT CERTIFICATION REQUIREMENTS (REFERENCE 3)

Full-Scale Test. The work in Reference 1 and other USAF-sponsored programs have shown that composites possess excellent durability. In

particular, the extensive data base developed in Reference 1 showed that composite structures, which demonstrated adequate static strength, were fatigue insensitive.

Thus, it is recommended that no durability full-scale test is required for all composite structures or mixed composite/metal structures with non-fatigue critical metal parts, provided the design development testing and full-scale static test are successful. For mixed structure, with fatigue critical metal parts, a two-lifetime ambient test will be required for fatigue validation of the metal parts.

DAMAGE TOLERANCE

Experience

Extensive work was conducted in Reference 2 to determine the influence of defects and damage on composite static strength and fatigue life. The results are presented in Reference 11, and are summarized in Figures 21-23. The data presented are representative of wing skin laminates fabricated from AS4/3501-6 and were obtained from 5-inch wide coupons.

A defect/damage severity comparison for compression static strength is presented in Figure 21. The plot relates damaged static strength to defect/damage severity. The data are also compared to the strength reduction for a 1/4-inch-filled unloaded hole. Porosity up to two percent, delaminations up to 1.5-inch diameter, and surface scratches are less than or equal to the strength reduction caused by a 1/4-inch hole. Fastener holes with delaminations around them show negligible strength loss compared to an unflawed fastener hole. In contrast, blunt impact damage causes a strength loss which significantly exceeds that of a 0.25-inch hole. Severe blunt impacts reduce strength by up to 60 percent; this is greater than the strength reduction of a 1.0-inch open hole. Clearly, therefore, impact damage is the most severe damage type for static compression strength.

A defect/damage severity comparison for compression-compression fatigue loading is shown in Figure 22. The material system is T300/5208, except for the porosity data which are the AS/3501-6 material system. The fatigue data show the same defect/damage severity trends as those observed

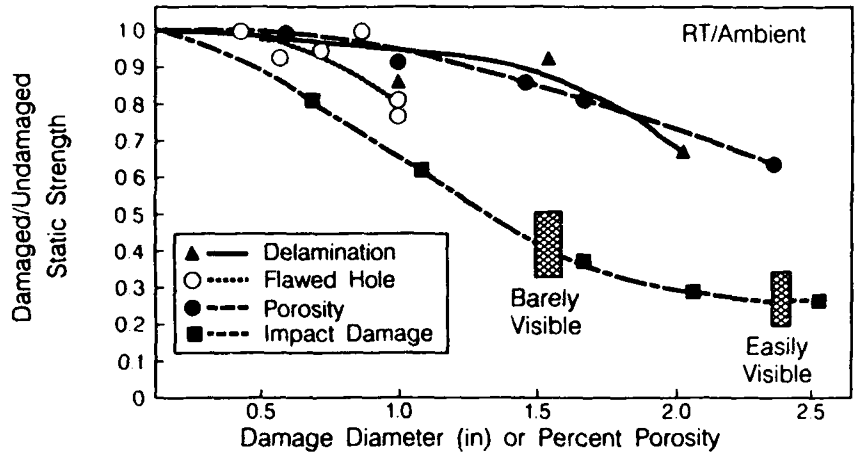


FIGURE 21. COMPRESSION STATIC STRENGTH DEFECT/DAMAGE SEVERITY COMPARISON

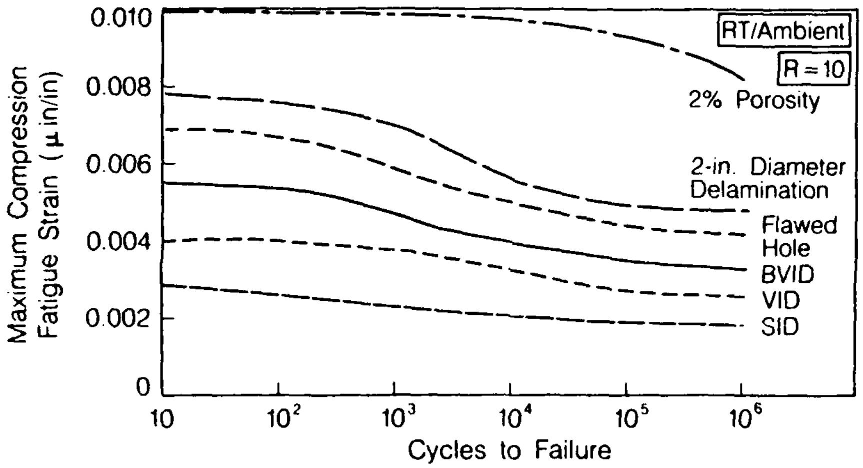


FIGURE 22. COMPRESSION FATIGUE LIFE DEFECT/DAMAGE SEVERITY COMPARISON

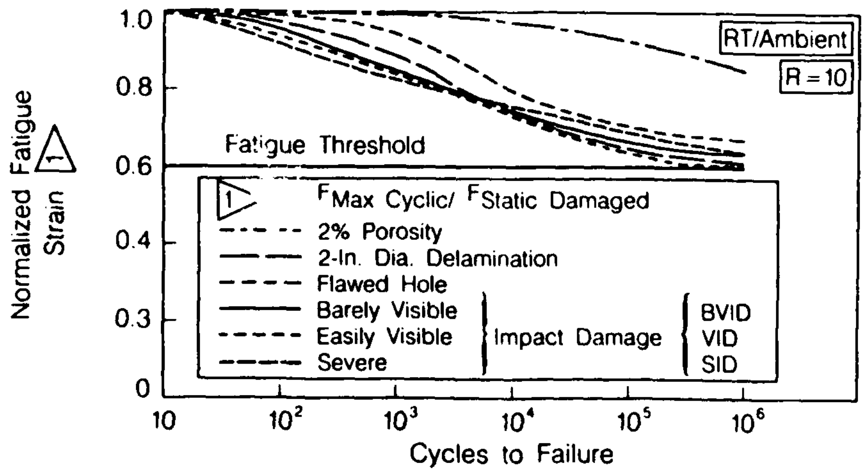


FIGURE 23. NORMALIZED COMPRESSION FATIGUE LIFE DEFECT/DAMAGE SEVERITY COMPARISON

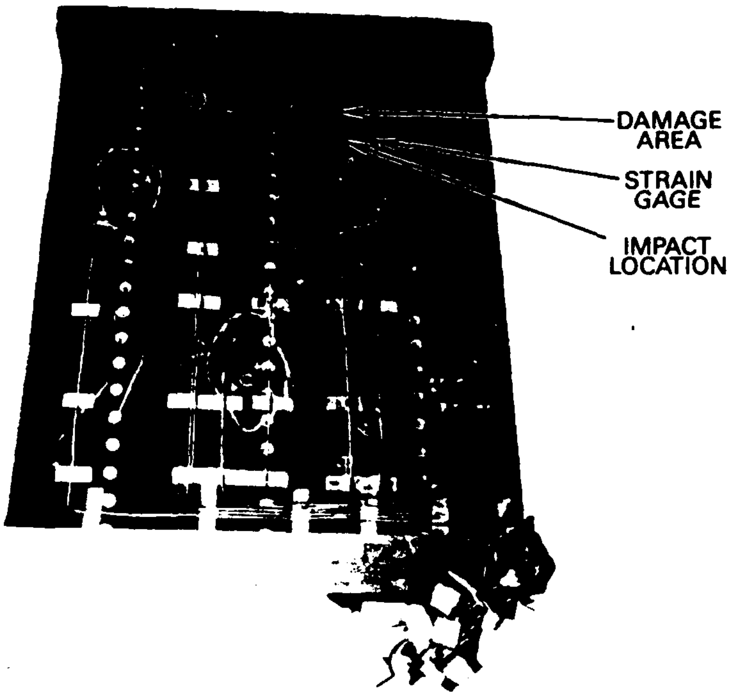
for static strength. Nonvisible and visible impact damage have the greatest fatigue sensitivity in terms of the fatigue strain required to give a life of 10^6 cycles. This is caused by the greater static strength sensitivity of these damage types.

The data in Figure 22 are replotted in Figure 23 in terms of normalized fatigue strain (maximum fatigue strain ÷ damaged static failure strain). These data show that, for all of the defect/damage types, a potential fatigue threshold (10^6 cycles) exists at 60 percent of damaged static strength for constant amplitude loading.

To check the applicability of these coupon data, extensive built-up structure damage tolerance testing was conducted in References 2 and 7. The work on 3-spar panels representative of fighter aircraft upper wing skin-to-spar attachments was summarized in References 2 and 11.

The specimen design is shown in Figure 24. The flat, stiffened panel specimen was loaded in compression through potted ends and was typical of fighter aircraft upper wing skin/spar attachment area. The skin was fabricated from 24 plies of double thickness AS4/3501-6 graphite/epoxy tape, which provides a nominal skin thickness of 0.25-inch. The channel spars were fabricated from 0.125-inch thick formed titanium. The specimen was designed to permit skin buckling to occur at approximately 5,000 μ in/in, which is typical of a fighter aircraft skin design.

Three spar test panels were impacted at 100 ft-lb at three skin locations, midbay, over the spar cap edge, and over the spar cap between fasteners. Maximum indentation depth on the impact surface was 0.05 inch. The influence of impact location on C-scan damage area is shown in Figure 25. Although some scatter is observed in the data, the mid-impact damage location clearly causes the largest damage area.



FIGU[illegible] 24. IMPACT DAMAGED 3-SPAR PANEL — 100 FT-LB

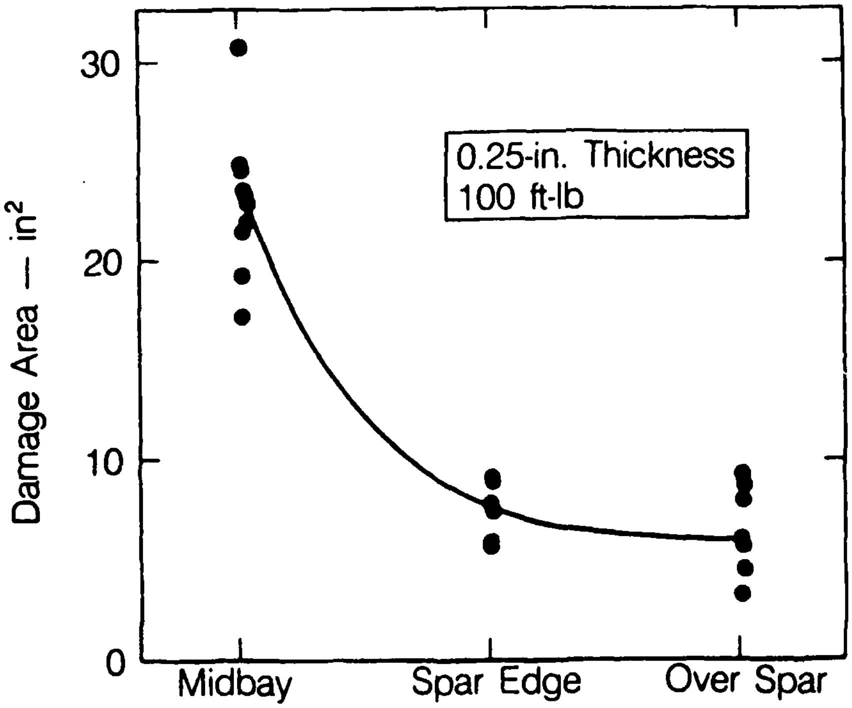


FIGURE 25. INFLUENCE OF IMPACT LOCATION ON C-SCAN DAMAGE AREA

Figure 26 shows Panel 37-1, which was subjected to a 100 ft-lb impact prior to static compression testing. Failure sequence was as follows. At a gross applied skin compression strain equal to 2,500 μin/in, the central region of the midbay impact damage area propagated rapidly to the spar attachments and arrested. Additional loading to a gross skin strain equal to 3,770 μin/in caused catastrophic panel failure through the midbay impact damage. This distinct two-stage static failure process permitted a 50 percent additional load-carrying capacity after initial failure. Further replicate tests again showed a 50 percent additional load-carrying capability after initial failure and arrest.

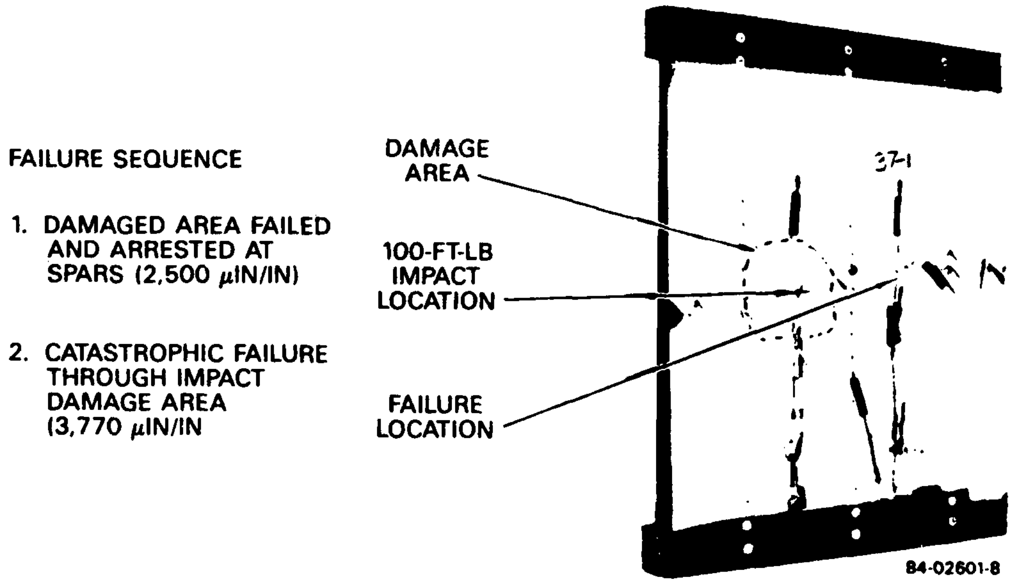


FIGURE 26. STATIC FAILURE OF IMPACT DAMAGED 3-SPAR PANEL — AS 4/3501-6

Additional tests were conducted on panels with midbay impacts ranging from 20 to 83 ft-lb. The results are summarized in Figure 27. All test panels (except 20 ft-lb impact damage) exhibited the two-stage failure sequence. The 20 ft-lb midbay damage panel exhibited only catastrophic failure (no arrest). Figure 27 shows that a significant difference exists between initial failure and final panel failure strains. Comparison with coupon test data in Figure 28 shows that initial failure and arrest in the built-up

panels correspond to catastrophic failure load in coupons. These data demonstrate significant a damage tolerance configuration scale up effect in built-up structures.

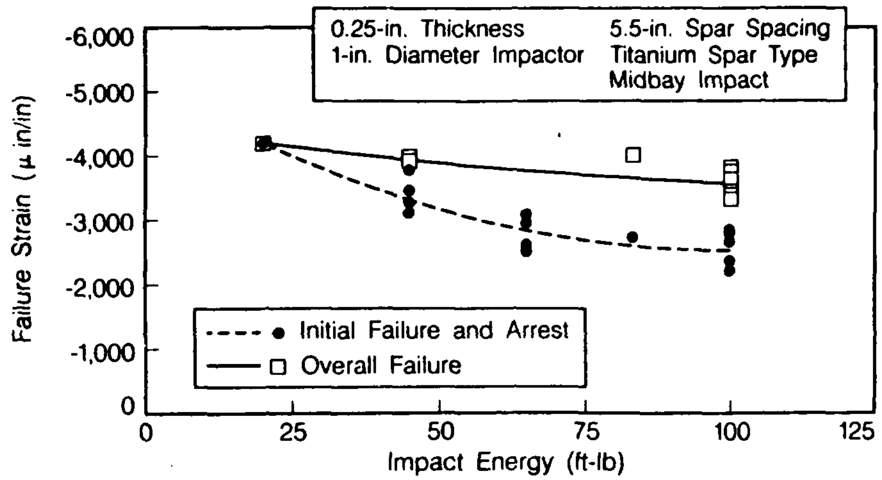


FIGURE 27. SUMMARY OF STATIC STRENGTH OF IMPACT DAMAGED 3-SPAR PANELS — AS4/3501-6

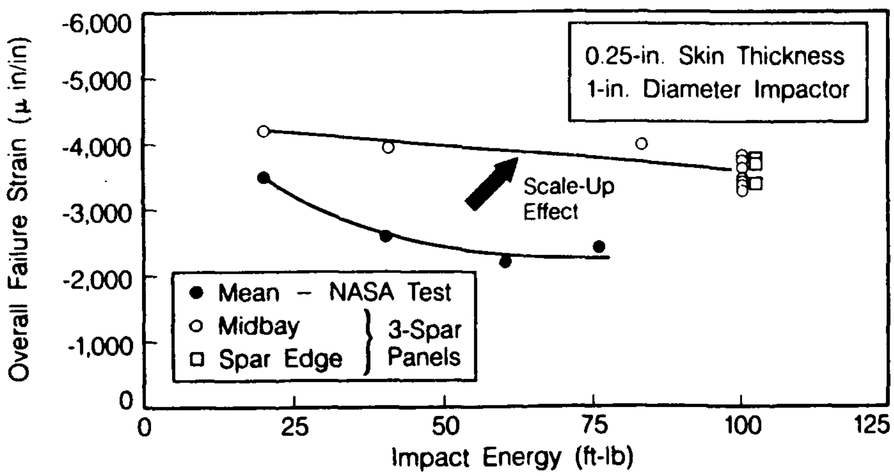


FIGURE 28. IMPACT DAMAGE TOLERANCE SCALE-UP EFFECT IN BUILT-UP STRUCTURE — AS4/3501-6

Lessons Learned

The lessons learned for composite damage tolerance are presented in Figure 29. These lessons learned highlight a conceptual difference in damage

tolerance certification for composites and metals. Figures 30 and 31 show the non-inspectable slow damage growth concept for metals and composites, respectively. For metals (Figure 30), residual strength decreases gradually over the aircraft service life as a fatigue crack initiates and grows. Thus, the exposure time where residual static strength is degraded is a small percentage of the total service life. In contrast for composites (Figure 31), residual strength degradation is not gradual, but takes place as a sudden large strength degradation. This occurs for two reasons. First, the impact event is random and can occur with equal probability on either the first or last day of the aircraft service life. Second, the impact event causes an immediate reduction of static strength. This leads to a potentially large exposure time in the degraded strength condition. Figure 32 summarizes this difference for composites and metals.

- Impact Damage Is the Most Severe Defect/ Damage Type
- Impact-Damage Areas and Static Strength Are Strongly Dependent on Structural Configuration
- Failure Modes of Impact-Damaged Built-Up Structure Are Significantly Influenced by Structural Configuration
- Significant Impact-Damage Tolerance Scale-Up Effects Exist for Built-Up Structure
- Impact-Damaged Structures Are Insensitive to Fatigue Loading

FIGURE 29. SUMMARY OF COMPOSITE DAMAGE TOLERANCE LESSONS LEARNED

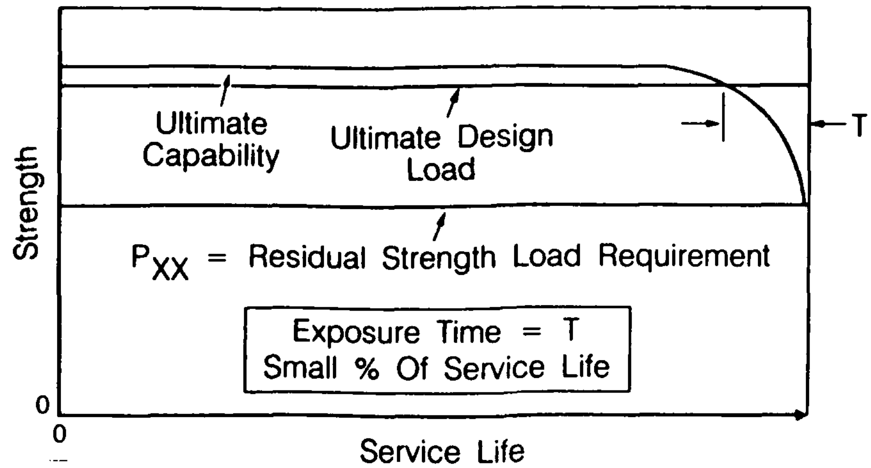


FIGURE 30. METALLIC NON-INSPECTABLE SLOW DAMAGE GROWTH

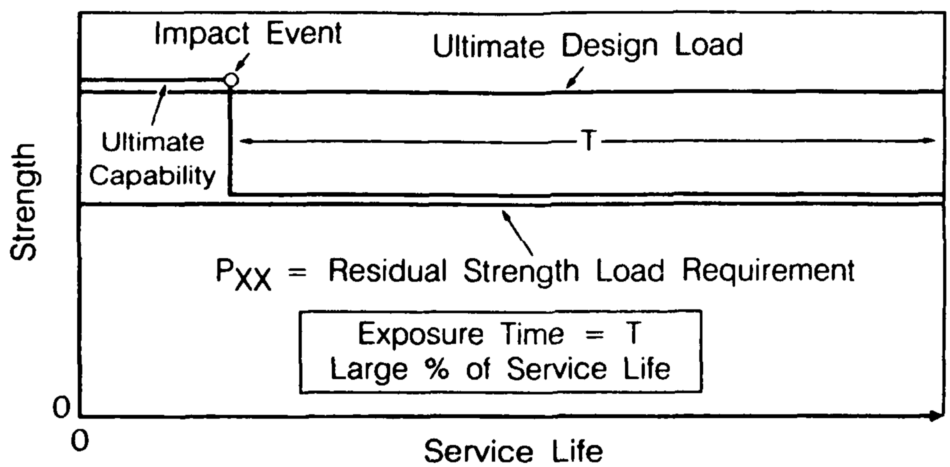


FIGURE 31. COMPOSITE NON-INSPECTABLE SLOW DAMAGE GROWTH

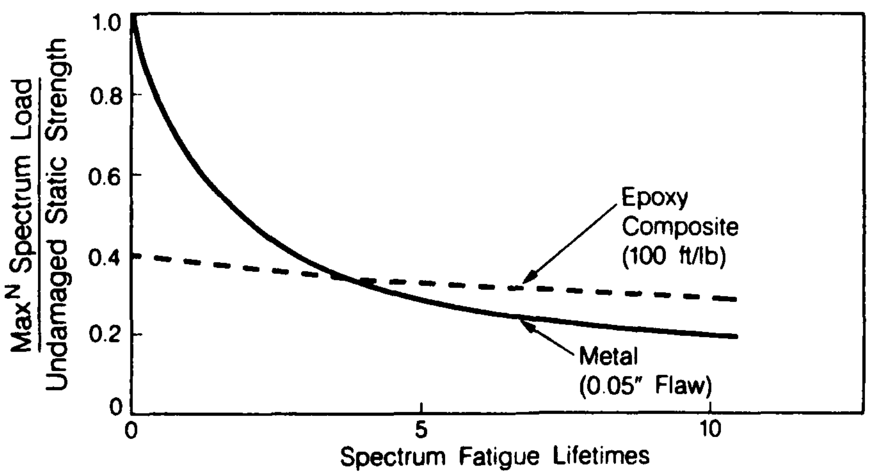


FIGURE 32. COMPARISON OF COMPOSITE AND METAL DAMAGE TOLERANCE

Certification Recommendations

The unique features of composite damage tolerance were recognized when draft USAF damage tolerance design requirements were developed in Reference 2. The highlights of the draft requirements are presented in Figure 33 and discussed in Reference 3. The damage assumptions in the draft requirements are presented in Figure 34. In practice, the impact damage requirement dominates design since it is the most severe. Figure 35 summarizes schematically the impact damage requirements. Two cut-offs were used: first, an impact energy cut-off equal to 100 ft-lb, which represents, conceptually, a tool-box dropped on its corner from approximately three feet; and second, a visibility cut-off at 0.10-inch dent depth, which represents, conceptually, damage detectable in a visual inspection. Figure 35 shows that the requirements do not potentially cover all non-visible damage; however, the 100 ft-lb impact is considered a conservative and potentially rare event (once per lifetime per aircraft fleet).

- Conceptually Equivalent to MIL-A-83444
- MIL Prime Format per MIL-A-87221
- Recognition of the Unique Property Characteristics of Composites
- Composite Defect/Damage Assumptions Significantly Different From Metals

FIGURE 33. HIGHLIGHTS OF DRAFT USAF COMPOSITE DAMAGE TOLERANCE DESIGN REQUIREMENTS

Flaw/Damage Type	Flaw/Damage Size
Scratches	Assume the Presence of a Surface Scratch 4.0-Inch Long and 0.02-Inch Deep
Delamination	Assume the Presence of an Interply Delamination That Has an Area Equivalent to a 2.0-Inch-Diameter Circle With Dimensions Most Critical to Its Location
Impact Damage	Assume the Presence of Damage Caused by the Impact of a 1.0-Inch-Diameter Hemispherical Impactor With 100 ft-lb of Kinetic Energy or With That Kinetic Energy Required To Cause a Dent 0.10-Inch Deep, Whichever Is Least

FIGURE 34. DAMAGE ASSUMPTIONS IN DRAFT USAF DAMAGE TOLERANCE DESIGN REQUIREMENTS

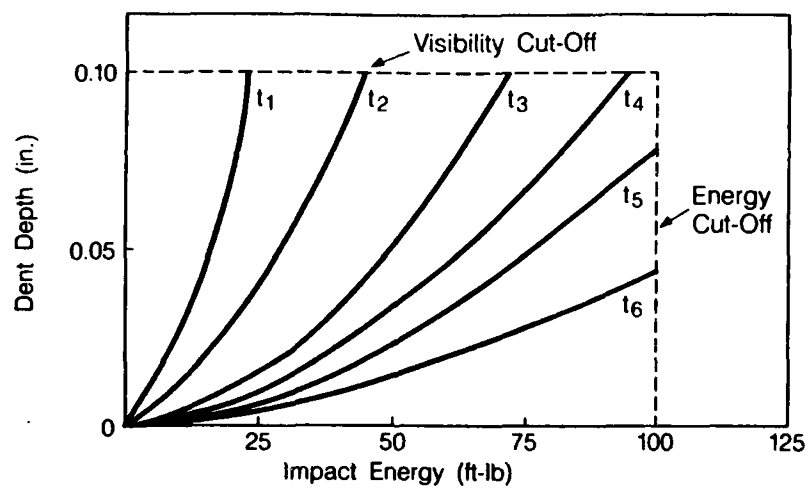


FIGURE 35. SUMMARY OF IMPACT DAMAGE ASSUMPTIONS IN DRAFT USAF DAMAGE TOLERANCE DESIGN REQUIREMENTS

The recommended compliance approach for the draft requirements is summarized in Figure 36. First, no significant damage growth is permitted in two design lifetimes. This is recommended because damaged composites have extremely flat S-N curves (Figure 22) and exhibit rapid unstable growth after growth initiation. Thus, it is not possible to control composite damage tolerance using the metal damage growth and inspection philosophy. An advantage of this compliance approach is that it eliminates inspection requirements.

- No Significant Damage Growth in Two Design Lifetimes
- No In-Service Inspections Required
- No Full-Scale Test Validation Required

FIGURE 36. RECOMMENDED COMPLIANCE APPROACH FOR THE DRAFT USAF DAMAGE TOLERANCE REQUIREMENTS

Finally, no full-scale test validation is required for composite damage tolerance certification. This is recommended because extensive testing in References 2 and 7 has shown that subcomponent validation tests accurately represent full-scale composite damage tolerance behavior.

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DEVELOPMENT OF A GRAPHITE/BISMALEIMIDE LEADING EDGE FOR THE F-111 (EF-111A) HORIZONTAL STABILIZER

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Abstract

The F-111 (EF-111A) horizontal stabilizer leading edge was selected as the baseline component to demonstrate the superior supportability characteristics of composites. The production component is a full-depth aluminum honeycomb sandwich construction that has historically shown in-service supportability problems. The composite leading edge (22.4 square feet in plan area) was designed as a form, fit and function replacement for the existing aircraft. Because of the aircraft's leading edge high service temperature (317 °F) and humidity environment (83% relative humidity (RH)), skins and stiffening members were designed of graphite/bismaleimide (Gr/BMI). At equivalent weight, acquisition and life cycle cost savings of 13% and 45% respectively, have been projected, for the composite leading edge over the existing aluminum honeycomb sandwich assembly.

1. INTRODUCTION

The objective of this on-going program is the development of a composite leading edge for the F-111 (EF-111A) horizontal stabilizer that displays optimum producibility, reliability, and maintainability characteristics. As such the program must demonstrate and validate low cost, innovative skin stabilization manufacturing methods for the leading edge, representative of secondary structure, which are cost-competitive with the full-depth honeycomb structure. Trade-off studies were initially performed to establish alternate designs, followed by in-depth cost/producibility/supportability analysis of the proposed composite structure compared to the existing component. Through the use of support oriented designs and durable, damage tolerant composites, we have shown enhanced producibility (at equivalent weight), for large scale integral composite structure with lower acquisition costs, increased reliability (minimum of fatigue failures) and lower maintenance costs (no corrosion, accessibility and visibility for inspection and repair). Due to the aircraft's high service temperature and humidity environment (317 °F and 83% RH) the composite selected is graphite/bismaleimide (Gr/BMI).

2. STRUCTURAL CONFIGURATION

The structural configuration of the Gr/BMI replacement leading edge component for the F-111 horizontal stabilizer is shown in Fig. 2-1. The structure is a large biconvex component which exhibits many of the complexities associated with highly loaded secondary structure. The maximum overall length of the leading edge component is approximately 180 in. and the maximum width 28 in. The height of the rear beam at the inboard end is 5.48 in.

The configuration selected to replace the present aluminum honeycomb core sandwich construction is a multi-rib structure with the ribs spaced at approximately 5.25 in. The arrangement gives the minimum number of ribs for manufacturing simplicity, and produces a structure comparable in weight to the existing metal component (99 lb per aircraft). To achieve maximum cost savings, the leading edge component has been designed as a unitized single cure structure to minimize the assembly procedure. The design permits disassembly of the main beam and closure rib after the cure cycle to permit removal of the solid internal forming tools. These components, together with the metal splice plates, are reassembled mechanically to complete the assembly.

3. STRUCTURAL DESIGN

In the existing aluminum skin/aluminum honeycomb core leading edge, the honeycomb core supports local air load on the skin and provides a shear path to carry air load to the front beam of the main structural box of the horizontal stabilizer. Moments are resisted by in-plane upper and lower cover loads that are transferred through splice joints to the main box.

In the composite integrally molded design (Fig. 2-1), full depth ribs, typically spaced at 5.25 in., perform the functions of the honeycomb core in the existing design. Normal air load is supported by skin panels spanning from rib-to-rib by a combination of bending and membrane action. The ribs then transfer the air load shear to the main box while the bending moment is balanced by cover skin loads as in the existing design. All attachments and interfaces are identical with the existing aluminum leading edge configuration.

The composite design is based on the following criteria:

- Ply orientations restricted to the 0°, 90°, and ±45° family
- Laminate ply orientations selected to

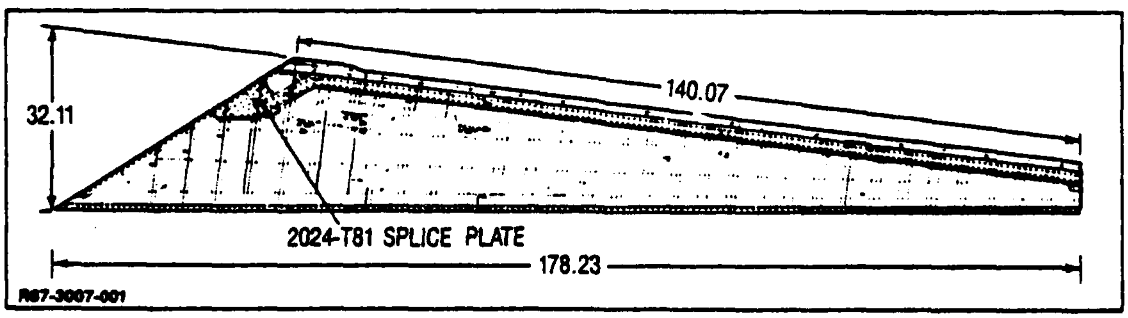

Fig. 2-1 Leading Edge Structure

provide filament-controlled load capability
- To reduce impact damage, a layer of glass/BMI fabric at 45° is placed at the inside surfaces of the cover skins
- All laminates are symmetrical about their midplane to minimize bending-stretching coupling effects
- All rib and beam webs have a large percentage of ±45 plies to minimize their axial stiffness (hence their induced axial load); and to provide maximum shear stiffness to minimize panel transverse shear deformation
- The two air passage skins meet or exceed the EI and GJ of the baseline aluminum skins
- The transverse strength of the skin panels is sufficient to distribute the normal air loads to the ribs
- Damage tolerance is explicitly addressed.

The upper and lower skin laminates consist of a 45° nickel-coated Gr/BMI air passage layer for lightning strike protection, an underlay of 0/90/±45 unidirectional Gr/BMI tape, and a back surface layer of style 112 fiberglass/BMI. The Gr/BMI rib webs are designed as constant thickness tapered panels that resist shear and crushing load. The forward beam closure member is designed to transfer shear loads from the rib webs to the main box through shear pins. Locations and pickups for the shear pins are identical to the existing design. For damage tolerance considerations, maximum fiber strains are limited to 2400 μin./in. at ultimate load.

Metal usage in the leading edge has been kept to a minimum. It is used only in those areas of the main torque box interface that experience high bearing stress and envelope interface restrictions. These parts are the upper and lower cover splice plates (Fig. 3-1), localized bearing plates on the main beam, and the closure rib main torque box interface. The forward beam bearing plates are fiberglass/BMI isolated aluminum, primarily to prevent the direct contact of the existing cadmium plating of the main box shear attachment pins with the graphite composite beam. (The use of cadmium or aluminum and graphite/epoxy in contact with each other results in galvanic corrosion.)

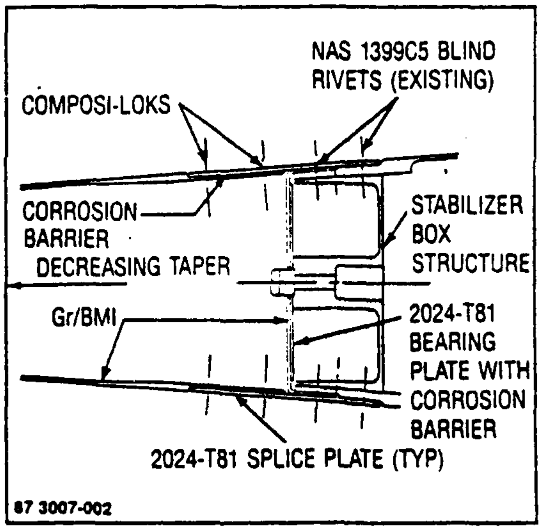

Fig. 3-1 Leading Edge Splice

The leading edge/main torque box interface hardware is the same as for the existing aluminum honeycomb design. The hardware selected to attach the composite structure within the assembly are Composi-Lok 130 degree flush and protruding head blind type fasteners. The tip of the leading edge, the arrowhead, is a supplemental subassembly for

supportability consisting of a metal glove and fiberglass-filled nylon (NYLAFIL) insert. This subassembly is subsequently mechanically fastened to the skin assembly, as shown in Fig.3-2.

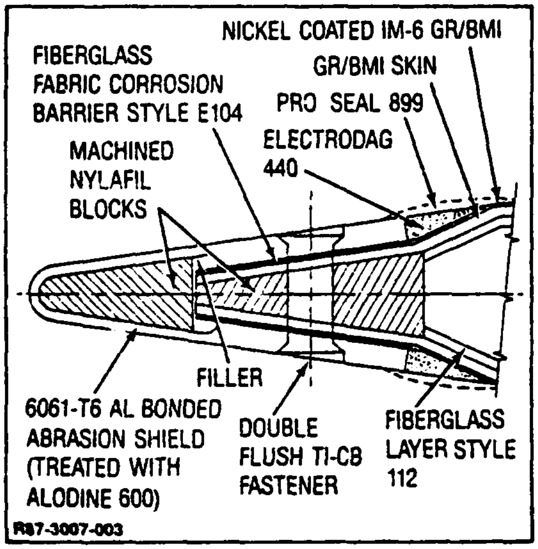

Fig. 3-2 Leading Edge Arrowhead Subassembly

4. MATERIAL DESIGN PROPERTIES

Three Gr/BMI material systems, IM-6/4001, IM-6/81-5 and IM-6/F650 were evaluated for use in the −67 °F to 317 °F wet (saturation at 83% RH) environment. The intermediate modulus fiber IM-6 with a Young's modulus of 40 x 10^6 psi was selected for the buckling critical design.

A test program involving over 375 coupons was performed to generate design properties for the three candidate material systems. The basic design properties obtained included: notched and unnotched tensile strength and modulus, notched and unnotched compression strength and modulus, shear strength and modulus, Poisson's ratio and bearing strength. The tests were conducted at −67 °F room temperature (RT), and the critical environmental condition 317 °F wet (saturated at 83% RH). The IM-6/F650 material system was shown to be superior in such key material properties as longitudinal compression modulus and strength, open hole tension and compression strength, and bearing strength. On the basis of these test results, IM-6/F650 (Hercules fiber and Hexcel resin) Gr/BMI was selected for fabrication of the F-111 (EF-111A) horizontal stabilizer leading edge.

5. QUALIFICATION TESTS

The leading edge is a secondary buckling critical structure, strength designed only in its mechanical attachment to the stabilizer torque box. As part of the Structural Test Plan for the F-111 (EF-111A) Horizontal Stabilizer Composite Leading Edge Component, tests were performed on single and combined load joint specimens. The purpose of the unidirectionally loaded multifastener joint elements was to verify details of the design concept for attaching the leading edge to the main box of the F-111 horizontal stabilizer. These included the fastener diameter and pitch, the allowable bearing load of the countersunk fastener, and load transfer between elements of the joint. The single fastener combined load joint specimens addressed the Mid-Span and Outboard areas critical design condition, and the critical thermal effects in the splice region.

5.1 Multifastener Joint Element

The specimens are 2.63 in. wide with the 3/16 in. diameter, 100° countersunk (on the aluminum) shear head Composi-Lok blind fasteners located at 1.13 in.

on center. The composite laminate edge distance from the centerline of the fastener is 0.56 in. The 2024-T81 aluminum splice plate is 0.070 in. thick and the laminate is $(\pm 45/90_2/\pm 45/0)s$ with a layer of 112 style fiberglass on each surface. The overall length of the specimen is 12.25 in. Figure 5-1 shows leading edge joint elements tested at RT and 317 °F wet.

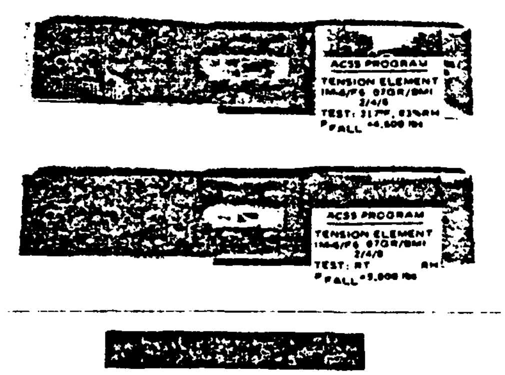

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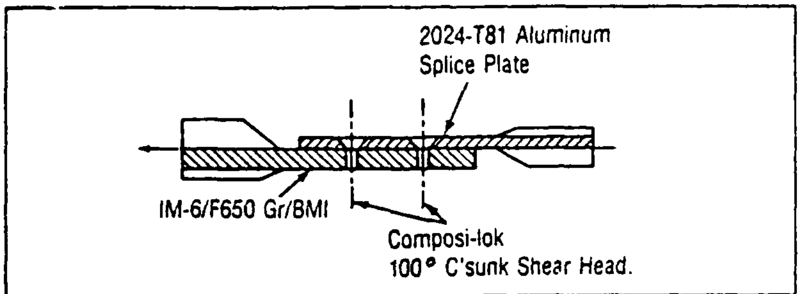
Fig. 5-1 Leading Edge Joint Elements After Test (Bearing Failures)

Three tension elements were tested at –67 °F, room temperature, and 317 °F with 83% RH. The specimens tested at –67 °F failed in bearing at 99% of the predicted load in bearing, but 105% of the predicted critical failure load in tension. The specimens tested at RT failed in bearing. The average failure load of these specimens was 101% of the predicted critical value. The three specimens tested at 317 °F and 83% RH failed in bearing; average failure occurred at 154% of prediction. Figure 5-2 lists the leading edge joint element analytical predictions versus test results.

5.2 Combined Load Single-Shear Joint Element

In the mid-span and outboard areas where the spanwise loads are significant, due to the deflection of the torque box, the behavior of the Gr/BMI laminate is such that, for the critical design condition, there is coupling between the chordwise bearing/bypass load and the spanwise load. To validate the splice joint design a representative element at the critical splice location (Rib No. 20) for Condition HT-4 (317 °F and 83% RH) was designed. The specimen consists of a 1.75 in. wide by 0.070 in. thick 2024-T81 aluminum plate fastened to a 20.0 in. long by 4.0 in. wide IM-6/F650 laminate $(\pm 45/0_2/\pm 45/90/90)s$, oriented as shown in Fig. 5-3 using a 100° countersunk 3/16 in. diameter Composi-Lok fastener. Figure 5-4 shows two of the instrumented specimens. The bolt bearing angle is in the direction of the 0° plies in the laminate (21.7° to the specimen axis). Thermal effects critical in the splice area (Rib No. 7) were addressed by the same element as shown in Fig. 5-3 but with a bolt bearing angle calculated to correspond to the critical P_x and P_y combination (46.9° to the specimen axis). Figure 5-5 shows the test setup for the off-axis combined load joint tests. Strip heaters were used to raise the temperature of the moisture conditioned specimens to 317 °F. The moisture loss during testing as measured by travelers was approximately 0.1% by weight. From a saturated moisture content at 83% RH of 1.35% by weight before the test, down to approximately 1.25% by weight right after the test.



TEST TEMP	$P_{TEST}/P_{PREDICTED}$			FAILURE MODE
	ALUMINUM BEARING	COMPOSITE BEARING (C.B.)	COMPOSITE NET TENSION	
-67°F	1.06	1.01	1.06	C.B.
	0.99	0.94	0.99	C.B.
	1.08	1.03	1.09	C.B.
AVG	1.04	0.99	1.05	
RT	0.95	0.96	0.94	C.B.
	1.02	1.03	1.01	C.B.
	1.03	1.04	1.02	C.B.
AVG	1.00	1.01	0.99	
317°F WET	0.92	1.51	0.90	C.B.
	0.94	1.54	0.92	C.B.
	0.95	1.57	0.93	C.B.
AVG	0.94	1.54	0.92	

Fig. 5-2 Leading Edge Joint Element Analytic Prediction vs Test Results

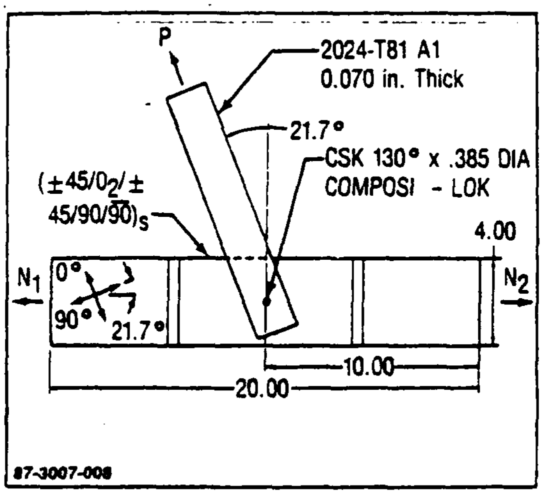


Fig. 5-3 Combined Load Joint Test Specimen (Bolt Bearing Angle 21.7°)

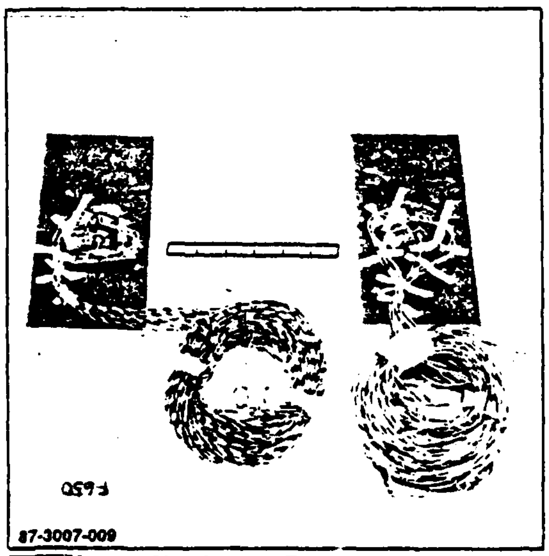


Fig. 5-4 Instrumented Joint Specimens

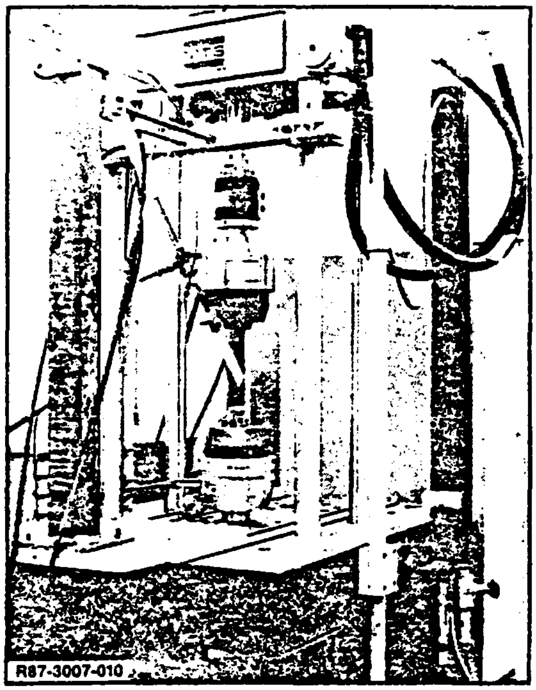


Fig. 5-5 Test Setup for Combined Load Joint Tests

The three specimens representative of the mid-span and outboard areas (Rib No. 20) failed in tension, as predicted by the computer program HOLES, at an average bypass load of 15,870 lb and a bearing load of 355 lb. These loads transformed in the laminate axis are shown in Fig. 5-6 as ratios of test results to design requirements. The comparisons are quite favorable.

The specimens failed on average at 138% of design ultimate load (DUL). Figure 5-7 shows a plot of the bearing bypass interaction curve for the critical splice location at Rib No. 20 as predicted by HOLES. The test data multiplied by a 0.8 design factor and the DUL condition are also shown.

Results of the specimens tested to address the critical thermal effects in the splice region (Rib No. 7) are shown in

SPECIMEN ID	TEST DATA/DESIGN REQUIREMENTS IN LAMINATE AXES		
	$\frac{N_{XTEST}}{N_{XDUL}}$	$\frac{N_{YTEST}}{N_{YDUL}}$	$\frac{P_{XTEST}}{P_{XDUL}}$
-5A	2.30	1.35	1.58
-6A	2.36	1.39	1.62
-7A	2.36	1.39	1.62
AVG	2.34	1.38	1.61

- ($\pm$45/0/$\pm$45/90/$\overline{90}$)$_S$ IM-6/F650 GR/BMI
- CONDITION HT-4 (317°F & 83% RH)

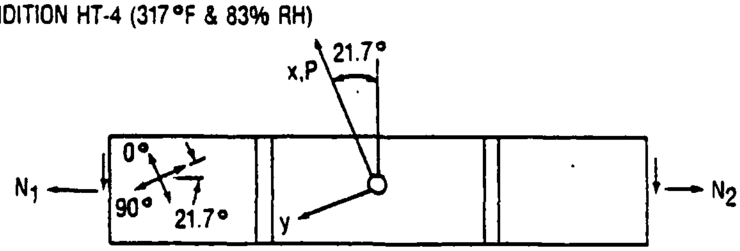


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Fig. 5-6 Design and Test Values for Critical Splice Location (Rib No. 20)

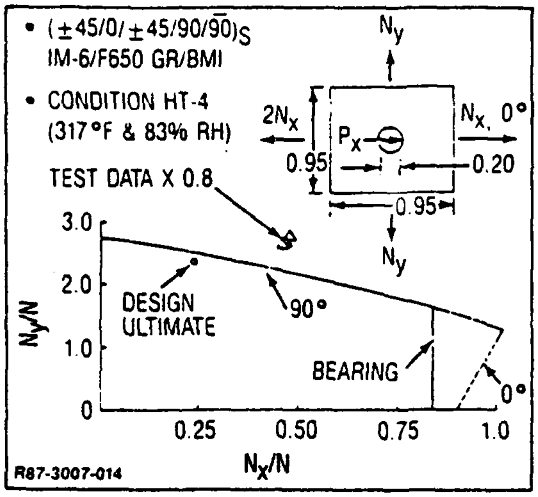

Fig. 5-7 Bearing Bypass Interaction Curve for Critical Splice Location (Rib No.20)

Fig. 5-8. These specimens failed in bearing at an average bypass load, N_1, of 10,170 lb and a bearing load, P, of 1097 lb. As seen in Fig. 5-8, the test data compares very favorably with the design requirements. The specimens failed in bearing at 127% of the DUL condition. A plot of the bearing bypass interaction curve for the critical splice location (Rib No. 7) generated using HOLES is shown in Fig. 5-9. The test data multiplied by a 0.8 design factor and the DUL requirements are also shown for comparison.

The successful completion of these tests allows for only a static test of the full size component at RT to qualify the leading edge for flight test.

CONCLUSION

An integrally-stiffened non-corroding composite leading edge for the F-111 (EF-111A) horizontal stabilator was designed. The design was achieved with production acquisition and life cycle cost savings and improved supportabil-

SPECIMEN ID	TEST DATA/DESIGN REQUIREMENTS IN LAMINATE AXES			
	$\frac{N_{XTEST}}{N_{XDUL}}$	$\frac{N_{YTEST}}{N_{YDUL}}$	$\frac{P_{XTEST}}{P_{XDUL}}$	$\frac{P_{YTEST}}{P_{YDUL}}$
-2	1.31	2.71	1.59	1.60
-3	1.25	2.58	1.52	1.53
-4	1.25	2.58	1.52	1.53
AVG	1.27	2.83	1.54	1.56

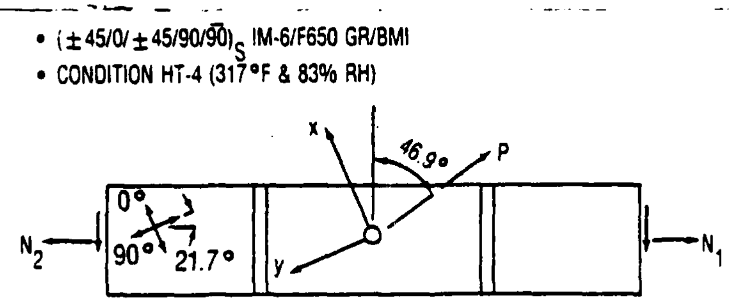

Fig. 5-8 Design and Test Values for Critical Splice Location (Rib No. 7)

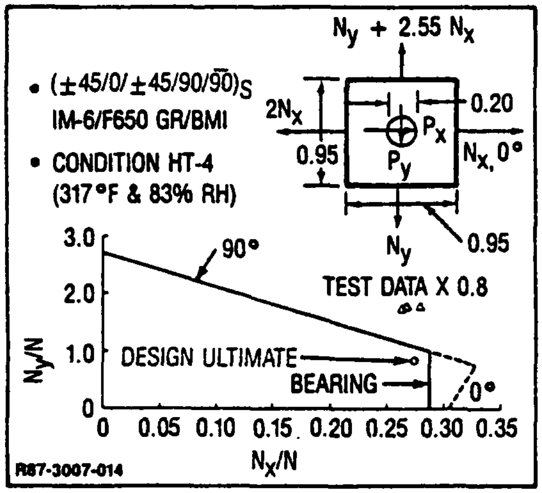

Fig. 5-9 Bearing Bypass Interaction Curve for Critical Splice Location (Rib No. 7)

ity. The component was designed

- without the use of honeycomb and its susceptibility to moisture intrusion, corrosion and subsequent delamination
- with sufficiently low strain levels to allow for the use of bolted repairs
- with penetrations to allow access to the substructure for NDI using fiber optics
- with integrally molded lightning strike protection
- with a new intermediate modulus graphite fiber and high temperature bismaleimide resin system.

ACKNOWLEDGEMENTS

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AN EXPERT SYSTEM ADVISOR FOR DAMAGE REPAIR OF COMPOSITE WING SKINS (REPAIRMAN)

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ABSTRACT

Temporary repair of battle-damaged composite wing skins is addressed by an expert system advisor that produces a bolted patch design suitable for the next mission requirements of the aircraft. The repair methodology utilized was developed under Air Force contract and addresses the temporary repair of damage holes up to seven inches in diameter using bolted patches of aluminum or steel sheets. The patch materials are available in the standard aircraft repair kit.

INTRODUCTION

New material systems, including composites, and sophisticated structural design techniques are combining to produce aircraft that are faster and more maneuverable than ever before. It is essential that there be damage repair procedures and methods available that are able to restore the performance of the airframe. Such repairs often require special training to install and, in addition, the repair must be developed by someone who has expertise in the design and analysis of bolted or bonded composite repairs. This presents particular problems in a battle damage situation when the repair needs to be made in a short amount of time and at a forward location where resources are limited.

Artificial Intelligence (AI) techniques and increasing computational capabilities are providing innovative solutions to many types of problems. Expert systems, in particular, allow computer programs to utilize rules of thumb, such as those used by experts, in reaching conclusions about the problem being

solved. It is now possible to build expert systems for focused problems that will run on small, yet powerful, microcomputer systems and in some cases to embed expert systems in the computers that are on the aircraft itself.

The REPAIRMAN system, described in this report, is a prototype system that demonstrates how computational and AI technologies can be brought together to provide an innovative solution to a supportability problem such as battle damage repair of composite wing skins. The methodology used in REPAIRMAN was developed under Air Force contract number F33615-83-C-3246 (References 1 and 2).

The system allows the user to input only the most top level information about the damage hole. This includes the shape, location, size, and orientation of the hole. The user must also select one of four available mission requirements for the subsequent use of the aircraft. This information is used, along with data from the system regarding the skin material and geometry, to arrive at a preliminary patch design. The patch design is analyzed using the BREPAIR program. BREPAIR is a FORTRAN analysis code that was developed under Navy contract number N62269-81-C-0297 (Reference 3). It is used to analyze bolted patch repairs of metal and composite substrates. The load and stress levels from the patch analysis are evaluated by the system and any needed modifications to the patch are determined. This new patch is then re-analyzed using BREPAIR and the cycle repeats itself until the system converges to a point where the design parameters are within a prescribed region close to but less than their respective allowables.

The expert system contains three basic sets of rules which provide knowledge for patch design and for interpreting the results from the BREPAIR routine. The first rule set evaluates the levels of the shear stress in the bolts, the strain in the skin, the stress in the patch, the bearing stress in the patch, and the bearing stress in the skin, and categorizes each of these five design parameters. The second set of rules considers the three design parameters associated with the bolted joints and determines an overall status for the joints. The third set of rules uses this result along with the status of the skin strain and the patch stress to determine the overall design quality and initiates the appropriate changes in the patch design if any are required.

The system makes use of a graphical interface to display both the damage hole on the wing as described by the user and to display the final patch design. This display supplements the description of the patch geometry and verifies that the input data is correct. The expert system was developed using the KEE software from Intellicorp on a Sun Microsystem 3/160 workstation, which provides a very useful environment for development. It is believed that ultimately such a system could reside on the aircraft as a part of a complete, on-board, structural integrity/repair expert system.

OVERALL PROGRAM STRUCTURE

A REPAIRMAN session begins with the system requesting information from the user about the damage to be repaired. This includes the size, shape, location and orientation of the damage hole. The damage can be either a circular or elliptical shape. The location is used to determine the wing skin thickness and the orientation is used to determine the stress concentration in the skin due to the presence of the hole. The user must also specify whether the hole is located in an upper or lower skin, which determines if the patch is for tension or compression loads.

The final information required are the requirements for the next mission of the aircraft. These determine the load level that the patch must be designed to carry. REPAIRMAN utilizes a graphical display of the wing, and the hole as described by the user, to provide visual verification of the input information to help minimize errors. The final patch design is also displayed graphically. The display resolution is currently low, and is only for illustration. A screen dump of the user interface is shown in Figure 1.

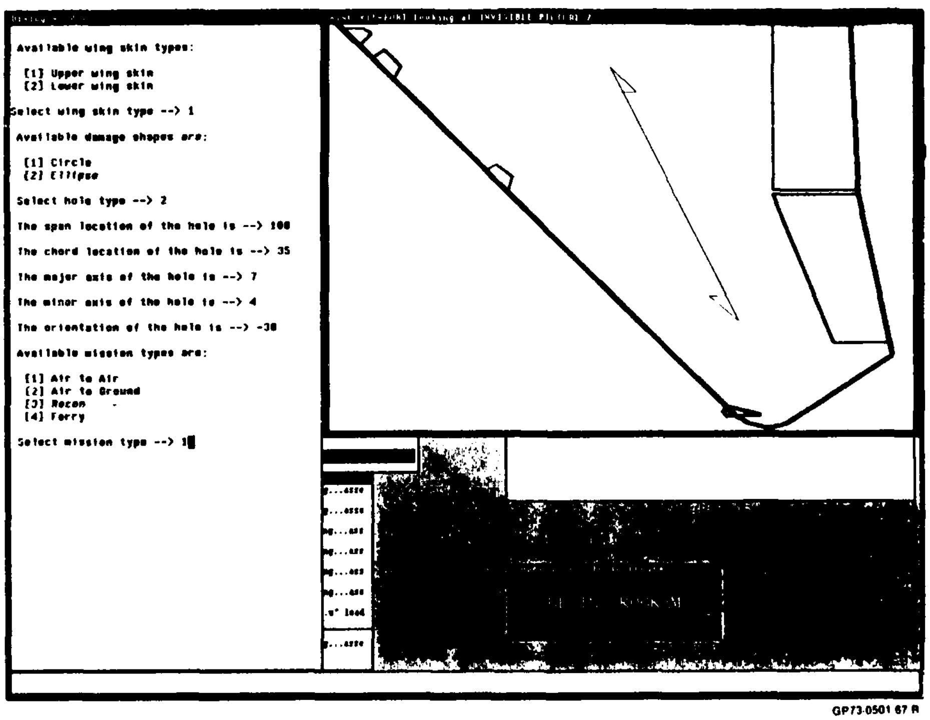


Figure 1. User Interface to REPAIRMAN

The first step in the solution procedure is for REPAIRMAN to determine what the effect of the hole is on the stress field in the skin. To accomplish this, BREPAIR is run for the hole with no patch present. The program provides the stress level at the

critical location on the hole outline. This value is used, along with the other properties to determine the patch material, fastener size, patch thickness and number of bolts for an initial patch configuration.

The initial patch proposal is analyzed using BREPAIR and the five design parameters are evaluated to determine the effectiveness of the patch design. These parameters are the stress in the patch, the strain in the skin, the maximum shear load in the bolts, the bearing stress in the skin and the bearing stress in the patch. Each critical parameter has an allowable value which determines an upper bound for that parameter. A tolerance factor of 0.8 is used to obtain a lower bound for the parameter. A parameter that is lower than the tolerance level is considered too low, one that is higher than the allowable is considered too high and one that is between the tolerance and allowable is considered OK.

Rules are present in REPAIRMAN that evaluate these five quantities and, if they are too high or too low, propose changes to the patch design and submit a revised design for analysis by BREPAIR. This process continues until an acceptable design is found. Figure 2 shows the structure and general flow of the REPAIRMAN system.

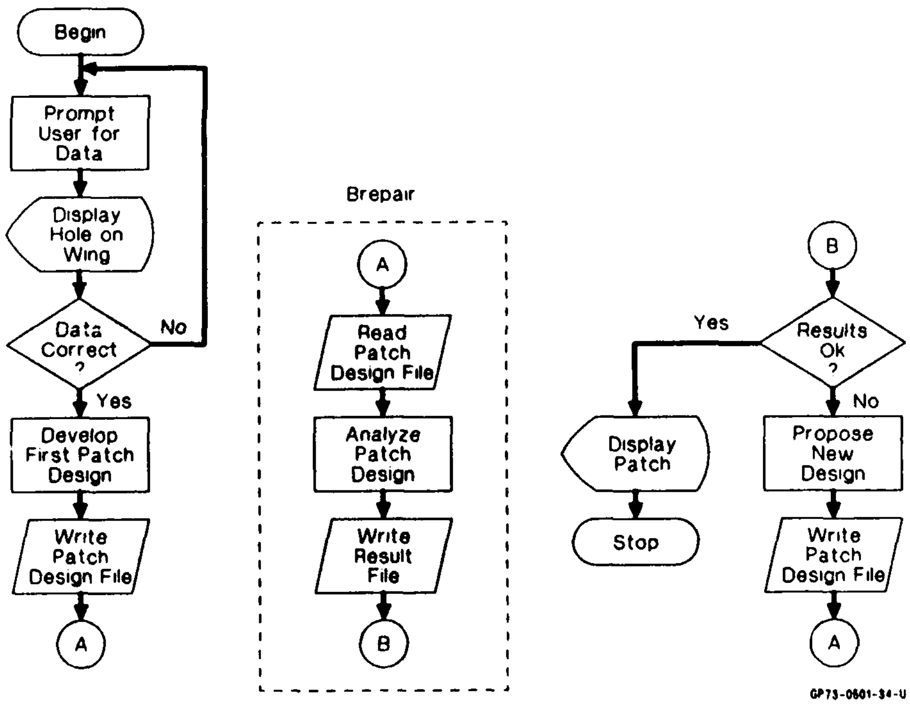


Figure 2. REPAIRMAN Flow Diagram

REPAIRMAN was developed on the SUN Microsystem 3/160 workstation using the Knowledge Engineering Environment (KEE) from Intellicorp. KEE provides a convenient method for developing artificial intelligence applications that combine object-oriented programming and rules. The objects in KEE are referred to as units and their characteristics are called slots. This terminology will be used when referring to objects in REPAIRMAN.

RULE SYSTEM

The REPAIRMAN system has 3 sets of rules that it uses to evaluate and propose changes to a patch design. These are called the Rank rules, the Joint rules and the Evaluation rules. The Rank rules are used to rank each of the 5 critical quantities as either too low, OK, or too high depending on where they fall relative to the allowable range for that parameter. The second rule set, Joint rules, evaluates the 3 quantities affecting the joints, the bolt shear, the bearing stress in the skin, and the bearing stress in the patch to determine an overall joint status. The status of the joints along with the skin strain level and the patch stress is used by the third rule set, the Evaluation rules, to determine what corrective action, if any, should be taken to improve the patch design. Five corrective actions can be taken: add or delete bolts, add or delete thickness on the patch, and increase the bolt diameter. These actions are usually applied singly, but two actions can be applied simultaneously in limited cases.

The KEE software system used to develop REPAIRMAN provides a very nice "english-like" format for expressing rules. Several typical rules from the Evaluation rule set are shown in Figure 3.

```
(EVAL.RULE.1
  (IF (THE RANK.JOINT.STATUS OF PATCH IS TOO.LOW)
      (THE RANK.SKIN.STRAIN OF SKIN IS TOO.LOW)
      (THE RANK.PATCH.STRESS OF PATCH IS TOO.LOW)
   THEN
      (CHANGE.TO (THE RULES.FIRE OF PATCH IS 1))
      (LISP (UNITMSG 'PATCHES 'DELETE.BOLTS))))

(EVAL.RULE.16
  (IF (THE RANK.JOINT.STATUS OF PATCH IS OK)
      (THE RANK.SKIN.STRAIN OF SKIN IS TOO.HIGH)
      (THE RANK.PATCH.STRESS OF PATCH IS TOO.LOW)
   THEN
      (CHANGE.TO (THE RULES.FIRE OF PATCH IS 16))
      (LISP (UNITMSG 'PATCHES 'THICKER.PATCH))
      (LISP (UNITMSG 'PATCHES 'ADD.BOLTS))))
```

Figure 3. Typical Rule Format

KNOWLEDGE BASE

REPAIRMAN's knowledge base is fairly straight forward with the 3 rule sets described above comprising the most significant part of it. There is also a class of units which contain the material properties that REPAIRMAN uses to develop the patch designs. REPAIRMAN works with only 3 objects in the knowledge base; the hole, the skin, and the patch. These three objects, however, have a fairly large number of slots (or descriptors) that define their state. These slots are listed in Tables 1 and 2. The slots are used to store the physical characteristics of the objects as well as the qualitative information about the status of the critical variables and the overall design.

ALLOWABLE.BOLT.SHEAR	PATCH.LENGTH
ALLOWABLE.PATCH.BEARING	PATCH.MATERIAL
ALLOWABLE.PATCH.BUCKLING	PATCH.STATUS
ALLOWABLE.PATCH.STRENGTH	PATCH.STRESS
BOLT.SHEAR	PATCH.THICKNESS
BOLT.TYPE	PATCH.WIDTH
CODE.COSMETIC.PATCH	RANK.BOLT.SHEAR
CODE.ERROR	RANK.JOINT.STATUS
CODE.NO.PATCH	RANK.PATCH.BEARING
CODE.PATCH.ANALYSIS	RANK.PATCH.BUCKLING
FOR.MISSION.TYPE	RANK.PATCH.STRESS
LOAD	RULES.FIRE
LOAD.TYPE	TOLERANCE
MINIMUM.BOLTS	TOLERANCE.BOLT.SHEAR
NUMBER.OF.BOLTS	TOLERANCE.PATCH.BEARING
PATCH.BEARING	TOLERANCE.PATCH.STRENGTH

Table 1. Slots Describing Patch Unit

A unit called Supervisor has slots that contain the method descriptions (LISP code) that perform many of the actions that REPAIRMAN takes during a problem solution. These methods are loaded with the knowledge base, but remain inactive until sent a message by the system.

HOLE SLOTS	SKIN SLOTS
LOCATION.CHORD	ALLOWABLE.SKIN.BEARING
LOCATION.SPAN	ALLOWABLE.SKIN.STRENGTH
MAJOR.AXIS	CODE.SKIN.ANALYSIS
MINOR.AXIS	RANK.SKIN.BEARING
ORIENTATION	RANK.SKIN.STRAIN
SHAPE	SKIN.BEARING
	SKIN.LOCATION
	SKIN.MATERIAL
	SKIN.STRAIN
	SKIN.THICKNESS
	TOLERANCE.SKIN.BEARING
	TOLERANCE.SKIN.STRENGTH

Table 2. Slots Describing Hole Unit and Skin Unit

SYSTEM LIMITATIONS

Limitations to the scope of the REPAIRMAN system came from two places. First, the original Air Force contract considered only battle damage repairs that could be built up from the materials available in the aircraft repair kit. These limitations are:

1) 1/4 and 5/16 inch diameter jo-bolts can be used to bolt the patch to the skin.

2) Aluminum sheet stock in 0.040, 0.050, 0.063, and 0.125 inch thicknesses is available.

3) Steel sheet stock in 0.016, 0.032, and 0.040 inch thicknesses is available.

4) The damage size is limited to a maximum major dimension of 7 inches.

Further constraints were introduced as a result of the intent to quickly produce a prototype system. Within that context further scoping was in order to limit the development to a manageable size. The limitations produced by the scoping process are:

1) The thickness of the wing skin is a uniform taper from 0.70 inches at the root to 0.10 inches at the tip.

2) Wing skin material is AS4/3501-6 Carbon-Epoxy composite.

3) The effects of the underlying substructure are not considered. In the original Air Force work (Reference 1) the bolt pattern determined for the patch had a 1.5" movement available to avoid such underlying structure.

PERFORMANCE OF REPAIRMAN

REPAIRMAN has designed patches for both tension and compression cases. The results of those cases compared well with the solutions developed by one of the experts who assisted in the development of the system. The patch designs compared well in both the number of bolts used, the size bolts used, the patch material used, and the thickness of the patch used. Table 3 shows a comparison of both a tension and compression patch design.

TENSION PATCH

HOLE: Elliptical
MAJOR AXIS: 6.0 in.
MINOR AXIS: 4.0 in.

LOAD: 4000 micro in./in.
ORIENTATION: 90.0 degrees
SKIN THICKNESS: 0.208 in.

	REPAIRMAN:	EXPERT:
NO. OF BOLTS	8	9
PATCH THICKNESS	0.06"	0.10"
PATCH MATERIAL	AL	AL

COMPRESSION PATCH

HOLE: Elliptical
MAJOR AXIS: 6.0 in.
MINOR AXIS: 4.0 in.

LOAD: 3300 micro in./in.
ORIENTATION: 90.0 degrees
SKIN THICKNESS: 0.208 in.

	REPAIRMAN:	EXPERT:
NO. OF BOLTS	14	12
PATCH THICKNESS	0.15"	0.16"
PATCH MATERIAL	AL	AL

Table 3. Comparison of Performance

AREAS FOR CONTINUED DEVELOPMENT

In working with REPAIRMAN, several areas were noted to need further investigation.

1) **ADDING BOLTS** - Currently bolts are added individually. Studies indicate that some bolt locations have less effect on the critical parameters than others. These locations are predictable and inclusion of this information will allow REPAIRMAN to make better decisions about how many bolts should be added to gain the needed change in the critical parameter. This will reduce the number of times that the BREPAIR program must be run and hence produces a solution much quicker.

2) **REFINEMENT OF RULES** - Further refinement of the rules is needed to produce a more detailed discrimination of the actions to be taken when the joint status is not acceptable. Currently the joint status is based on bolt shear, skin bearing and patch bearing stresses. When the joint status is not acceptable, a change is made based on which of these three conditions is critical. The rules could be refined to determine what action or combination of actions might better suit a particular combination of critical joint parameters. Although the system works in its present state, a more optimum patch design might be achieved with the benefit of this further discrimination.

3) **PARAMETRIC STUDIES** - Additional studies are needed with BREPAIR to determine how individual parameters affect the final solution. This additional information would perhaps allow REPAIRMAN to reach a more optimum design with fewer iterations through the BREPAIR program. The current system produces an acceptable patch design, but does not attempt to balance the 5 criticai quantities to produce an optimum patch design. The number of bolts that must be installed directly affects the amount of time required for the patch installation and thus it might be argued that reducing the number would be the best design. This type of approach might be achieved with a deeper understanding of how each of the parameters affects the solution.

REFERENCES

1) "Battle Damage Repair of Composite Structures", On-going Air Force Contract No. F33615-83-C-3246.

2) Hinkle, T. and Hoehn, G., "Verification of Analytical Methodology for Designing Repairs to Composite Skin, Vol. I and II", Report, Air Force Contract No. F33615-83-C-3246.

3) Bohlmann, R.E., Renieri, G.D., Horton, D.K., "Bolted Repair Analysis Methodology", Final Report, N62269-81-C-0297, May 1982.

C-141 Repair of Metal Structures by Use of Composites

Presentation at USAF ASIP Conference

San Antonio, Texas December 1-3, 1987

J.B. Cochran, Lockheed-Georgia Divison

Dr. Tom Christian, WR-ALC/MMSRD

D.O. Hammond, WR-ALC/MMSRD

GA-7353-1

C-141 Repairs Using Composite Materials

Background

- Technology Pioneered by Australian Research Labs (ARL) for RAAF C-130, Mirage, F-111, MACCHI Aircraft
  - Over 500 Repairs
  - Excellent Results
- WR-ALC Study Contract to Lockheed 1986-1988
  - Develop Composite Repairs Concepts for Five Structural Locations
    - Variety of Designs and Applications
    - Limited Hardware Development To Probe Concepts
- Presentation Addresses Technology Aspects and In-Service Opportunity

GA-7353-2

BACKGROUND

REPAIR OF METAL STRUCTURES BY USE OF COMPOSITES IS A TECHNOLOGY PIONEERED BY THE AUSTRALIANS FOR USE ON RAAF C-130 AND OTHER AIRCRAFT. OVER 500 REPAIRS HAVE BEEN SUCCESSFULLY APPLIED ON STRUCTURES SUBJECT TO CORROSION AND FATIGUE CRACKING. AFTER LOCKHEED (GEORGIA DIVISION) BECAME INTERESTED IN THIS TECHNOLOGY, WR-ALC AWARDED A STUDY CONTRACT TO DEVELOP REPAIRS FOR SELECTED LOCATIONS IN THE C-141 AIRCRAFT.

THIS PRESENTATION TODAY ADDRESSES THE TECHNOLOGICAL ASPECTS WE HAVE LEARNED TO DATE AND WILL INCLUDE THE FIRST IN-SERVICE REPAIR APPLIED TO A USAF AIRCRAFT.

C-141 Repairs Using Composite Materials

Benefits of Composites Repairs

- Materials Related
  - High Stiffness/Strength of Fibers Permits Thinner Repair Members
  - Ability To Tailor Strength/Stiffness to Direction of Loads
- Application Related
  - Highly Effective for Stress Corrosion Cracks
  - Advantages in Fatigue Due to Bonding in Lieu of Fasteners
- Economics
  - Easy To Mold to Complex Shapes
  - Reduced Repair Time

GA-7353-3

BENEFITS OF COMPOSITE REPAIRS

THIS SLIDE ADDRESSES THE PRACTICAL BENEFITS OF COMPOSITE REPAIRS ON METAL STRUCTURE FROM THE STANDPOINT OF

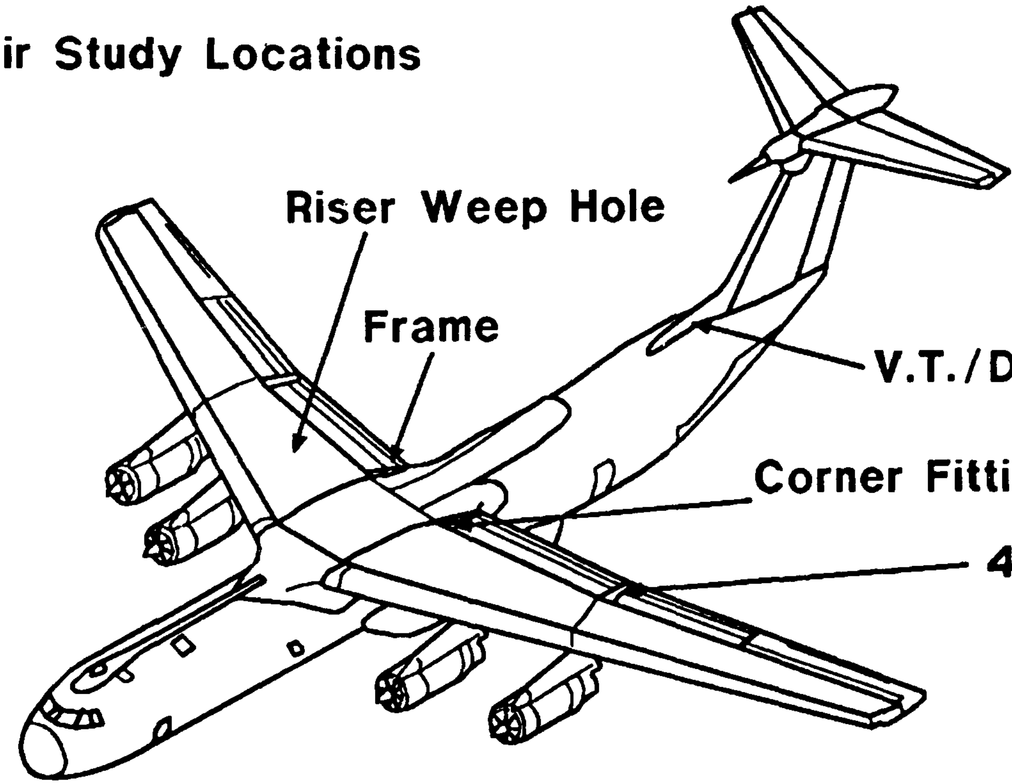
1) MATERIALS THAT PROVIDE HIGH STIFFNESS/STRENGTH FOR SMALLER REPAIR MEMBERS.

2) APPLICATION IN AREAS SUSCEPTIBLE TO CORROSION AND FATIGUE CRACKING WHERE BONDING IN LIEU OF FASTENERS PROVIDES ADDITIONAL REPAIR LIFE.

3) ECONOMICS DUE TO EASE OF FABRICATION AND REDUCED AIRCRAFT DOWN TIME FOR REPAIR.

C-141 Repairs Using Composite Materials

Repair Study Locations



GA-7353-4

THE FIVE LOCATIONS BEING STUDIED ARE SHOWN ON THIS DRAWING.
SKETCHES OF THE INDIVIDUAL SITES FOLLOW.

Feasibility Study Locations

Location	Characteristics
WS405 Joint	External Boron Doubler Bridges "Kinked" Joint*
Wing Riser (Weep Hole and Rib Clip Attachment)	Straightforward; Boron or Graphite; JP-4 Environment
WS77 Inner Wing Corner Fitting	Difficult Geometry. Study In-Place Molded Angle; Graphite; JP-4 Environment; End Fastener Effect Relief

*Major Effort to Date

GA-7353-5

Feasibility Study Locations

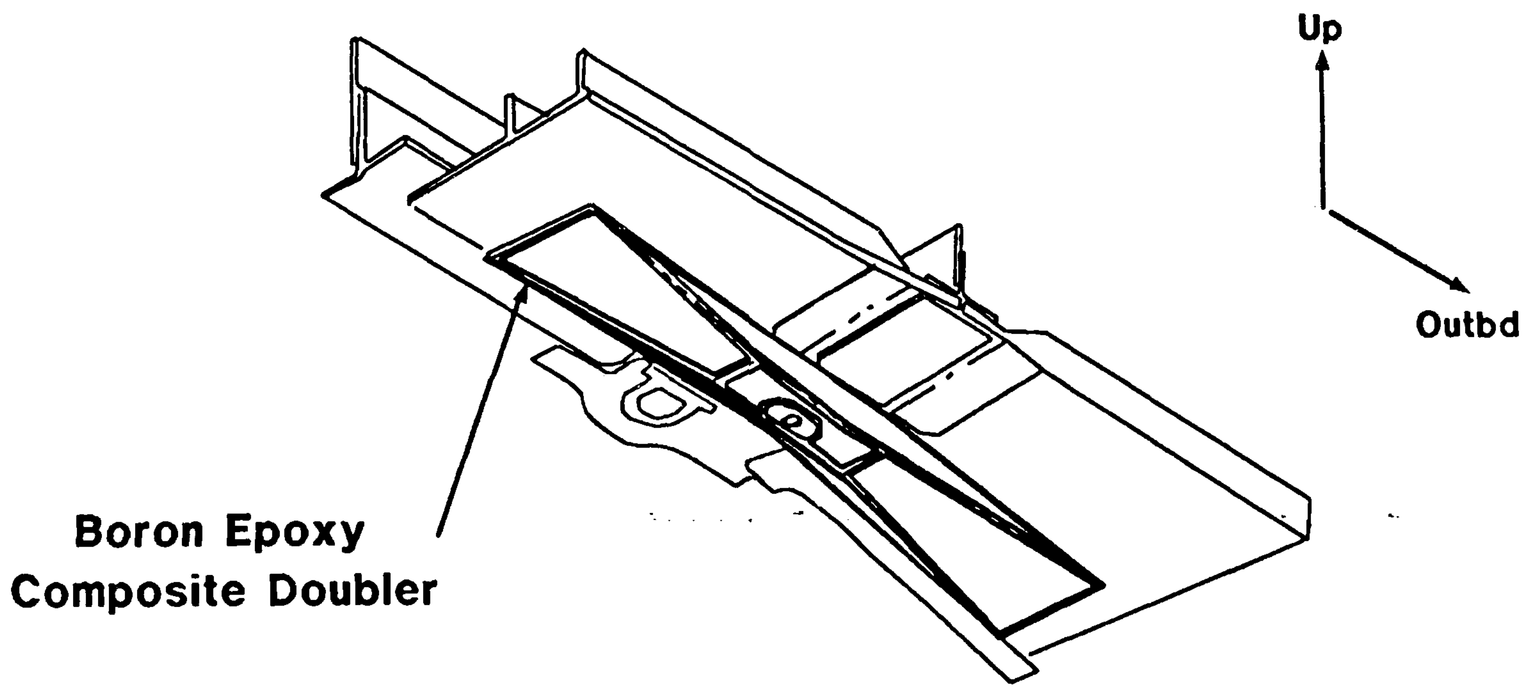
Location	Characteristics
Vertical Stabilizer / Dorsal Intersection	Boron Application; Two Directional Loading; Field Stress Reduction
FS958 Main Frame	Pocket Area; Stress Corrosion Cracking

GA-7353-6

STUDY LOCATIONS, CHARACTERISTICS

THESE TWO SLIDES PRESENTS A BRIEF SUMMARY OF THE SELECTED STUDY LOCATIONS AND THEIR RESPECTIVE CHARACTERISTICS. THESE AREAS REPRESENT BOTH FATIGUE AND CORROSION CRACKING SUSCEPTIBLE AREAS ON THE C-141 AIRCRAFT. THE BORON AND GRAPHITE REPAIRS ARE APPLICABLE IN CRACK RETARDATION, END FASTENER EFFECT RELIEF, AND FIELD STRESS REDUCTION.

C-141 Composite Repair
WS405 Rear Beam Splice



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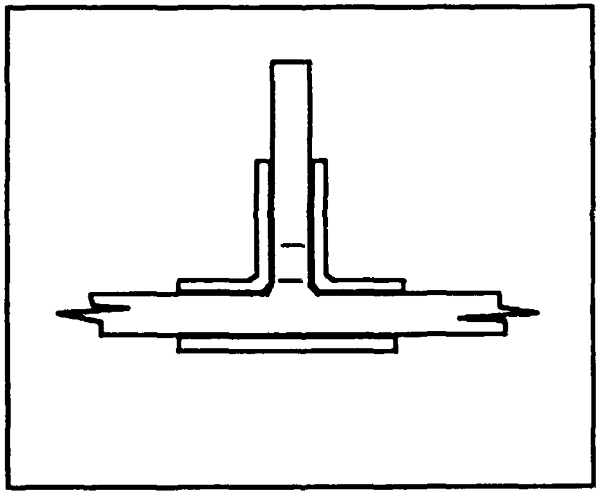
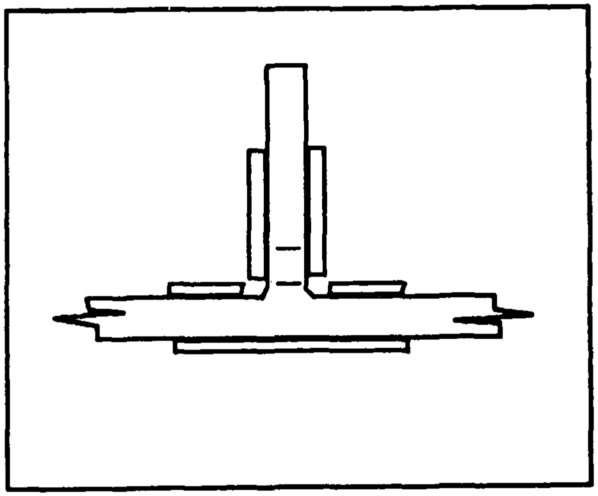
THIS SLIDE AND THE FOUR THAT FOLLOW GIVE A PICTORIAL REPRESENTATION OF THE AREAS STUDIED FOR COMPOSITE REPAIRS. THE MOST OBVIOUS AND TIME SAVING FEATURE OF THESE PARTICULAR REPAIRS IS THE FACT THAT NO FASTENER REMOVAL OR INSTALLATION IS INVOLVED IN THEIR APPLICATION.

C-141 Composite Repair Weep Hole Concept



Graphite Concept



Boron Concept

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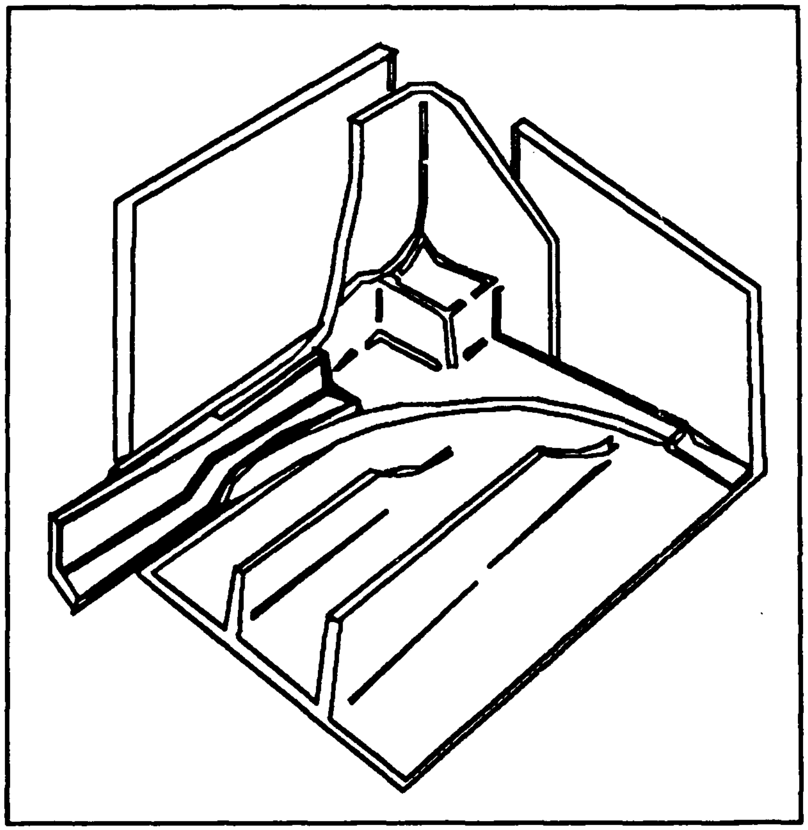
C-141 Composite Repair



Corner Fitting

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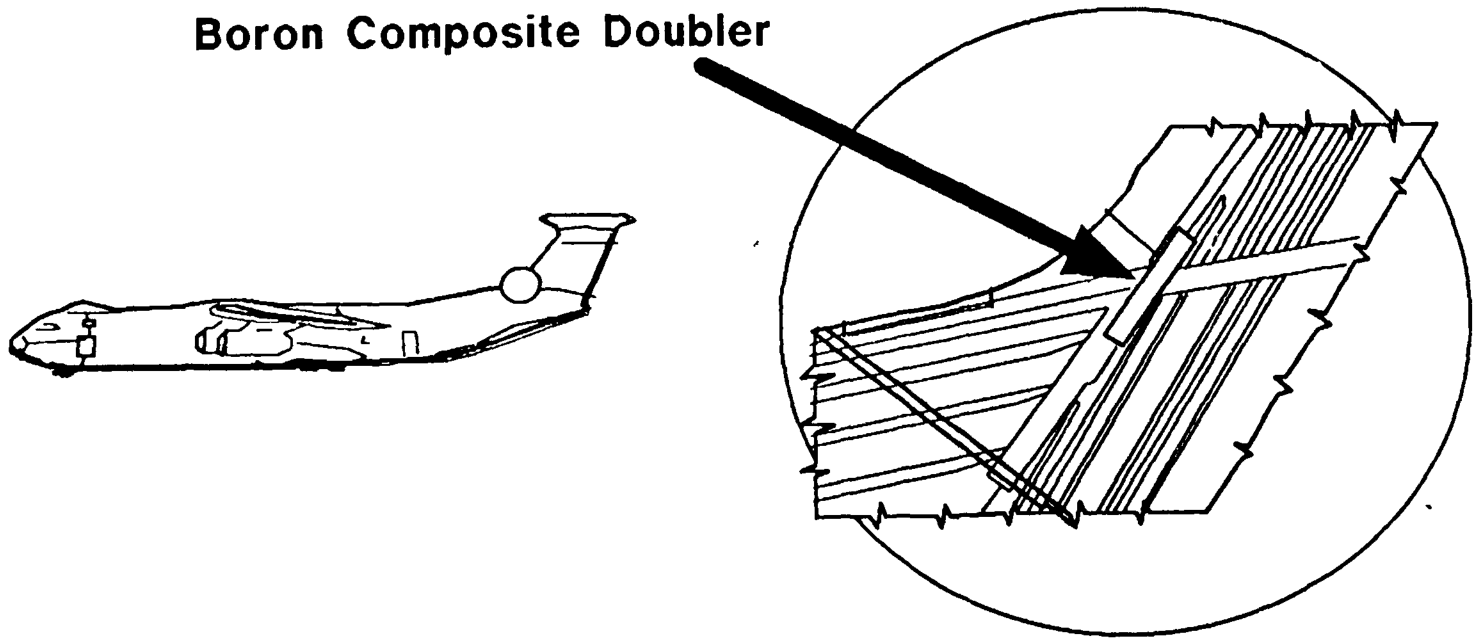
C-141 Composite Repair
Dorsal/Vertical Stabilizer



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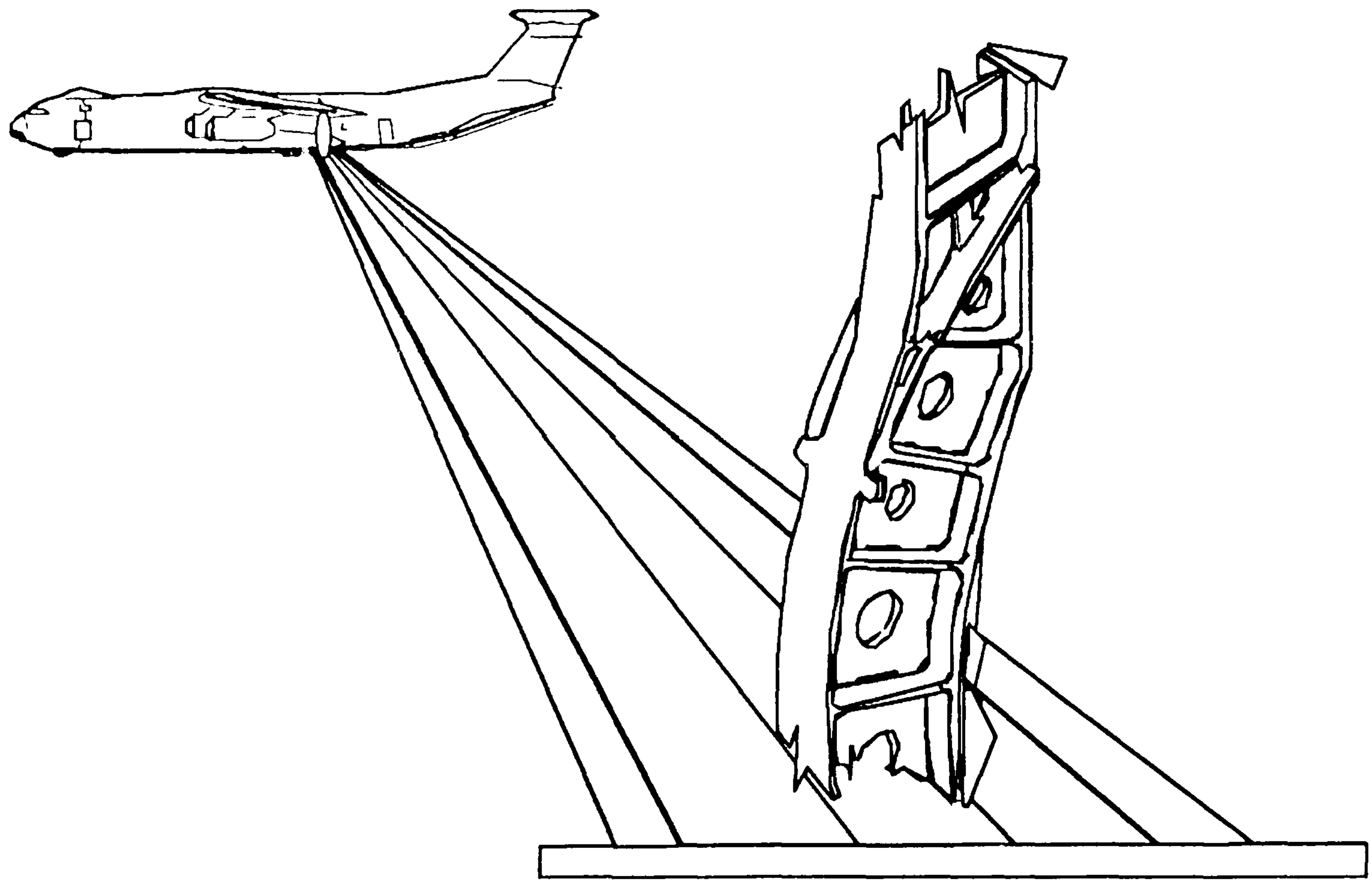
C-141 Composite Repair
FS958 Mainframe



GA-7353-11

Structural Design Requirements, Analyses

- No Fastener Removals From Damaged Structure
- No Fasteners To Attach Patch
- Retard Damage Progression Significantly
- Reliable, Practical Fab/Installation Method
- Minimum Development Testing
  - Use/Extend ARL Methods Wherever Possible
  - Detailed Finite Element Modeling and Analyses
- Benefit Force Management
  - Cost/Time Effective Over Metal Repair
  - Reduced Inspection Burden

GA-7353-12

STRUCTURAL DESIGN REQUIREMENTS, ANALYSES

IN ORDER TO DESIGN THE MOST COST EFFECTIVE REPAIR, CERTAIN GOALS OR OBJECTIVES WERE ESTABLISHED FROM THE BEGINNING. AS A PARTICULAR CONCEPT WAS STUDIED OR EVALUATED, IT HAD TO MEET OR EXCEED THE REQUIREMENTS SHOWN ON THIS SLIDE.

ANALYSES MADE EXTENSIVE USE OF FINITE ELEMENT MODELING, NOT ONLY FOR STRENGTH AND DURABILITY, BUT TO ENSURE THAT THE DESIGN WOULD REDUCE THE INSPECTION BURDEN.

Materials Comparison

	Benefits	Limitations
Boron	• Superior Stiffness • Low Electrical Conductivity • No Galvanic Couple • Closer Match of Thermal Expansion Coefficient	• Large Minimum Bend Radius • Expensive • Difficult To Fabricate
Graphite	• Ease of Fabrication • Available in a Variety of Forms • Low Cost	• High Galvanic Potential • More Difficult To Inspect After Bonding

GA-7353-13

MATERIALS COMPARISON

THIS SLIDE PROVIDES A GENERAL COMPARISON BETWEEN THE TWO CANDIDATE MATERIALS UNDER CONSIDERATION FOR USE IN REPAIRS OF METAL STRUCTURES. THE BENEFITS AND LIMITATIONS MUST BE CONSIDERED WHEN DETERMINING THE TYPE OF REPAIR APPLICATION, DIFFICULTY OF FABRICATION, AND THE DEGREE OF INSPECTABILITY REQUIRED FOR ADEQUATE FORCE MANAGEMENT.

Basic Repair Procedure

- Fabricate Composite Patch
  - Precure or Partial Cure
- Metal Surface Preparation
  - Remove Anodic Film
  - Degrease
  - Etch/Anodize/Silane
  - Apply Adhesive Primer
- Doubler Surface Preparation
  - Partially Cured Doubler - Remove Peel Ply
  - Fully Cured Doubler - Abrade and Degrease

GA-7353-14

Basic Repair Procedure (Cont'd)

- Apply Adhesive to Patch
- Position Patch on Structure
- Cure
- Inspect
- Seal

GA-7353-28

BASIC REPAIR PROCEDURE

THESE TWO SLIDES GIVE THE BASIC STEPS IN FABRICATION AND INSTALLATION OF THE COMPOSITE PATCH ON METAL STRUCTURE. DURING FABRICATION, PARTIAL CURE MAY BE USED TO FACILITATE FINAL CONTOUR FIT ON THE AIRCRAFT. SURFACE PREPARATION AND TREATMENT IS PROBABLY THE SINGLE MOST IMPORTANT STEP IN THE INSTALLATION PROCEDURE. A GOOD, SUCCESSFUL BOND WILL INSURE REPAIR INTEGRITY. THE PROPER HEAT AND PRESSURE CYCLE ARE ALSO VERY IMPORTANT TO INSURE A SUCCESSFUL BOND.

Critical Parameters in Adhesive Selection

- **Cure Temperature**
  - Low Temperature Cure Minimizes Induced Thermal Stresses

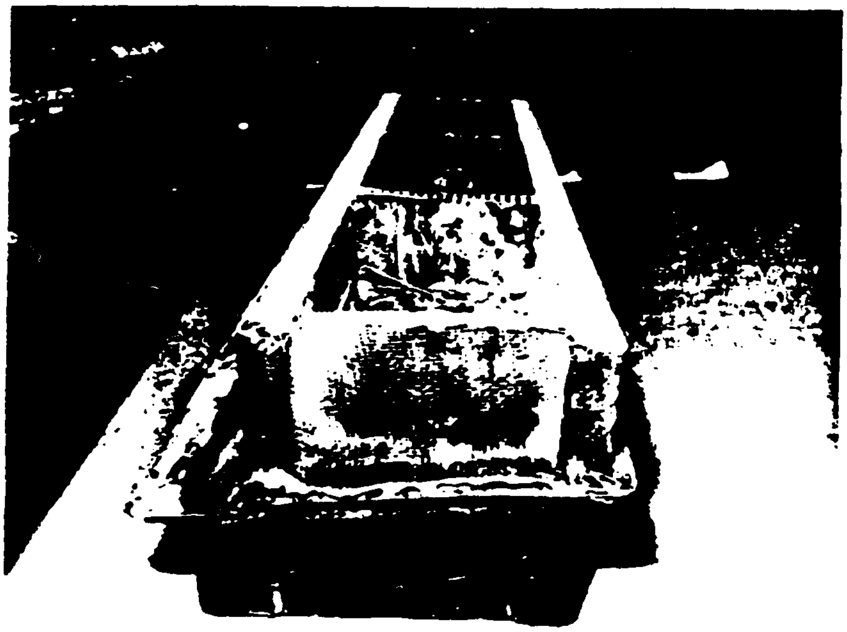
- **Strength Properties Effects Due to:**
  - Moisture
  - Aircraft Fluids
  - Temperature
  - Glass Transition Effects

GA-7353-15

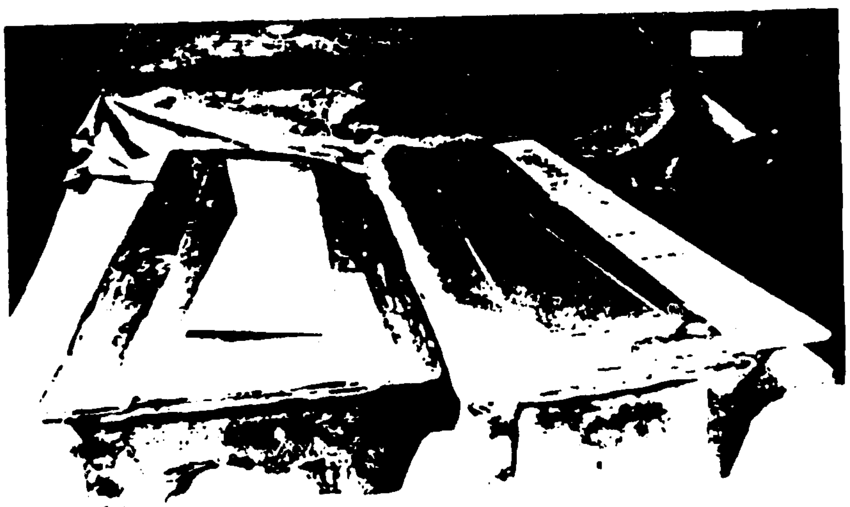
CRITICAL PARAMETERS IN ADHESIVE SELECTION

THERE ARE A NUMBER OF GOOD ADHESIVES THAT CAN BE USED IN THIS TYPE OF REPAIR APPLICATION, BUT THE PARAMETERS SHOWN ON THIS SLIDE ARE CONSIDERED THE MOST CRITICAL FOR SUCCESSFUL BONDING OF COMPOSITE PATCHES TO METAL SURFACES.

Use Color Slides
(already on hand)



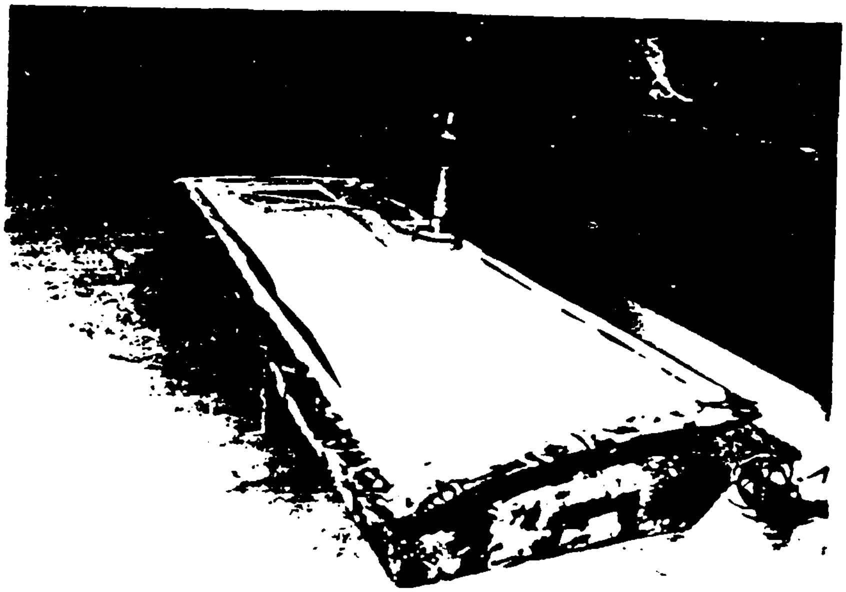
RP 3522-8



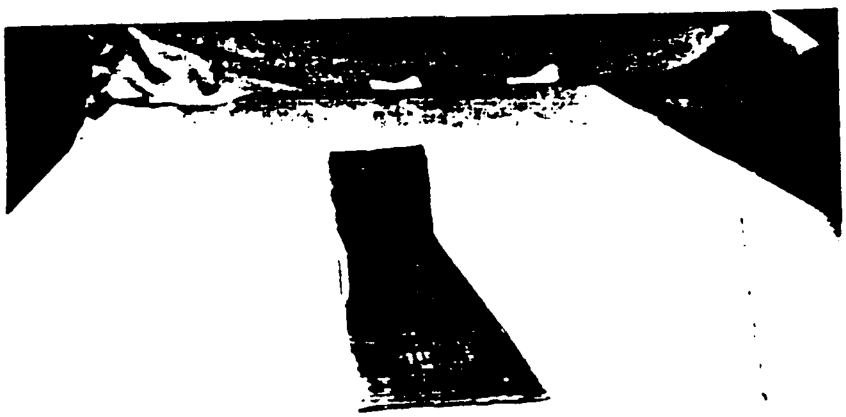
RP 3522-5

THE W.S. 405 REPAIR CONCEPT WAS EVALUATED BY FABRICATING A BORON REPAIR DOUBLER. FOR CONVENIENCE, A SLIGHTLY DOUBLE-CONTOURED PYLON SURFACE WAS USED TO SIMULATE THE WING JOINT REGION. A SPLASH WAS MADE OF THIS SURFACE. THIS IS THE SPLASH MOLD.

THE REPAIR DOUBLER OUTLINE HAS BEEN DRAWN ON THE SPLASH (RIGHT SIDE OF PHOTO), AND A FABRICATION MOLD MADE FROM IT (LEFT SIDE). [THE FINAL PART IS SHOWN ON THE FABRICATION MOLD.]



RP 3522-1



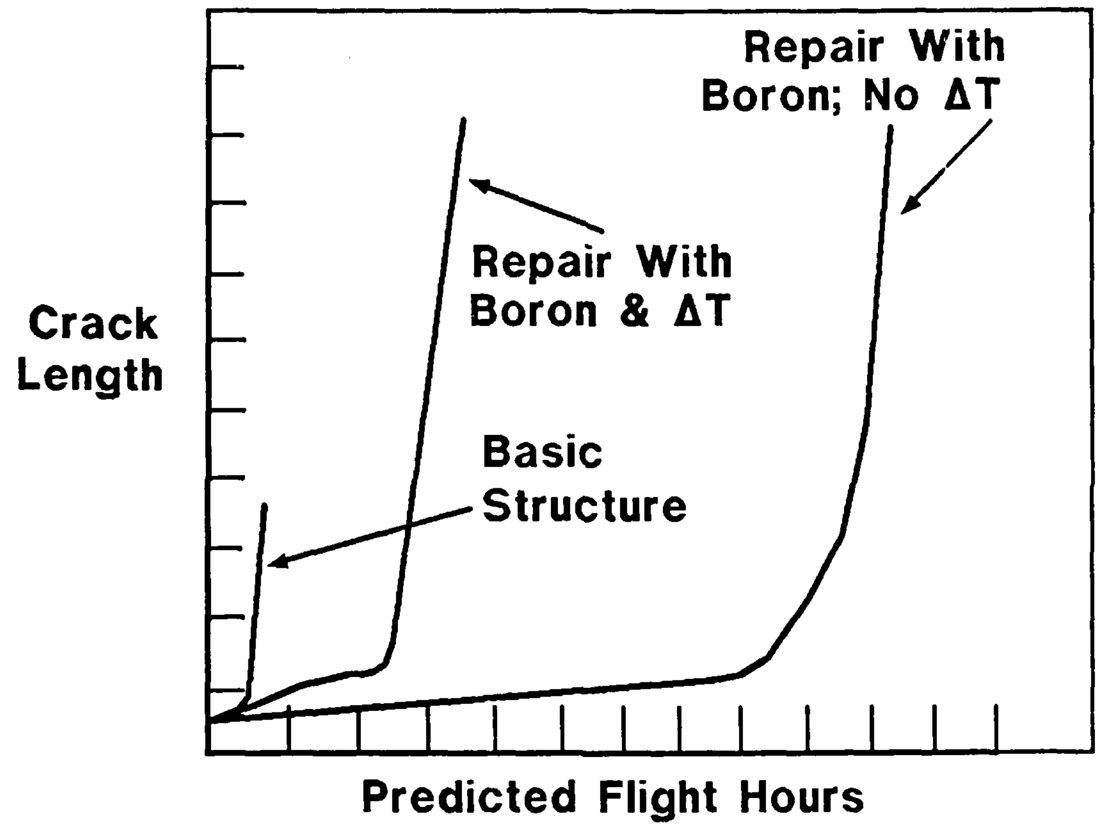
RP3522-10

15

THE BORON PLIES AND ADHESIVE ARE LAID UP ON THE FABRICATION MOLD AND A VACUUM BLANKET USED TO HOLD THE LAYUP FOR INSTALLATION IN THE AUTOCLAVE. THE REPAIR PART WAS CURED FOR 70 MINUTES AT 350°F AND 85 PSIG.

THE FINAL W.S. 405 BORON COMPOSITE REPAIR PART IS SHOWN HERE. DUE TO UNSYMMETRICAL 45° PLIES, THE PART WARPED SLIGHTLY AND WAS CONSIGNED TO THE "*LESSONS LEARNED*" CATEGORY. A REDESIGNED PART WAS SUCCESSFULLY FABRICATED.

C-141 Weep Hole Analysis

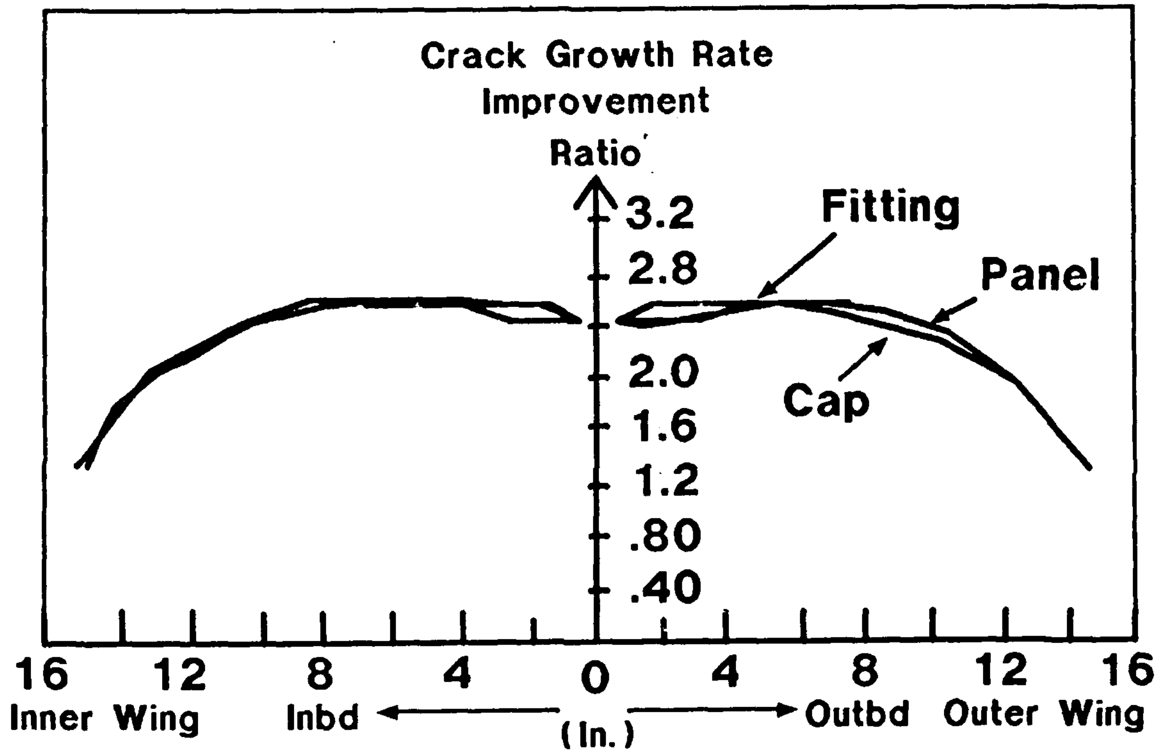


GA-7353-16

C-141 WEEP HOLE ANALYSIS

THIS SLIDE GRAPHICALLY PRESENTS THE EFFECTS ON REPAIR LIFE AS A RESULT OF RESIDUAL STRESS INDUCED BY TEMPERATURE CHANGES. FOR THE WING WEEP HOLE REPAIR AREA, THE LIFE AT ROOM TEMPERATURE IS REDUCED BY 70% WHEN SUBJECTED TO A TEMPERATURE DIFFERENTIAL OF 135°F, BUT IS STILL A FACTOR OF 6 OVER THE UNREPAIRED STRUCTURE.

C-141 WS405
Boron Repair Analysis



Crack Growth Rate With Boron Repair
at -65°F vs Unrepaired Structure

GA-7353-17

C-141 WS 405 BORON REPAIR ANALYSIS

THIS SLIDE GRAPHICALLY PRESENTS THE FATIGUE CRACK GROWTH IMPROVEMENT DUE TO INSTALLATION OF THE BORON DOUBLER. DUE TO THE COMPLEXITY OF THE LOAD PATHS AT THE JOINT, THE CONSERVATIVELY CALCULATED IMPROVEMENT FACTOR IS ONLY 2.4 WHEN THERMAL EFFECTS OF -65° ON STRESS LEVELS ARE CONSIDERED.

Testing

- Minimum Budget Program
- Use ARL Experience and Published Data
- Use Finite Element Models in Lieu of Full-Scale Tests
- Use Simple Tests To Correlate Results From Models
- Install Repairs Directly on Service Aircraft for Evaluation
  - No Degradation of Basic Structure Strength
  - Monitor Service Experience

GA-7353-18

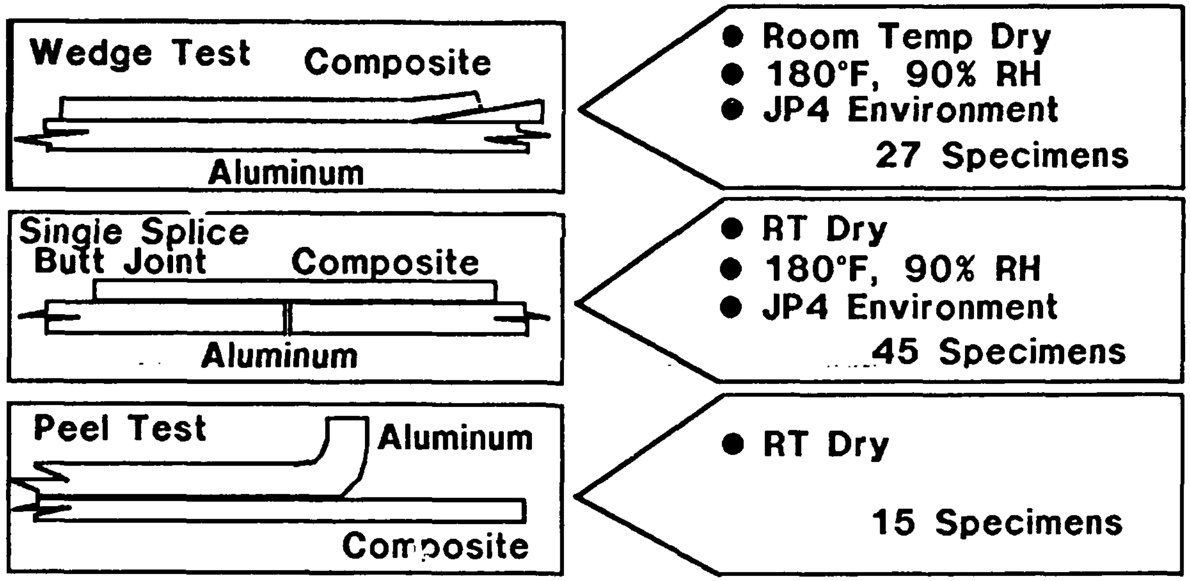
TESTING

AS WITH ANY NEW REPAIR TECHNOLOGY, A TEST PROGRAM TO PROVIDE DESIGN AND APPLICATION VERIFICATION IS NECESSARY TO SUBSTANTIATE THE ANALYTICAL PROCESS. AS SEEN BY THIS SLIDE THE TEST PROGRAM ENVISIONED WILL BE AT MINIMUM COST, TAKING ADVANTAGE OF AUSTRALIAN KNOWLEDGE AND FINITE ELEMENT MODELING TO CORRELATE SIMPLE TESTS. ALSO, INSTALLATION OF COMPOSITE REPAIRS ON SERVICE AIRCRAFT (WHEN OPPORTUNITY ARISES) WILL GIVE SERVICE EXPERIENCE TO STRENGTHEN CONFIDENCE IN COMPOSITE REPAIR USAGE.

Test Plan
Surface Preparation

Types of Preparation:

- Silane Treatment
- Phosphoric Acid Etch
- STP 57-004
  LASC Baseline



- Room Temp Dry
- 180°F, 90% RH
- JP4 Environment

27 Specimens

- RT Dry
- 180°F, 90% RH
- JP4 Environment

45 Specimens

- RT Dry

15 Specimens

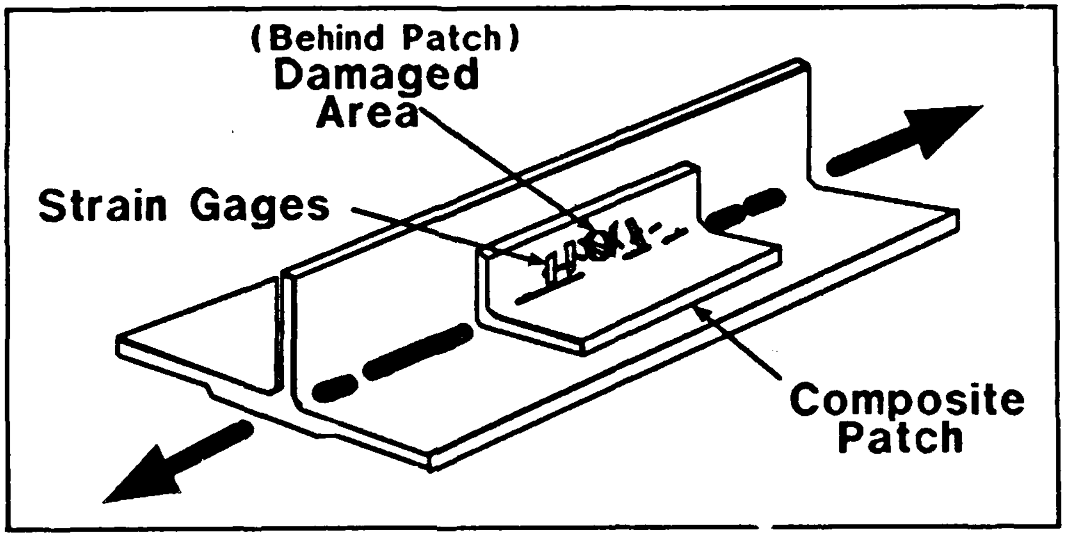
Total 87 Specimens

GA-7353-19

TEST PLAN - SURFACE PREPARATION

A MINIMUM BUDGET TEST PLAN PLUS RELIANCE ON EXISTING INFORMATION IS CONSIDERED APPROPRIATE TO THIS STUDY. 87 SIMPLE SPECIMEN TESTS ARE PLANNED TO CHECK ADEQUACY OF SURFACE PREPARATION AND BOND STRENGTH. A COMPOSITE STRIP WILL BE BONDED TO AN ALUMINUM STRIP FOR EACH COUPON. THREE TYPES OF SURFACE PREPARATION WILL BE EVALUATED. WEDGE TESTS, LAP SHEAR, AND PEEL TESTS WILL BE PERFORMED FOR EACH TYPE OF SURFACE PREPARATION. TESTS AT ROOM TEMPERATURE, HOT/WET, AND IN A JET FUEL ENVIRONMENT ARE PLANNED. DATA WILL BE COMPARED WITH PUBLISHED DATA AND AUSTRALIAN RESEARCH LABS' FINDINGS. SURFACE PREPARATION AND BONDING ARE THE MOST CRUCIAL ELEMENTS OF THE COMPOSITE REPAIRS CONCEPT.

Component Test



- Wing Weep Holes Test
- Specimens and Fixtures Available From Earlier Tests
- Static Test One Specimen
  - Strain Gages
  - Correlate to Finite Element Model
- Durability Test One Specimen
  - Flight by Flight Loads
  - Service Aircraft Spectrum

GA-7353-20

COMPONENT TEST

AS PART OF THE MINIMUM COST TEST PROGRAM, EXISTING TEST SPECIMENS FROM C-141 WEEP HOLE TESTS WILL BE USED. PATCHES WILL BE INSTALLED OVER EXISTING SPECIMEN CRACKS AND STATIC AND DURABILITY TESTS PERFORMED. A CORRELATION ANALYSIS WILL BE PERFORMED USING FINITE ELEMENT MODELING.

In-Service Opportunity

- Crack Found in Center Wing Lower Surface Panel of Service Aircraft 650256
- Designed, Fabricated, Installed Boron Repair in Two Weeks
  - Estimate for Metal Repair - Five to Six Weeks
- Damage Tolerance Analysis (DTA) Indicated No Further Action Should Be Necessary
  - Inspection Planned To Confirm Integrity

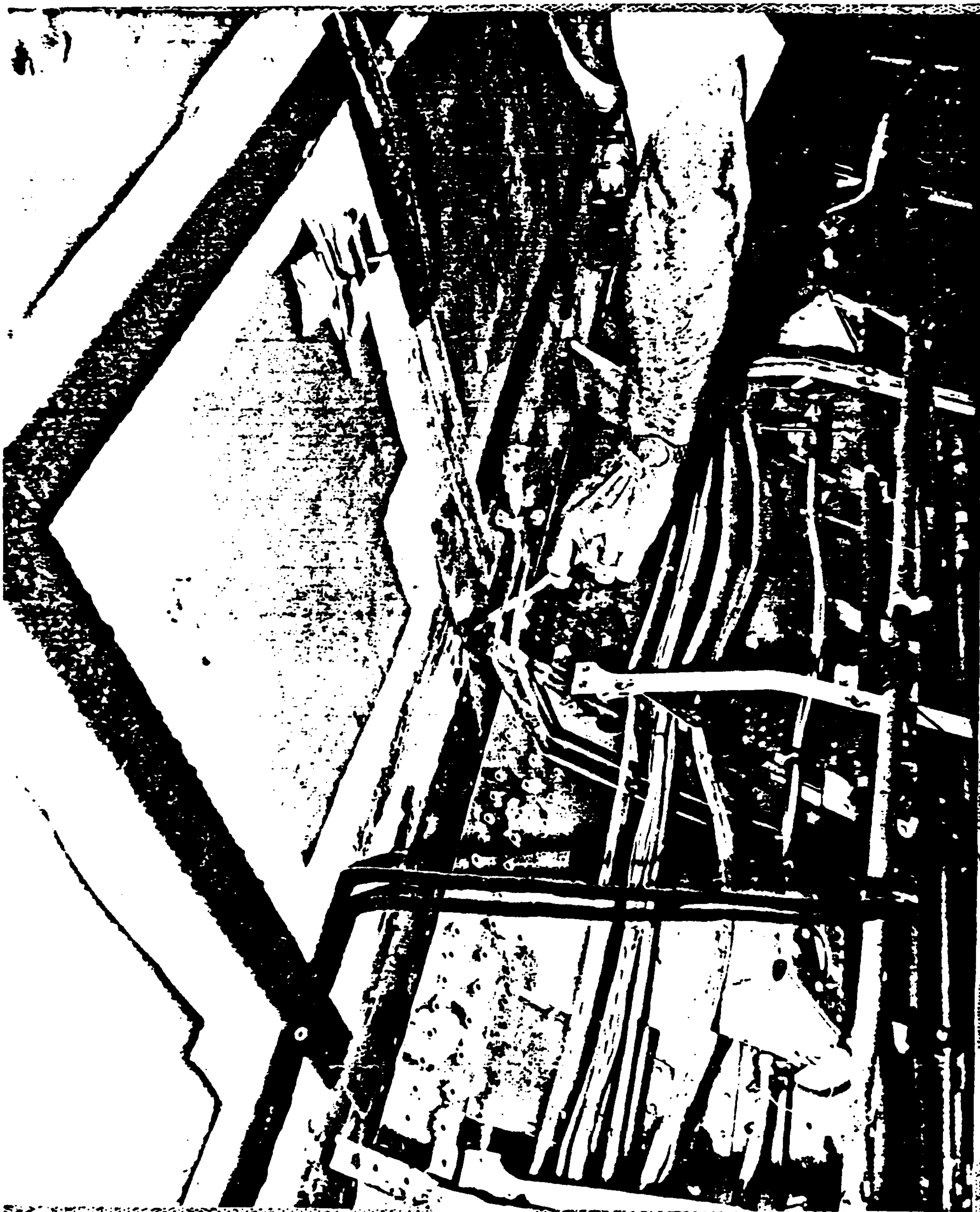
GA-7353-21

IN-SERVICE OPPORTUNITY

DURING THE C-141 CENTER WING REPAIR ACTIVITY ON AIRCRAFT 65-0256, A 1 1/2 INCH FATIGUE CRACK WAS DISCOVERED. QUICK CONSULTATION WITH WR-ALC ENGINEERING AND LOCAL AFPRO/QA RESULTED IN THE DECISION TO "GO FOR IT" BY REPAIRING THE PANEL USING BORON DOUBLERS.

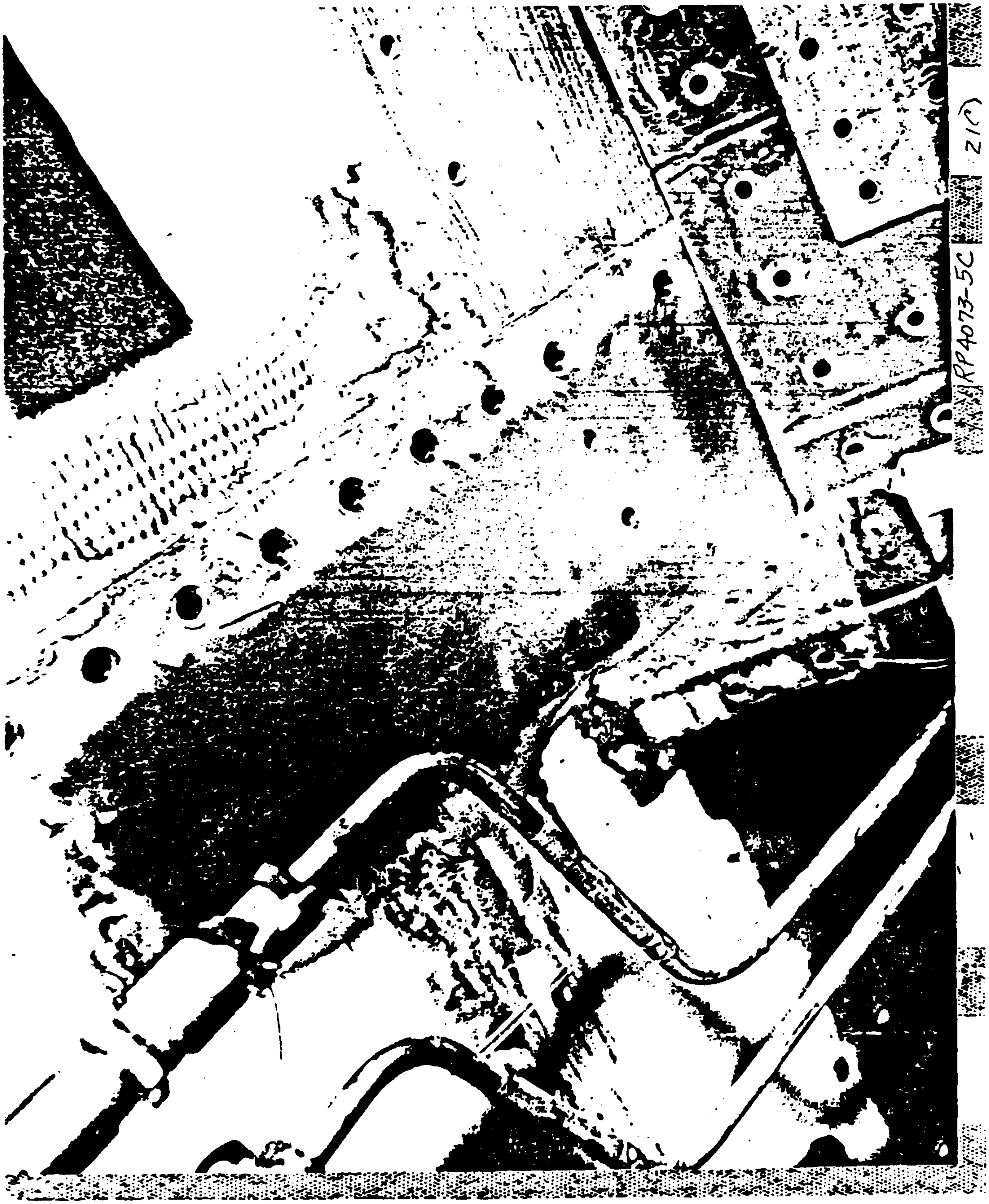
THIS DECISION RESULTED IN 3 TO 4 WEEKS LESS AIRCRAFT DOWNTIME AND SEVERAL HUNDRED MANHOURS SAVED; BUT MORE THAN THAT, THIS REPAIR PIONEERS THIS TYPE TECHNOLOGY ON USAF AIRCRAFT WHICH EVENTUALLY WILL SPREAD TO MANY OTHER AIRCRAFT.

THE CRACK HAD BEEN CUT OUT WITH A CHORDWISE SLOT AND A PLUG APPROXIMATELY 0.5" X 1.5" INSTALLED FOR FUNCTIONAL REASONS. BORON REINFORCING DOUBLERS WERE THEN DESIGNED TO APPROXIMATELY NULLIFY THE STRESS CONCENTRATIONS DUE TO THE SLOT. THE LOCATION OF THE REPAIR IS IN THE BACKGROUND IN THIS PICTURE. ONE OF THE DOUBLERS IS BEING HELD BY JOE COCHRAN.

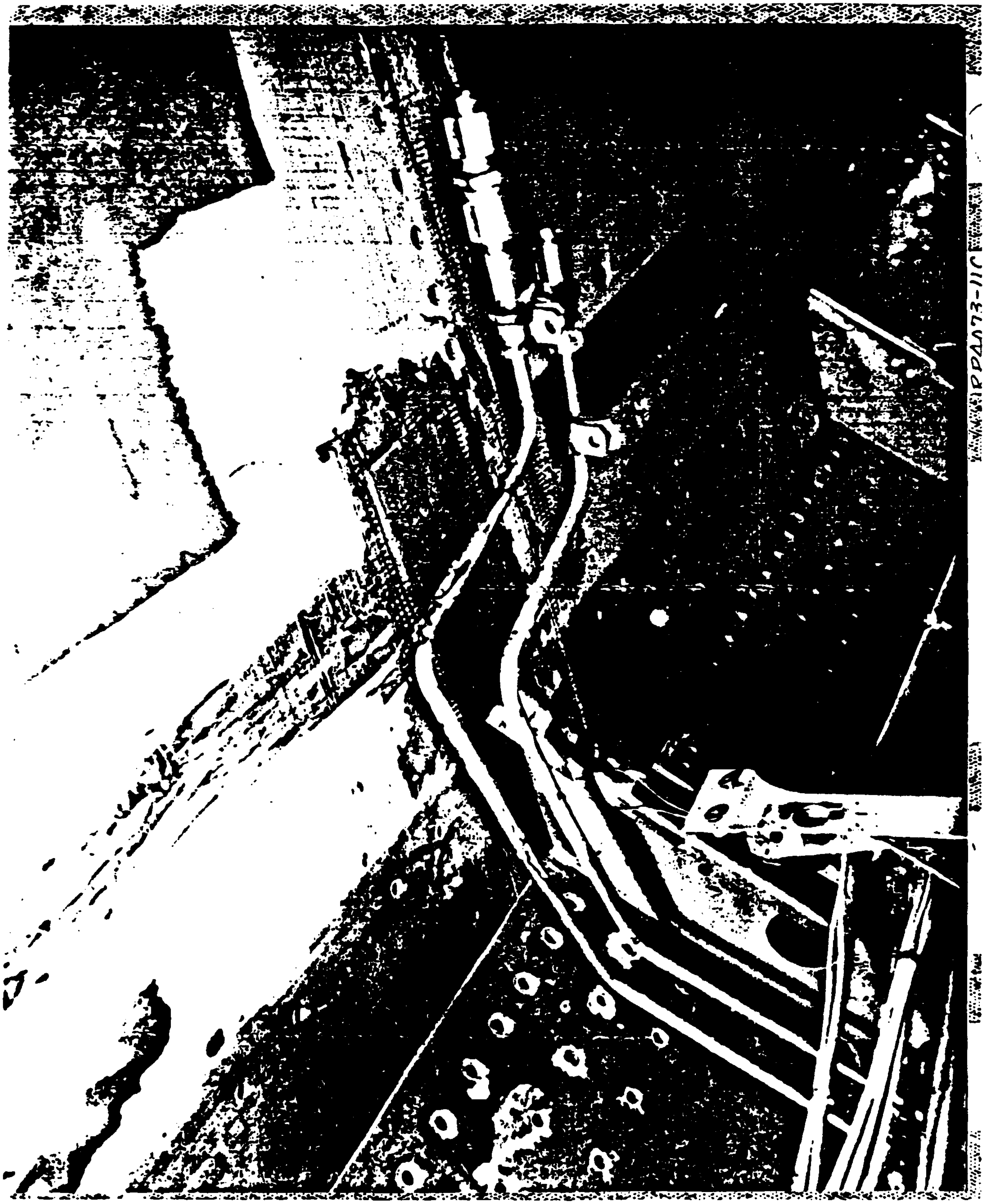


NRPA073-6C

THE LOCATION OF THE CRACK IN THE CENTER WING LOWER SURFACE PANEL IS SHOWN HERE. THE REGION IS CONGESTED ~UE TO THE INTERSECTION OF THE WING WITH THE FUSELAGE. DRAG ANGLES ABOVE AND BELOW THE WING PANEL HAD BEEN REMOVED TO ACCOMPLISH THE CENTER WING REPAIR.


RPA073-5C

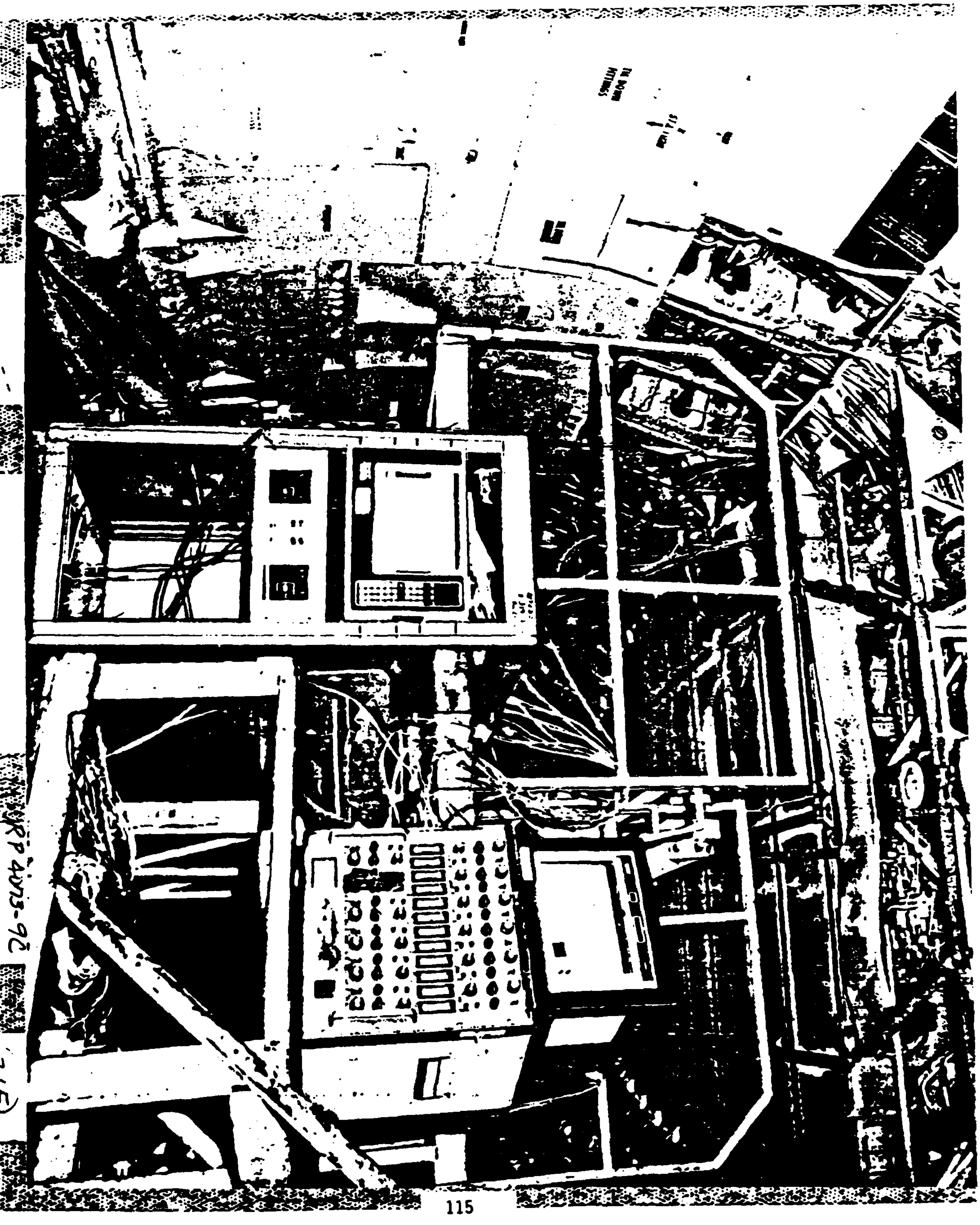
FOR THIS REPAIR, SPACE CONSTRAINTS PREVENTED GRIT BLASTING FOR SURFACE CLEANING. SANDING WAS USED INSTEAD. PRIOR TO THE BONDING OF THE COMPOSITE REPAIR DOUBLERS, ALL SPANWISE SPLICE FASTENERS EXCEPT THOSE COMMON TO THE DRAG ANGLE WERE REINSTALLED. (THEY HAD BEEN REMOVED FOR INSPECTION AND WERE NOT COVERED BY THE DOUBLERS.)



THE REPAIR DOUBLERS WERE SIZED TO APPROXIMATELY DUPLICATE THE PANEL THICKNESS TIMES MODULUS OF ELASTICITY. ALL FIBERS ARE LONGITUDINAL, AND THE PATCHES ARE NEARLY SYMMETRICAL (ABOVE AND BELOW, AND INBOARD AND OUTBOARD OF THE PLUG). THE 3" WIDE EIGHT PLY DOUBLERS ARE FABRICATED FROM AVCO RIGIDITE 5505/4 BORON/EPOXY PREPREG. THEY ARE SYMMETRICALLY STEP PLIED WITH PLIES 1, 4, 5, AND 8 ELEVEN INCHES LONG; PLIES 2 AND 4 8.4 INCHES LONG; AND PLIES 3 AND 6 5.8" LONG. THE DOUBLERS WERE AUTOCLAVE CURED AT 350°F AND 85 PSIG FOR 70 MINUTES.

A VERIFICATION BOND CYCLE WAS MADE PRIOR TO BONDING TO DETERMINE CORRECTNESS OF DOUBLER FIT, ADEQUACY OF PRESSURE APPLICATION TECHNIQUE, AND HEATER BLANKET CAPABILITY AND TEMPERATURE DISTRIBUTION. THE PROCEDURE DUPLICATED THAT PLANNED FOR THE ACTUAL INSTALLATION, EXCEPT THAT A 1 MIL LAYER OF TEFLON WAS USED ON BOTH SURFACES OF THE FM 73 ADHESIVE TO PREVENT BONDING TO THE WING AND THE DOUBLERS.

ZONE HEATERS, SILICON RUBBER, INSULATION, AND PRESSURE PLATES WERE POSITIONED AND HELD IN PLACE WITH BOLTS USING EXISTING FASTENER HOLES. FOR THE ACTUAL INSTALLATION, A SPREADER BAR WAS USED TO APPLY EXTRA PRESSURE TO ONE AREA OF THE BOTTOM DOUBLER.



THOROUGH SURFACE PREPARATION, COMPLETED WITH DEGREASING, ACID PASTE ETCH, AND AN APPLICATION OF BR127 PRIMER, WAS ACCOMPLISHED FOLLOWING THE VERIFICATION BOND CYCLE. THE INSTALLATION BOND CYCLE APPLIED A TEMPERATURE OF 175 - 190°F FOR EIGHT HOURS, FOLLOWED BY 250°F FOR 1 1/2 HOURS. AN AUTOMATIC INSTRUMENT WITH TWELVE INDEPENDENT THERMOSTATIC HEAT RATE CONTROLS WAS USED FOR HEATING. A 24 CHANNEL TEMPERATURE RECORDER WAS USED TO MONITOR WING TEMPERATURE. THE SETUP AND EQUIPMENT ARE SHOWN HERE.

SIX ALUMINUM LAP SHEAR SPECIMENS, WHICH WERE PREPARED AND BONDED SIMULTANEOUSLY WITH THE DOUBLERS, WERE TESTED TO CONFIRM BOND STRENGTH. THE AVERAGE VALUE WAS 5,070 PSI.

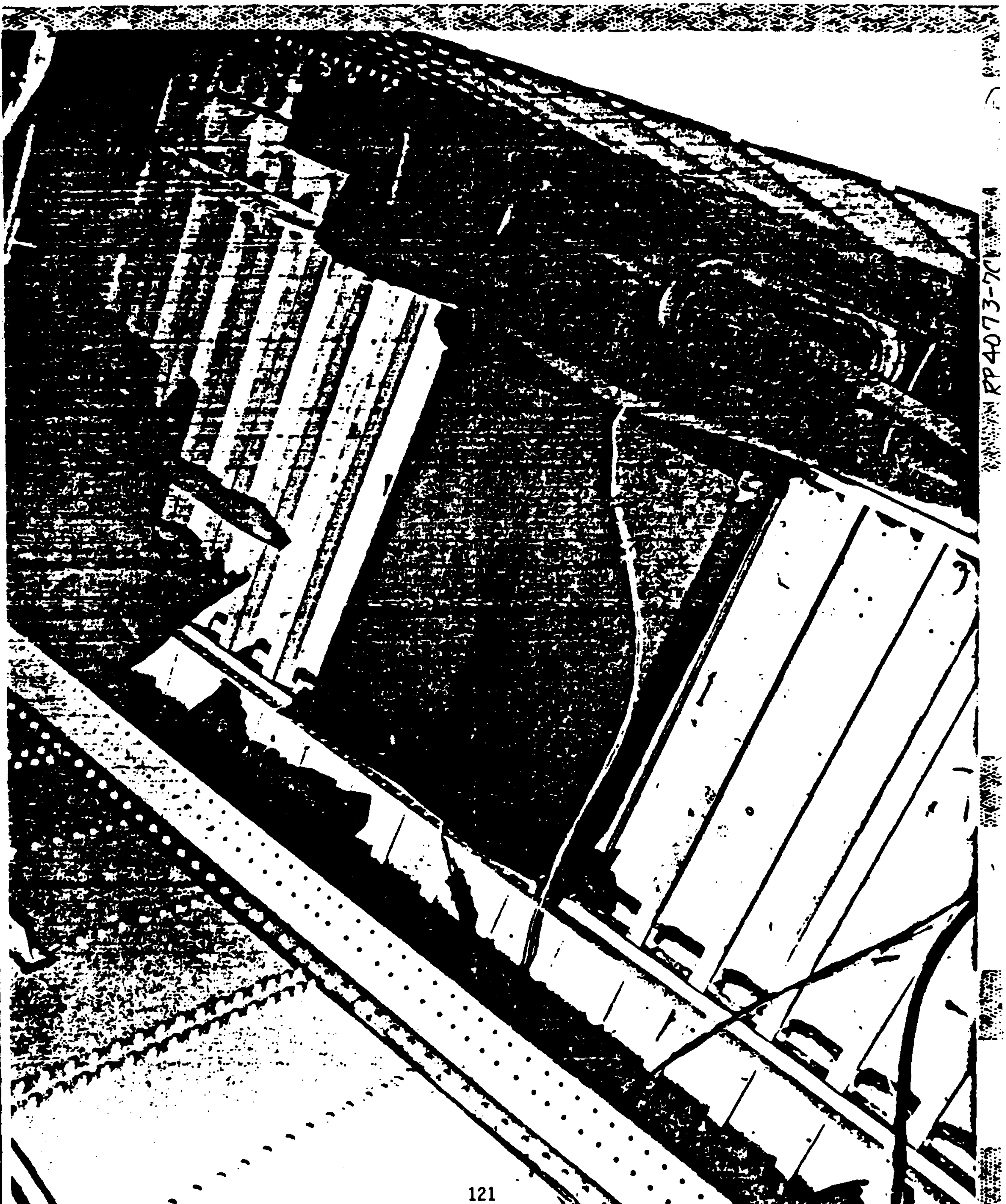

RP4073-17C

THE UPPER SURFACE (INNER) DOUBLER WAS BONDED SIMULTANEOUSLY WITH THE LOWER SURFACE (OUTER) DOUBLER. THE LOCATION IS SHOWN HERE.

RP40'73-120 (H) 212

HOLES IN THE DOUBLERS WERE TO ACCOMMODATE THE DRAG ANGLE FASTENERS. THEY ARE NOT USED FOR DOUBLER FASTENING.

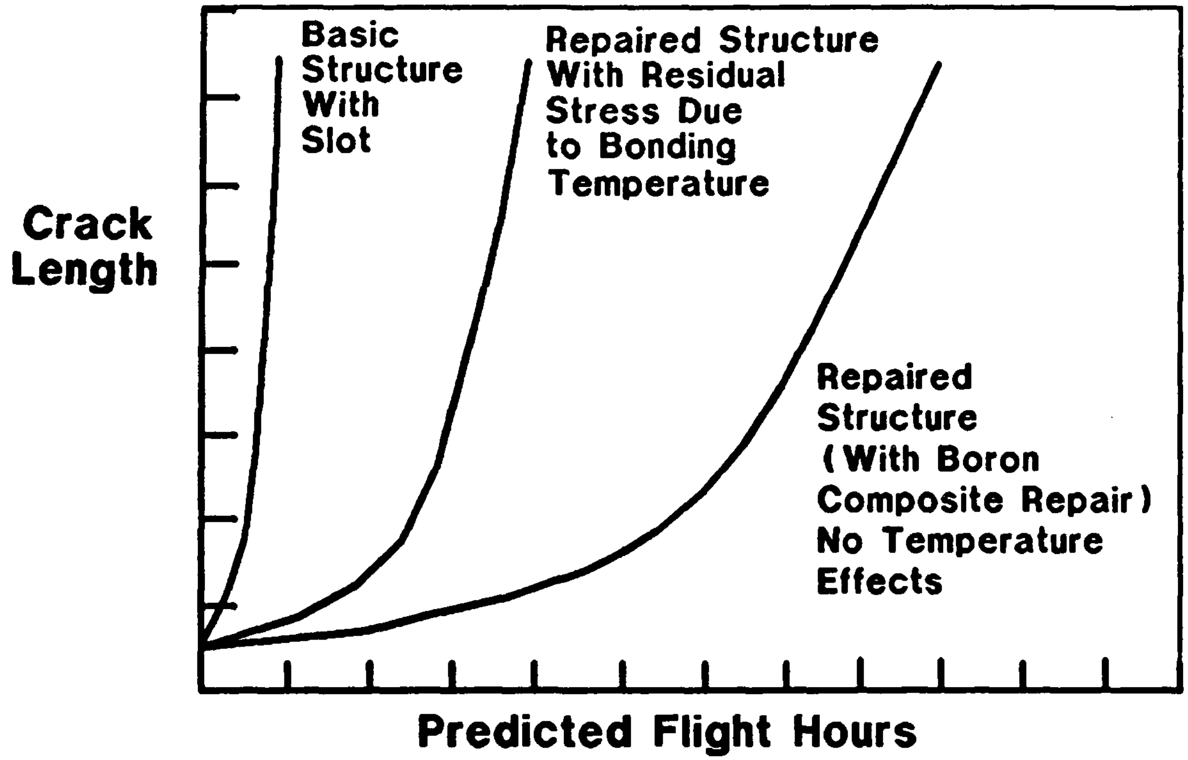
THE THINNESS OF THE DOUBLERS WAS VERY BENEFICIAL TO THE REPAIR OF THIS CONGESTED REGION.


RP4073-7C

THIS IS THE BONDING SETUP FOR THE UPPER DOUBLER.

C-141B

Predicted Crack Growth From Rogue Flaw in Basic Panel A/C Serial 650256 (Gelac 6107) Pnl 10 Lwr With Composite Repair of 1.52 Inch Chordwise Slot



GA-7353-22

1

THE CONSERVATIVE CRACK GROWTH CALCULATIONS SHOWN HERE INDICATE THAT THERE SHOULD BE NO FURTHER PROBLEM ON THIS AIRPLANE AT THIS LOCATION. THE INCREASE IN LIFE IS APPROXIMATELY A FACTOR OF 4.3 ASSUMING 10 KSI RESIDUAL STRESS DUE TO THERMAL EFFECTS (T), OR A FACTOR OF 9.8 NOT CONSIDERING T. AIR FORCE ULTRASONIC INSPECTION FOR BOND INTEGRITY AND X-RAY INSPECTION AS A CHECK ON CRACKING UNDER THE DOUBLERS ARE PLANNED AT APPROPRIATE PDM'S TO CONFIRM THE COMPOSITE REPAIRS STRUCTURAL INTEGRITY.

Non-Destructive Inspection (NDI) of Boron Repair

- Ultrasonic
  - Performed in Immersion Tank
  - Inspect for Disbonds in Repair Parts and Installation
  - Use Standards To Calibrate
  - Signal Is Affected by Elements in Material
  - Inspect Before Final Sealing
  - Sealant Covering Repair Makes Ultrasonic Inspection Impossible
- Radiographic
  - Inspect Basic Structure Under Rapair
  - A_{NDI} = 0.5 In. (Under 0.12 Repair)
- Eddy Current
  - Inspect Material Under Repair
  - Can Only Detect Surface Defects
  - A_{NDI} = 0.25 In. (Under 0.12 Repair)
  - Sealant Covering Repair Makes Eddy Current Inspection Impossible

GA-7353-23

NDI OF BORON REPAIR

SINCE THE REPAIR OF THE C-141 CENTER WING LOWER SURFACE PANEL IS THE FIRST OF ITS KIND, AN NDI TECHNIQUE HAD TO BE DEVELOPED FOR THIS PARTICULAR LOCATION AND PROVIDED TO THE AIR FORCE FOR FOLLOW-ON INSPECTIONS. THIS SLIDE GIVES THE THREE METHODS INVESTIGATED WITH SOME NOTED FEATURES OF EACH. THE NATURE OF THE REPAIR RESULTED IN THE SELECTION OF RADIOGRAPHIC AS THE FOLLOW-ON INSPECTION PROCEDURE. ALTHOUGH ANALYTICAL CALCULATIONS INDICATE NO FOLLOW-ON INSPECTIONS REQUIRED, THIS REPAIR WILL BE INSPECTED AT EACH MAJOR ISO TO RECORD PATCH BEHAVIOR.

Benefits

- Benefits to ASIP Manager
  - Potential Cost, Reliability, Down Time Improvements Over Metal Repairs
  - Repairs and Preventive Repairs With Maximum Benefits
    - No Basic Structure Degradation From Added Fastener Holes
    - Optimum Tapering at Ends of Repair
    - Can Mold To Fit Contours
  - Minimizes Inspection Burden
  - Minimizes Development and Testing Cost/Time
    - Earlier Introduction Into Service Application
- RAMTIP Candidate

GA-7353-24

BENEFITS

COMPOSITE-ON-METAL REPAIR TECHNOLOGY OFFERS TREMENDOUS BENEFITS TO THE ASIP MANAGER AND MEETS THE OVERALL OBJECTIVES OF R&M 2000. BASICALLY, THE FEATURES LISTED ON THIS SLIDE CAN BE SUMMARIZED AS "PROVIDING MORE, WITH LESS, FOR LESS".

Lessons Learned

- Use Australian Technology Wherever Possible
- Design a Balance of Cross Plies To Minimize Warpage
- Precuring Desirable Where Possible
- Proper Surface Preparation Essential
- 175°F Adhesive Cure Practical
- Significant Heat Sink Effects
- Aluminum Foil Cover To Permit Eddy Current Inspection

GA-7353-26

LESSONS LEARNED

DEVELOPMENT OF THIS REPAIR TECHNOLOGY FOR USE ON THE C-141 AIRCRAFT HAS BEEN SUCCESSFUL BECAUSE WE TOOK EVERY ADVANTAGE WE COULD OF THE AUSTRALIAN WORK. NO "RE-INVENTING OF THE WHEEL", BUT RATHER ENHANCING THE TECHNOLOGY TO MEET PECULIAR C-141 REQUIREMENTS. THE ITEMS LISTED ON THIS SLIDE ARE SOME OF THE MORE PERTINENT FEATURES WE HAVE LEARNED.

Summary

- Significant Benefits by Composite Repairs
  - Can Reduce Cost and Down Time for Fab, Installation, Inspection
  - Slow or Arrest Crack Growth
- *Can Apply Prototypes Without Awaiting Tests*
- Process Is in Development Stage
  - Depot Level Repair / Mod at Present
- Benefits RAMTIP, ASIP Manager Initiatives
  - Available Now
  - Need Not Await Testing

GA-7353-27

SUMMARY

IN CLOSING THIS PRESENTATION, IT MUST BE STATED THAT THERE IS STILL MUCH TO DO AND LEARN BEFORE THIS REPAIR TECHNOLOGY WILL BE ACCEPTED BY THE USING COMMUNITY. WE HAVE SEEN THAT SIGNIFICANT BENEFITS CAN RESULT FROM THESE TYPE REPAIRS IN TERMS OF COST, AIRCRAFT DOWNTIME AND MAINTENANCE. WE KNOW THAT PROTOTYPE REPAIRS CAN BE INSTALLED WITHOUT AWAITING TEST CONFIRMATION. ALTHOUGH THE PRESENT CAPABILITY IS AT DEPOT LEVEL, EVENTUAL FIELD REPAIR CAPABILITY IS PRACTICAL. AND FINALLY, THIS TECHNOLOGY AFFORDS THE ASIP MANAGER A BENEFICIAL FORCE MAINTENANCE TOOL WITH SIGNIFICANT R&M SAVINGS.

CHARACTERIZATION OF IMPACT DAMAGE IN COMPOSITES

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AND

S. DONALDSON

MECHANICS AND SURFACE INTERACTIONS BRANCH, AFWAL/MLBM

AIR FORCE WRIGHT AERONAUTICAL LABORATORIES/MATERIALS LABORATORY

WRIGHT-PATTERSON AFB, OHIO 45433

ABSTRACT

Some of the most critical flaws encountered during the service life of a composite component result from impact damage due to foreign objects (FOD). The seriousness of this problem has led to much work in both the Nondestructive Evaluation and Mechanics communities to characterize and assess the effects of such damage. This paper will report on a new ultrasonic method to nondestructively produce and display images of the damage on a ply-by-ply basis with all of the data being collected during a single scan. Specimens consisting of 32 ply quasi-isotropic graphite/epoxy and graphite/PEEK composites were examined. Selected data from a specimen of each material that was subjected to a 20 J (15 ft-lb) impact will be presented to demonstrate the power of this new technique.

INTRODUCTION

Laminated composites are now used as engineering materials on aerospace, automotive, and marine structures. They are often the material of choice when high stiffness, high strength, and minimum structural weight are required. However, composites are subject to several forms of damage, including transverse ply cracking, fiber breakage, fiber/matrix debonding, and delamination.

Delamination is the separation of plies by planar cracks in the resin layer between the plies. Delamination damage can significantly reduce the laminate strength, especially under compressive loading. Delaminations may be caused by manufacturing flaws, transverse impact from tool drops, runway debris, or hail, or may grow from structural details such as free edges or ply build-ups. Destructive sectioning [1] or de-plying [2] of impact damaged laminates has shown that very often the damage is comprised of many delaminations, each of different size, shape and ply interface location through the thickness. Unfortunately, the extent of the delamination damage typically may not be identifiable from visual surface evidence. Non-destructive methods such as the "coin-tap", through-transmission, and pulse echo ultrasonic methods have been used to determine the extent of internal delamination damage. These methods were useful only in determining the outline of the maximum extent of all superimposed delaminations, but gave no indication as to the depth of the individual delaminations. Structural analysis of the compression behavior

of delaminated laminates has shown that the critical load, or load required to extend an existing delamination, depends upon the depth of the delamination [3]. Therefore, a technique was needed to non-destructively determine not only the maximum extent of delamination damage, but also the size and depth of individual delaminations.

More advanced instrumentation than is currently in common laboratory and field usage can provide some detail as a function of depth by using multiple time gates (a hardware or software function which isolates the signal returning at a specified time delay relative to a reference) on the detected and filtered signal. However, even this instrumentation is not adequate to completely characterize the delamination pattern resulting from impact. The advent of high speed transient recorders with the ability to digitize, store and transfer large amounts of data has provided the capability to capture and analyze the entire signal without reducing the information content through the detection and filtering process. It was found that considerably better resolution, especially in the near surface region, is obtainable using relatively simple signal processing methods [4]. This paper will display the vast improvement possible when this advanced capability is utilized for the characterization of impact damage in composites, discuss the equipment and procedures required for its implementation as well as the limitations inherent in its application.

LAMINATES

The laminates used in this study were 32 ply $[0/+45/-45/90]_{4S}$ quasi-isotropic panels. Two material systems were chosen. The first was Hercules AS4/3501-6 graphite/epoxy, which represents a first generation brittle matrix system in common use on current aerospace structures. The second system chosen was Imperial Chemical Industries AS4/PEEK APC-2 graphite/thermoplastic composite. This system was chosen because of the ductile nature of the thermoplastic polyetheretherketone (PEEK) matrix, which has been shown to lead to significant improvements in the compression strength of impacted plates [5].

IMPACT EVENT

An impact energy of approximately 20 J (15 ft-lb) was selected for presentation in this paper because it produced the largest amount of internal damage along with minimal surface deformation and best demonstrates the power of the new NDE technique. The laminates were impacted at low velocity using the instrumented impact apparatus developed by Sjöblom [6], and is shown in Figure 1. The impact set-up consisted of a variable weight pendulum released from a pre-determined height to achieve the desired impact energy. The laminate incoming and exiting velocities were measured using a timing apparatus adjacent to the specimen. The impactor had a 12.7 mm (0.5 in.) diameter steel spherical tip. The laminates were simply supported by a 100 mm (3.94 in.) diameter ring. The

initial impact energy in the graphite/epoxy was 20.2 J (14.8 ft-lb). The energy of the rebounding impactor was 10.9 J (7.9 ft-lb). The 46% energy loss was due to delamination initiation and propagation, fiber breakage, transverse cracking, and vibration damping and dissipation through the support fixture. The graphite/PEEK plate was impacted at 20.3 J (14.9 ft-lb). The residual rebound energy in the impactor was 8.4 J (6.2 ft-lb), corresponding to a 59% energy loss.

EXPERIMENTAL NDE PROCEDURE

The impacted specimens were ultrasonically scanned using the laboratory scanning system shown schematically in Figure 3. The system consists of standard components such as a water tank in which the specimen is immersed in order to provide a coupling medium for the ultrasonic waves between the transducer and the specimen, a computer controlled motion control system to scan the transducer and a conventional ultrasonic pulser/receiver to excite the transducer and amplify the received ultrasonic waves. In addition to the standard components, a LeCroy Model 8828B 200 MHz Transient Recorder is used to digitize and store the ultrasonic waveforms. The entire system is controlled by a PDP 11/23 computer with commands and data sent over an IEEE-488 bus. The data is stored in a 30 MByte Winchester hard disk and displayed on a color video monitor, a laser printer or a color ink jet printer.

The specimen is scanned in a raster pattern using a focussed ultrasonic transducer. A relatively long focal length, 75 to 150 mm (3 to 6 in.) in water, is used in order to produce a narrow, collimated beam in the material which will give good lateral resolution. In contrast to conventional systems, the entire ultrasonic waveform is digitized at each point. No attempt is currently made to retain the entire waveform in memory, however, since this would produce intolerably large data files. Instead, one uses prior knowledge about the material and the defects or properties being sought. For the case of a planar laminated specimen and a delamination type defect, it is known that the defects will produce reflections which are out of phase with the input waveform and delayed in time by some integer multiple (N) of the round trip transit time (T) for an ultrasonic wave in a single layer. Thus, if one records the amplitude of the received signal at a time delays equal to NT relative to the largest peak of the front surface reflection (point R in Figure 4), the signal level at locations where the returning signal has been reflected from a delamination will be negatively (or positively if R is negative) perturbed relative to those points where there is no reflector present. Since the entire waveform has been digitized, we are able to record the amplitude at all points corresponding to an interface in the material and thus build up an image of the delaminations present on each layer in the material. In order to minimize the effect of electronic noise and variations in the thickness of each layer, a number of points (typically 3) on each side of the typical time delay for a given layer are examined by the computer and the most negative (or positive) value recorded. This results in a series

of software gate locations and widths as shown in Figure 4 (solid bars) being used for the acquisition. As can be seen from the waveform containing the flaw signal in Figure 4, there is a significant difference in the amplitude of the ultrasonic waveform at the gate location corresponding to the arrival time of the defect signal (indicated by the arrows) and much smaller differences in adjacent gates. For specimens with small amounts of curvature, a front surface tracking gate is utilized to ensure consistent time delays within the specimen (point F in Figure 4).

A key to being able to utilize the resolution achievable with this new data acquisition method is the ability to display the resulting image in a multilevel format, either color or grey scale. The method for display of the data which we found gave the best results required that a histogram of the data be calculated first to provide information on the frequency of occurrence of a given amplitude level. The resulting display utilized either an equal number of data points in each image amplitude range or split the image amplitudes into an equal percent of the entire overall data range. Although the number of data amplitude levels was 256, we typically limited the video display a maximum of 64 and 25 for hard copy output due to the inability of the human eye to perceive more and the limitations of the hard copy units. A suitable choice of levels allows one to emphasize different features in the resulting image.

RESULTS

In this section we will display selected results which illustrate the significant improvement in resolution achievable using the software gated ultrasonic method. The graphite/epoxy specimen was scanned using a 5 MHz, 50 mm (2 in.) focal length ultrasonic transducer which was defocussed from the front surface by 1/2 the specimen thickness. The defocussing results in a relatively collimated ultrasonic beam in the specimen for good lateral resolution.

The significant defect features (i.e. delaminations) are displayed on a ply-by-ply basis (Figure 5). Each delamination assumes a "peanut" or "bow-tie" shape which can be correlated with the material's properties and the impacting energy (7). The lower ply delaminations are not completely imaged as they are shielded or shadowed from the ultrasonic energy by the delaminations located in the previous interfaces. Figure 5(a) shows the damage at the first interface. Very little damage is detected here, the indication is most likely the slight impression resulting from the impact, but the subsequent layers show a continual increase in the severity of damage. Figure 5(b) displays a striking example of the peanut-shaped delamination along the +45 degree direction at the second interface. The third interface (Figure 5(c)) shows the effect of the adjacent plies as the delamination pattern turns 45 degrees and increases in size. Also evident, is the effect of the previous interface's delamination as the ply separation under the prior interface cannot be imaged. A careful

examination of the area adjacent to the shadow of the delamination in the upper layer shows some evidence that it bridged down into this layer through a crack between the fibers (4). Subsequent damage in further layers is apparent in Figures 5(d,e,f) as the damaged area continues to increase in size and changes orientation. The bridging phenomenon is especially evident in Figure 5(d).

In order to illustrate how different materials can be analyzed and compared, data from a second material was collected. A graphite/PEEK composite panel was scanned using a 10 MHz, 75 mm (3 in.) focal length ultrasonic transducer, also defocussed to the midpoint of the specimen thickness. A different transducer was used to show that superior resolution could be obtained utilizing the software gating technique at various frequencies. Figure 6 shows the delaminations present in the first six interfaces.

The cumulative effect of these delaminations is outlined in Figure 7. These maps of the individual delaminations were made for this figure by hand tracing the outline of the unshadowed portion of each delamination between subsequent plies. The images shown in Figures 5 and 6 as well as images collected for the other layers which were not shown for the sake of brevity provided data for these maps. Future maps will be made using a software image subtraction method which is currently under development. The layer-by-layer impact damage in the graphite/epoxy panel (Figure 7(a)) is summarized opposite the damage resulting from a similar impact load in a

thermoplastic matrix material (Figure 7(b)). The ultrasonic damage mapping presents an excellent procedure to document the differences between these two materials. The ply-by-ply imaging enables the investigator to follow growth of delaminations through the damaged area. The assembly of those images allows one to see the cumulative effect of damage through the material and recognize any significant patterns that may exist.

CONCLUSIONS

The value of digitizing rf ultrasonic data with software processing has been demonstrated in this paper. The method has the ability to resolve closely spaced delaminations which could not be separated using conventional methods and should provide the composite developers with a powerful tool to characterize the behavior of new materials.

The damage imaged in this paper portrays several significant features of the impact damaged areas provides insight into the damage tolerance of these materials. Layer-by-layer images display the growth of the internal damage. Figure 6 displays the cumulative damage effect in each material. Layers affected by impact forces which resulted in delamination appear in regular patterns as one follows the damage radially from the center of the impact site. The affected layers continually appear in various multiples of four. This correlates with the lay-up intervals for these quasi-isotropic materials as damage tends to cluster in the radial direction of the adjacent ply's orientation. Destructive evaluation [4]

displays this regular pattern of delaminations. The energy absorption capabilities of the two matrix materials (PEEK and epoxy) is graphically shown in Figure 6. The graphite/epoxy composite distributes the impact load to a larger area within the material. This load is also more evenly distributed through the material as evidenced by the larger quantity of layer delaminations (Figure 6(a)). The thermoplastic PEEK matrix contains the damage to a smaller cross-sectional area in Figure 6(b) as damage does not spread out far from the impact site and is more concentrated on several interfaces. This correlates with the design preference of the more ductile PEEK matrix absorbing the energy and containing the damage to a limited area.

ACKNOWLEDGEMENTS

We would like to thank R. Martin and B. Frock of the Univ. of Dayton Research Inst. for providing assistance with the software and image processing and M. Ruddell, UDRI, for providing assistance with the data collection.

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5. Carlile, D.R., and Leach, D.C., "Damage and Notch Sensitivity of Graphite/PEEK Composites," 15th National SAMPE Technical Conference, pp.82-93, October 1983.

6. Sjöblom , P., "Simple Design Approach Against Low-Velocity Impact Damage," 32nd International SAMPE Symposium, pp. 529-539, April, 1987

7. Liu, D., "Impact-induced Delamination - A View of Bending Stiffness Mismatching" (Preprint, submitted to Journal of Composite Materials) June, 1987.

Figure Captions

1. Low-velocity impact apparatus from Sjöblom [6].

2. High resolution ultrasonic scanning system

3. Ultrasonic waveforms at unflawed and flawed locations in specimen; vertical axis - amplitude, horizontal axis - time, time position F - front surface tracking location, time position R - front surface reference location, arrows indicate third gate location.

4. Ultrasonic images of the defect patterns at selected interfaces of a 32 ply graphite/epoxy specimen; (a) through (e) - interfaces 1 through 5, (f) - interface 7, delaminations show up as the darkest (highest amplitude) area in each image.

5. Ultrasonic images of the defect patterns at selected interfaces of a 32 ply graphite/PEEK specimen; (a) through (f) - interfaces 1 through 6, delaminations show up as the darkest (highest amplitude) area in each image.

6. Delamination maps generated from the ultrasonic images showing ply-by-ply patterns; (a) graphite/epoxy specimen, (b) graphite/PEEK specimen.

FIGURE 1

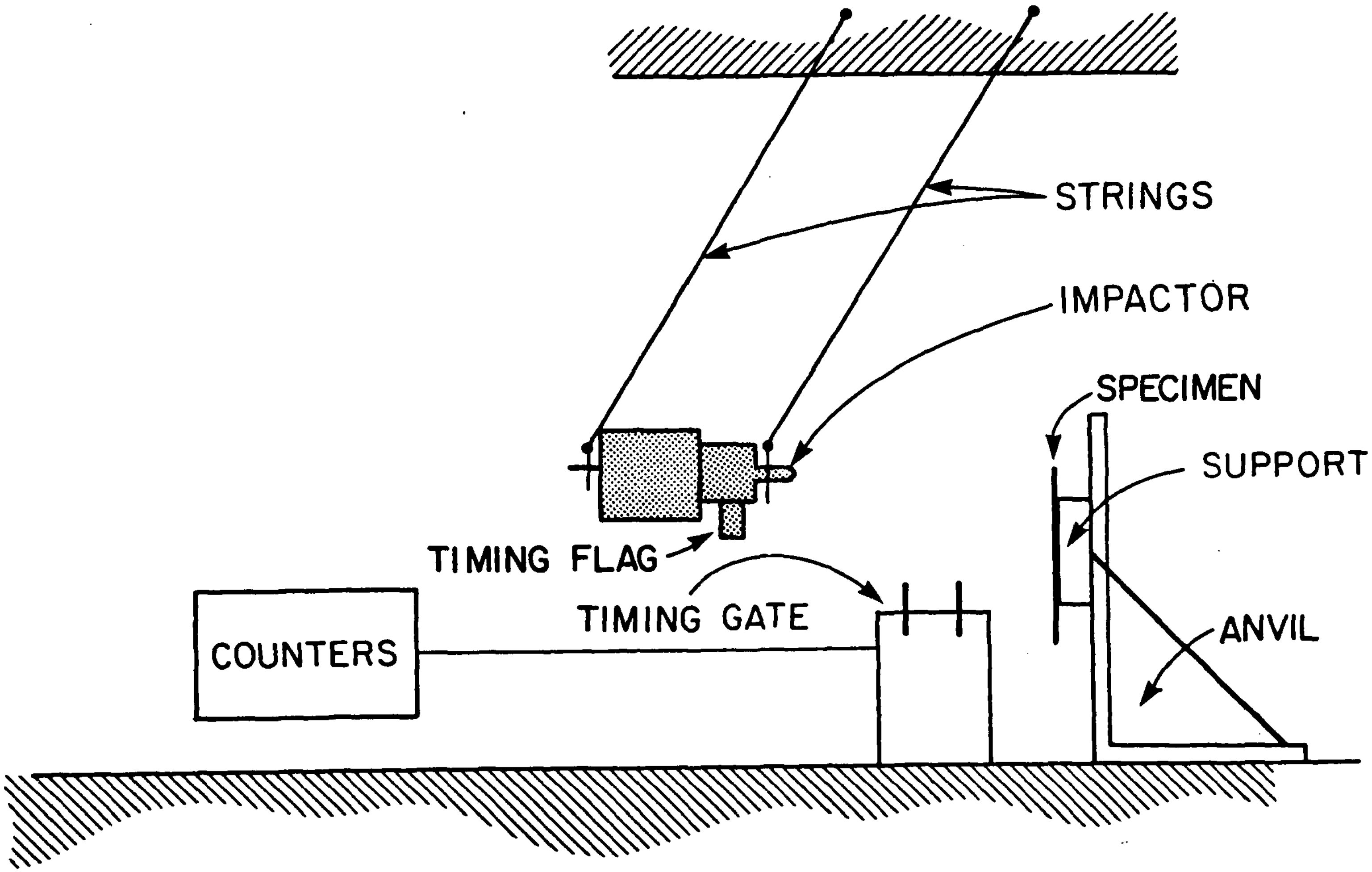

STRINGS
IMPACTOR
SPECIMEN
SUPPORT
TIMING FLAG
TIMING GATE
ANVIL
COUNTERS

FIGURE 2

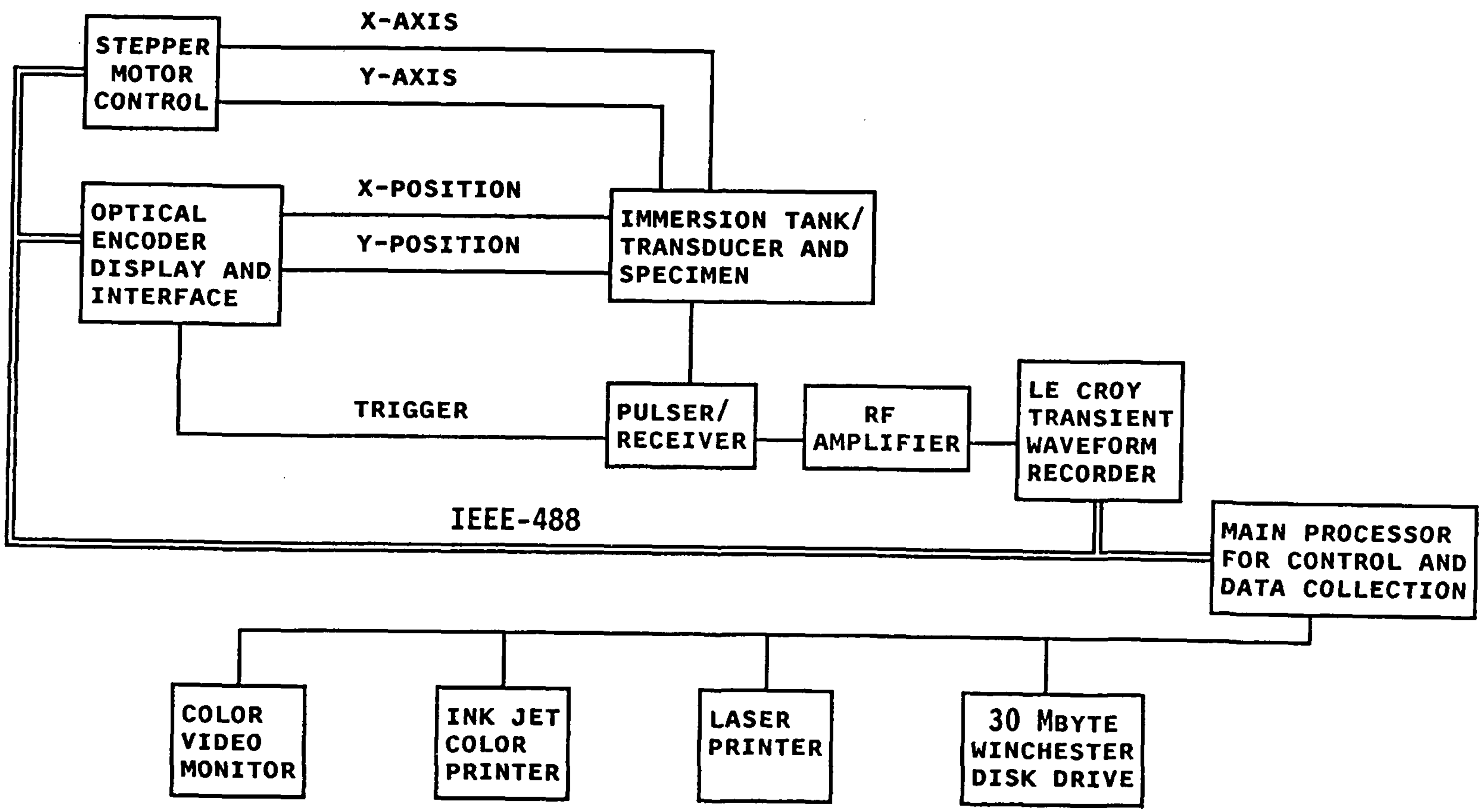
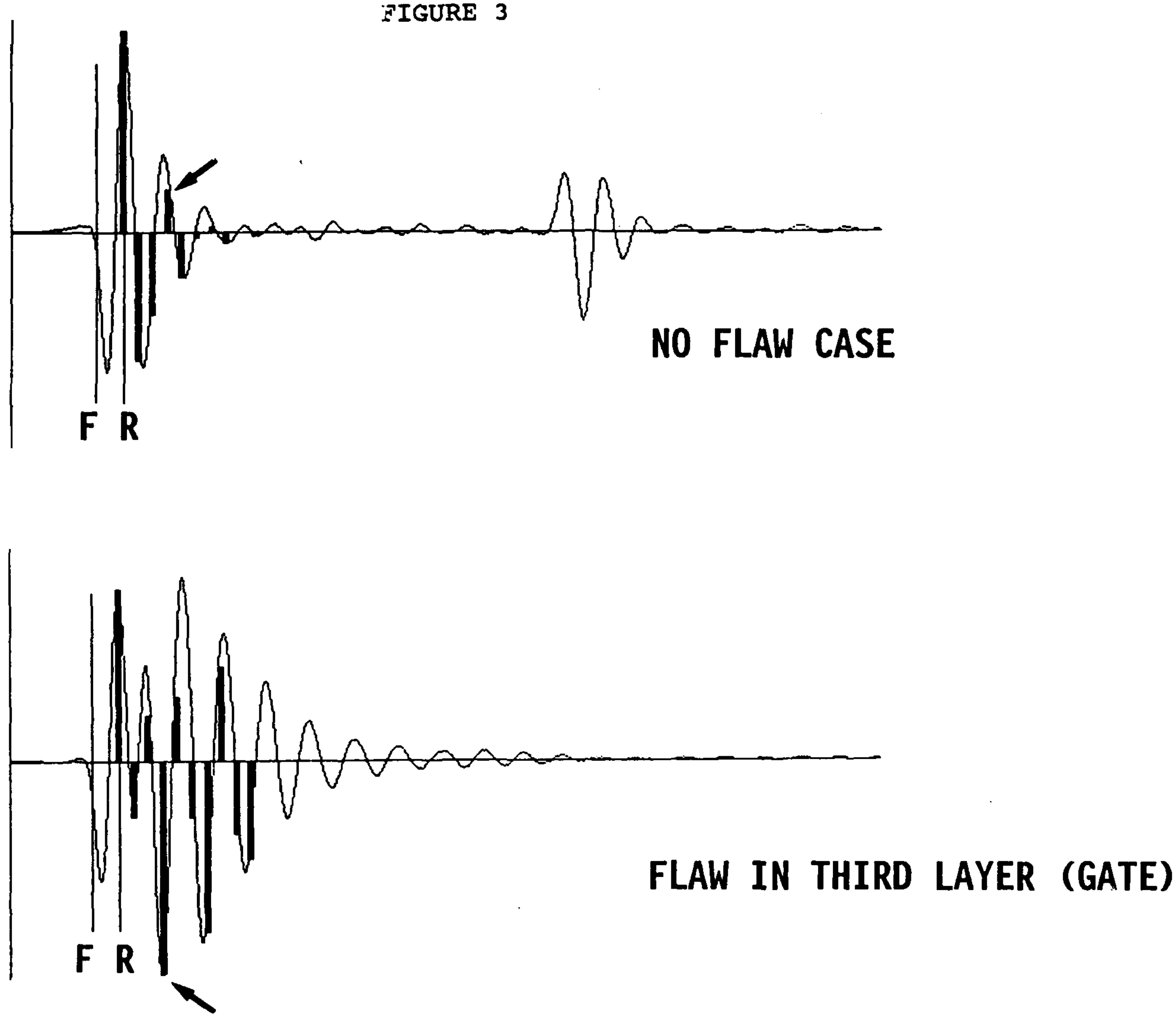

STEPPER MOTOR CONTROL
X-AXIS
Y-AXIS
OPTICAL ENCODER DISPLAY AND INTERFACE
X-POSITION
Y-POSITION
IMMERSION TANK/ TRANSDUCER AND SPECIMEN
TRIGGER
PULSER/ RECEIVER
RF AMPLIFIER
LE CROY TRANSIENT WAVEFORM RECORDER
IEEE-488
MAIN PROCESSOR FOR CONTROL AND DATA COLLECTION
COLOR VIDEO MONITOR
INK JET COLOR PRINTER
LASER PRINTER
30 MBYTE WINCHESTER DISK DRIVE

FIGURE 3



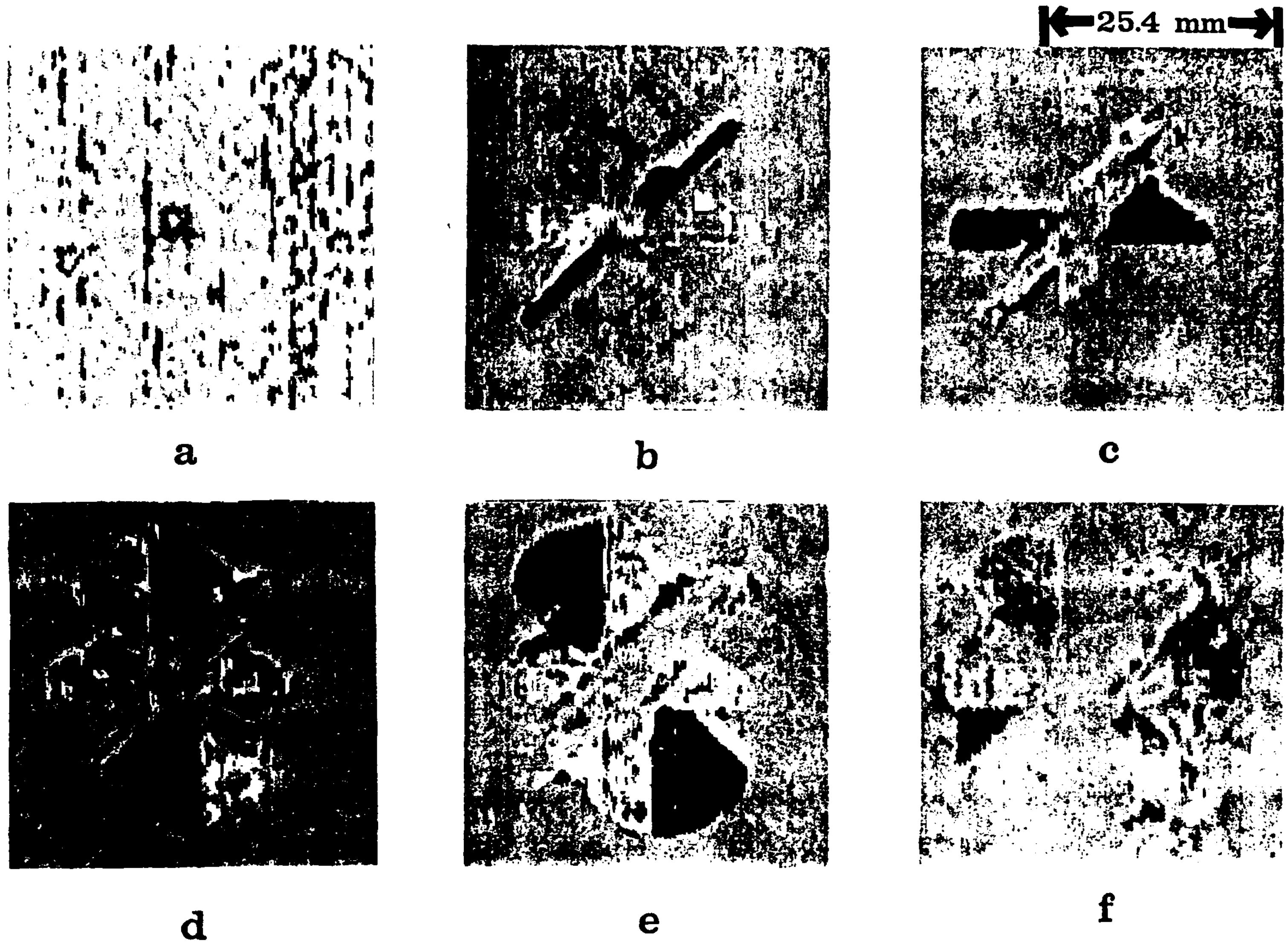


FIGURE 4

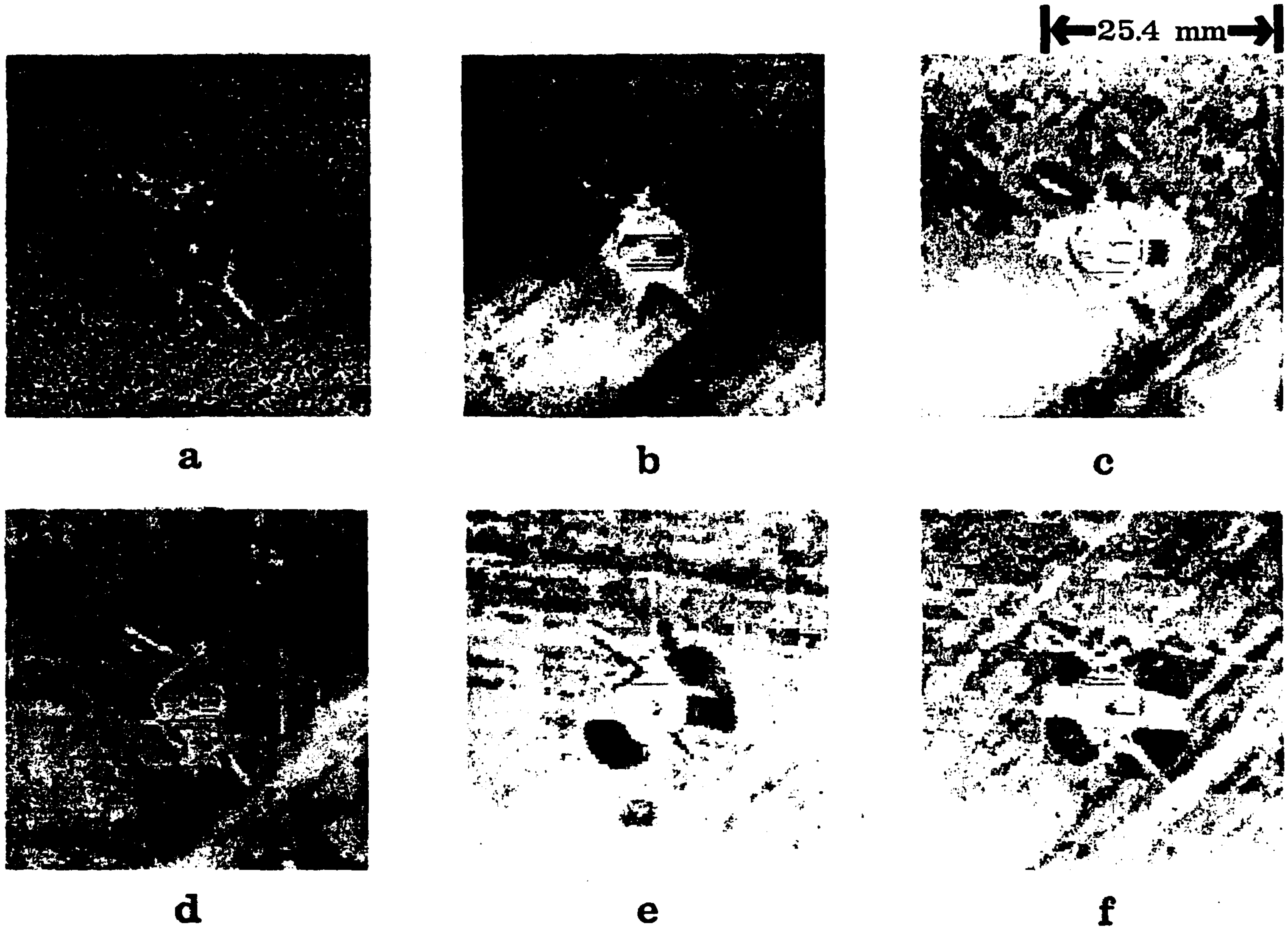


FIGURE 5

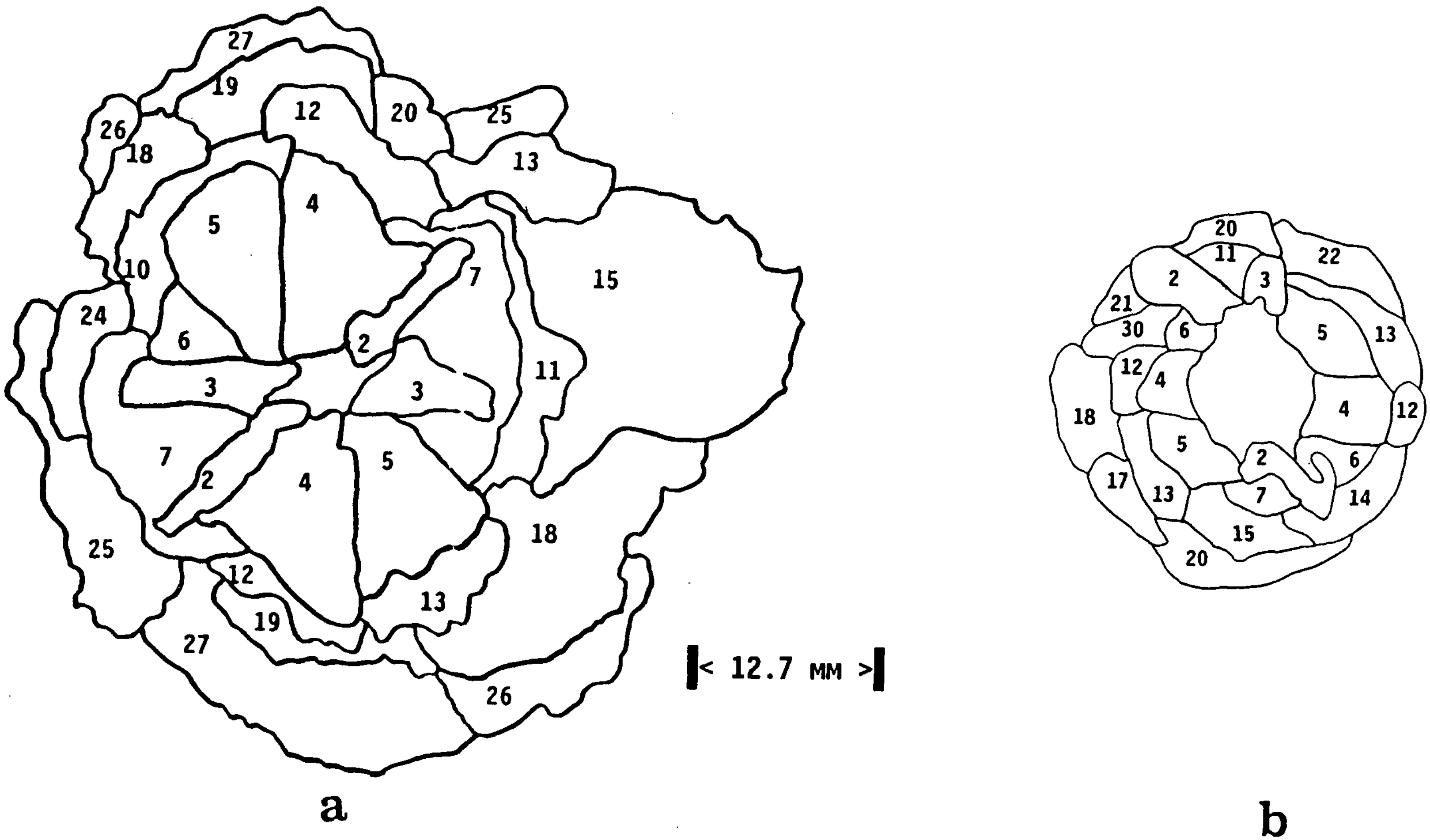


FIGURE 6

SESSION II: METALLIC STRUCTURES

Damage Tolerance In Pressurized Fuselages

A LESSONS LEARNED REVIEW OF PRESSURIZED FUSELAGE DEVELOPMENT FROM A LARGE DAMAGE TOLERANCE VIEWPOINT

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STRUCTURAL INTEGRITY CONFERENCE

SAN ANTONIO, TEXAS 1-3 DECEMBER 1987

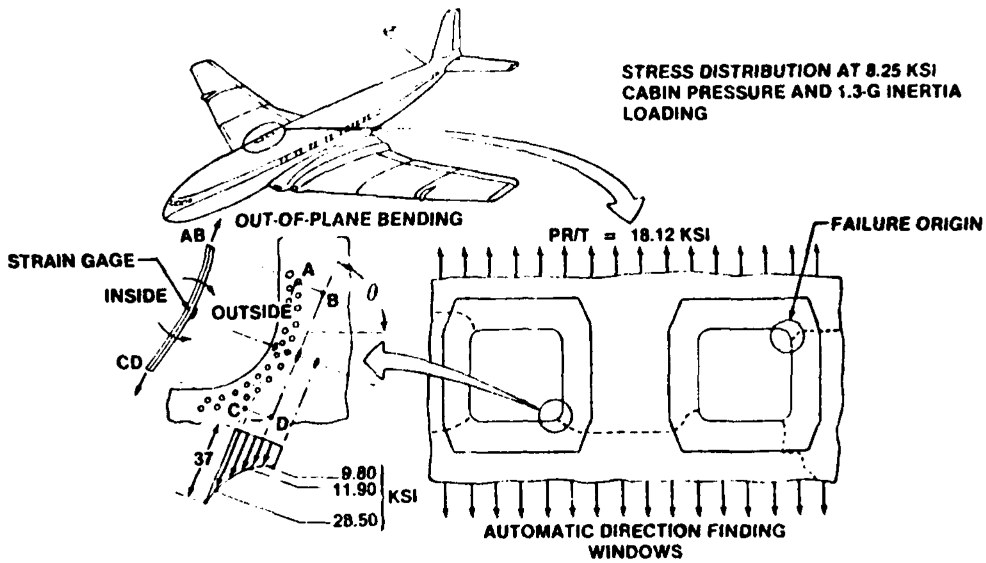
COMET I YOKE PETER - FIRST JET TRANSPORT AIRCRAFT TO ENTER SCHEDULED AIRLINE SERVICE

DESIGN LIFE VERSUS HIGH TIME FOR COMMERCIAL TRANSPORTS

AIRCRAFT	DESIGN LIFE		HIGH TIME		AS OF DATE
	HOURS	FLIGHTS	HOURS	FLIGHTS	
DC-8	50,000	25,000	74,050	43,604	SEP 1986
DC-9	30,000	40,000	58,512	83,798	SEP 1986
DC-10	60,000	42,000	55,686	20,109	SEP 1986
L-1011	60,000	36,000	37,001	21,249	JUN 1986
707	60,000	30,000 (1)	76,285	35,235	SEP 1986
720	60,000	50,000	67,745	43,588	SEP 1986
727	60,000	60,000	65,814	64,227	SEP 1986
737	45,000	75,000	58,450	81,689	SEP 1986
747	60,000	20,000	67,048	24,241	SEP 1986

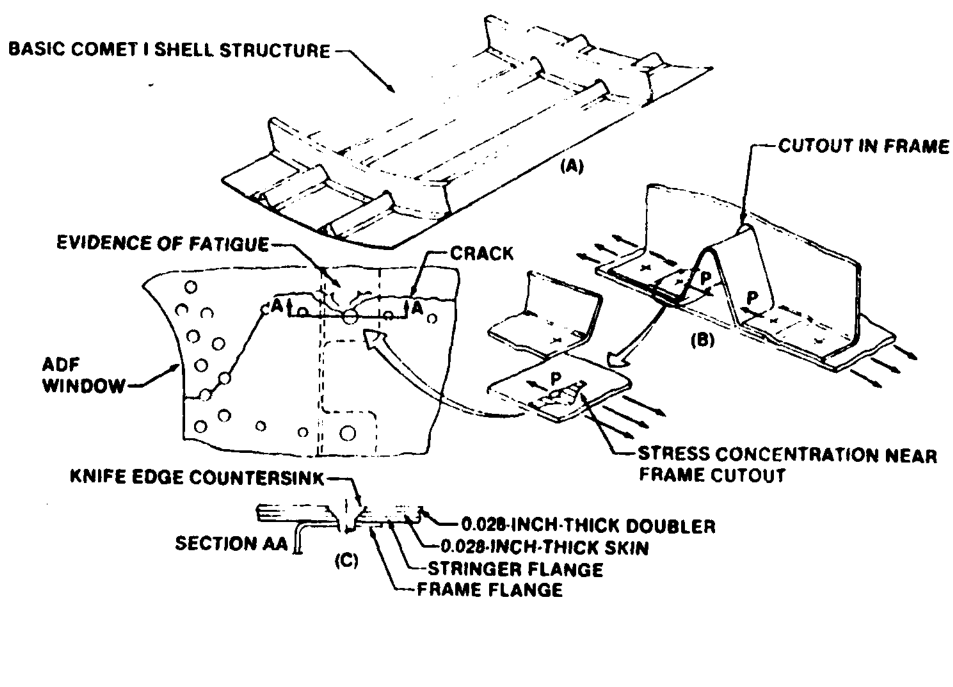
(1) 50,000 FOR SOME MODELS

PROBABLE FAILURE ORIGIN COMET I YOKE PETER

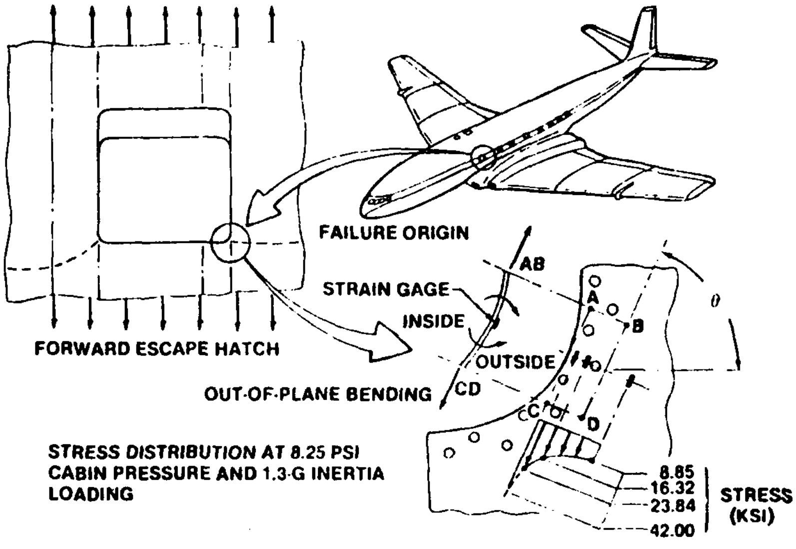


BASIC COMET I SHELL CONFIGURATION

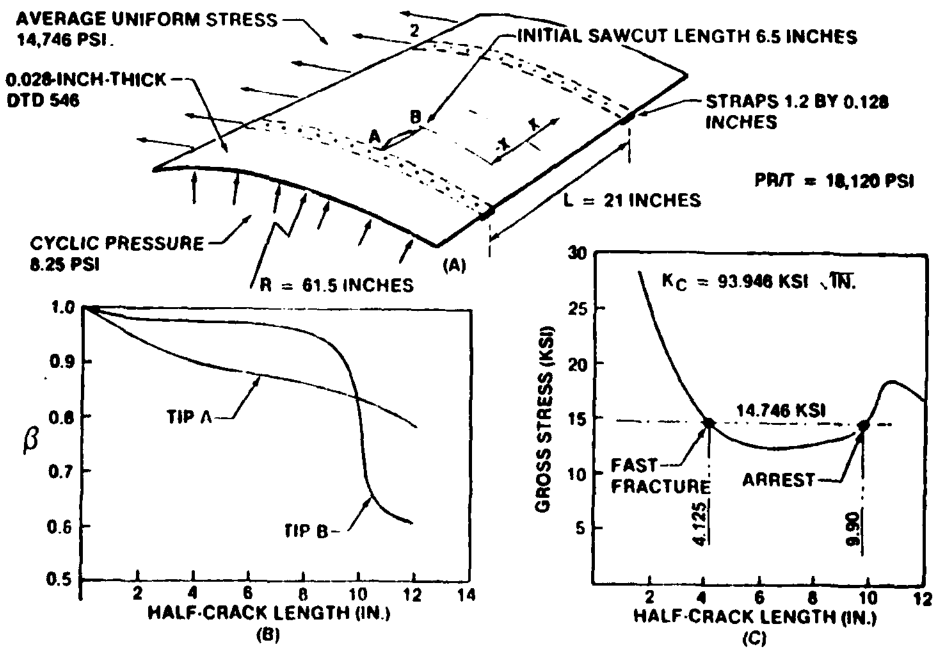
PROBABLE FAILURE ORIGIN - YOKE PETER



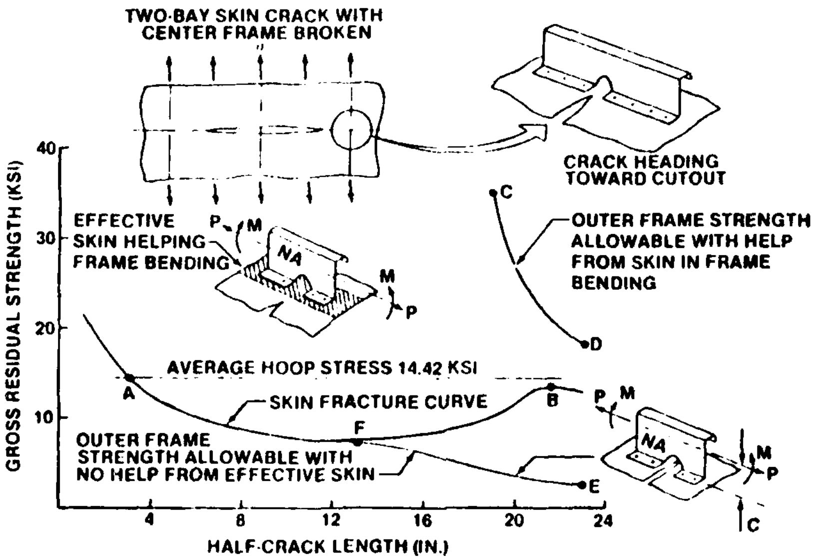
PROBABLE FAILURE ORIGIN
COMET I YOKE UNCLE TEST AIRCRAFT



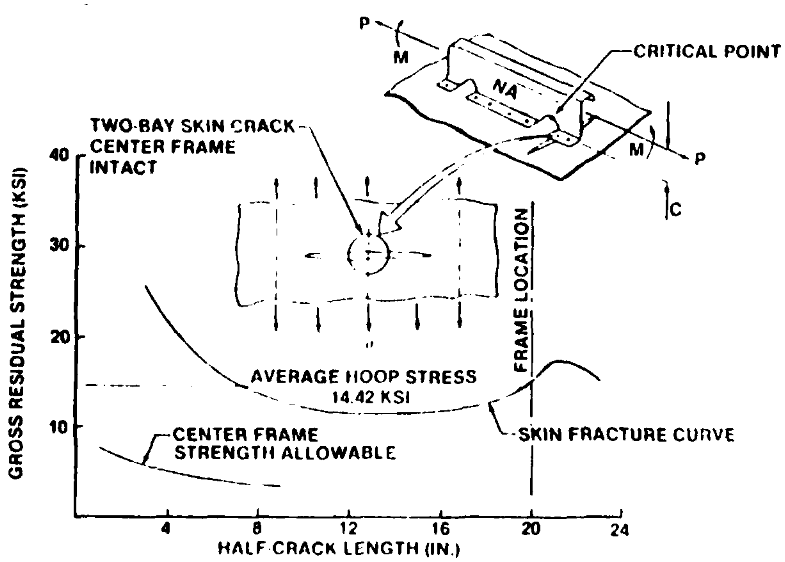
RESIDUAL STRENGTH ANALYSIS AND TEST RESULTS
ON COMET I FUSELAGE WITH CRACK STOPPER STRAPS
(WILLIAMS TEST)



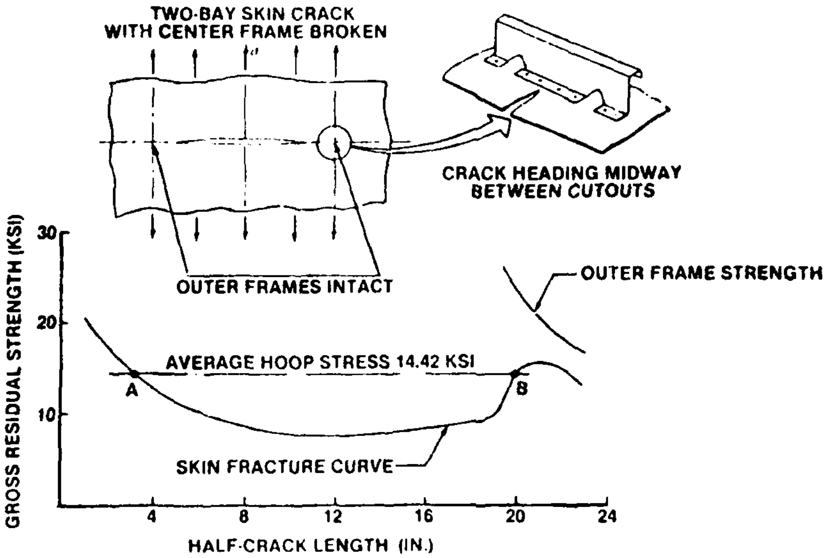
RESIDUAL STRENGTH DIAGRAM FOR TWO-BAY CRACK WITH CENTER FRAME BROKEN - CRACK HEADING TOWARD NOTCH (COMET I TYPE CONFIGURATION)



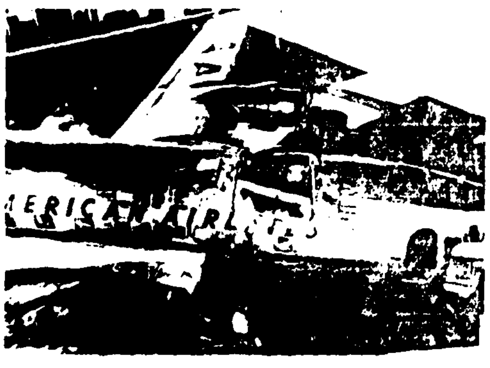
RESIDUAL STRENGTH DIAGRAM FOR TWO-BAY CRACK WITH CENTER FRAME INTACT - CRACK HEADING TOWARD NOTCH (COMET I TYPE CONFIGURATION)



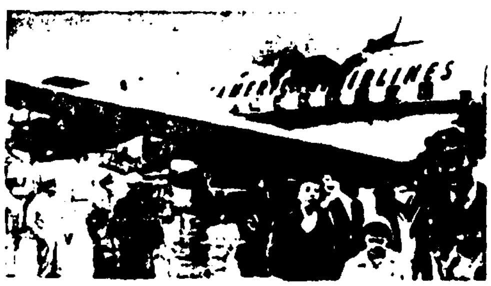
RESIDUAL STRENGTH DIAGRAM FOR TWO-BAY CRACK WITH CENTER FRAME BROKEN - CRACK HEADING BETWEEN CUTOUTS (COMET I TYPE CONFIGURATION)



DC-6 AND DC-7 PROPELLER BLADE FAILURE INCIDENTS

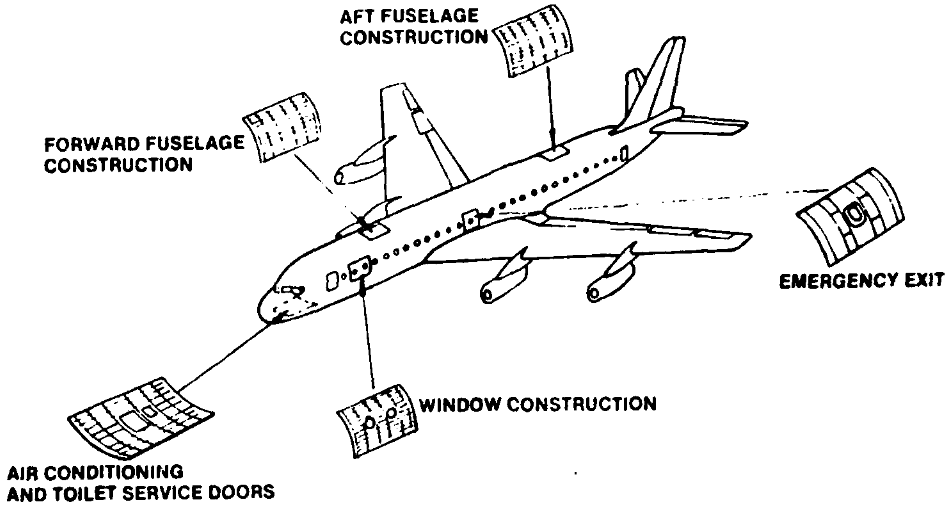


(A) PROPELLER BLADE DAMAGE, PRESSURIZED DC-6, NEAR DENVER, 22 AUGUST 1950

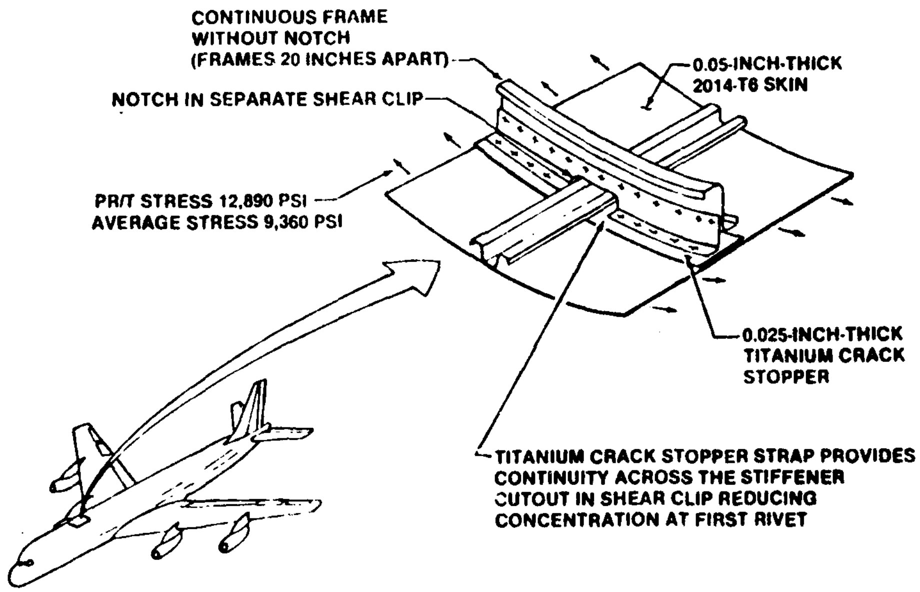


(B) PROPELLER DAMAGE, PRESSURIZED DC-7, NEAR MEMPHIS, 5 MARCH 1957

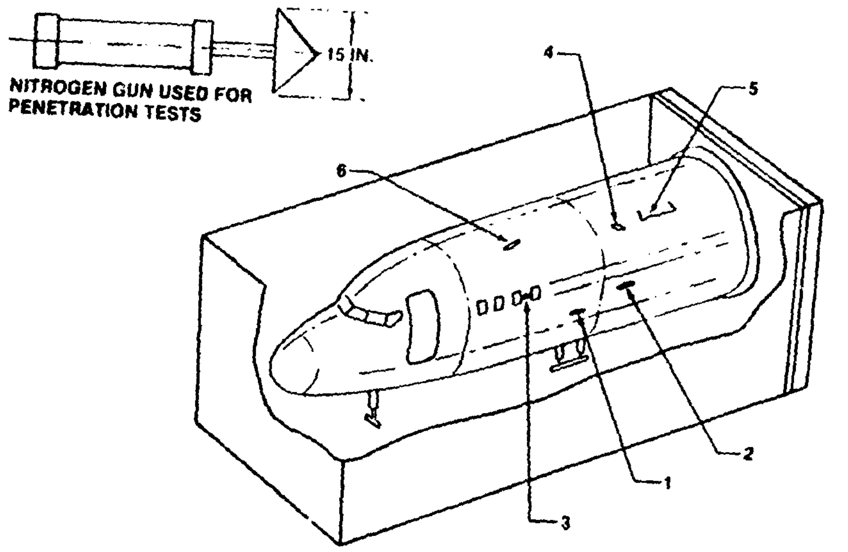
DC-8 FUSELAGE FATIGUE TEST PANELS REPRESENTING VARIOUS AREAS OF THE AIRPLANE



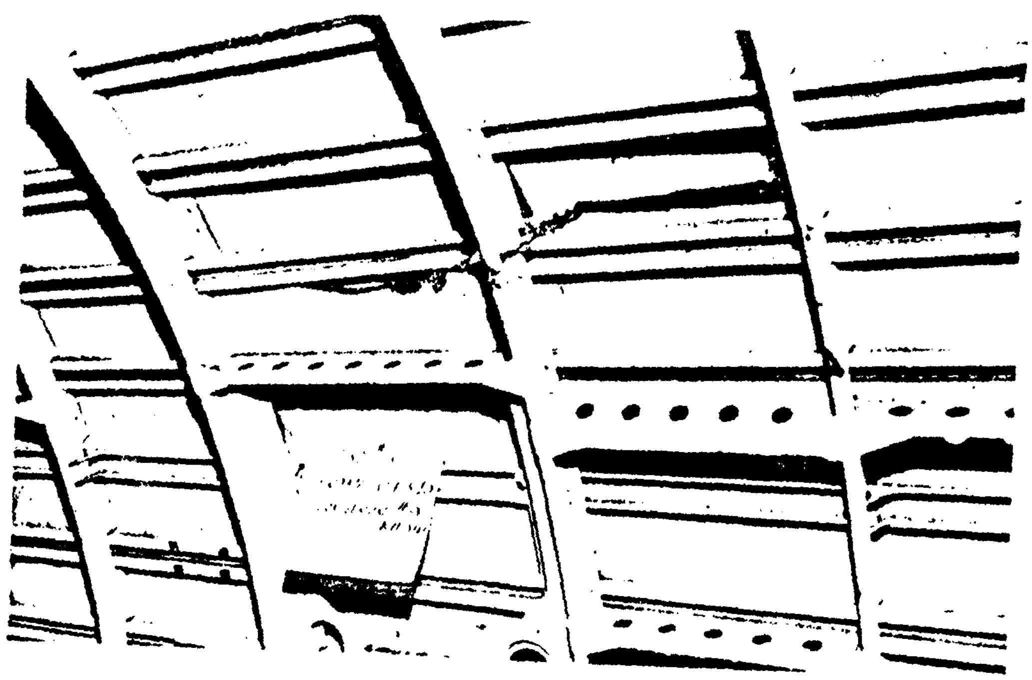
MINIMUM GAUGE CONSTRUCTION FOR DC-8 FUSELAGE



DC-8 WEDGE PENETRATION TESTS

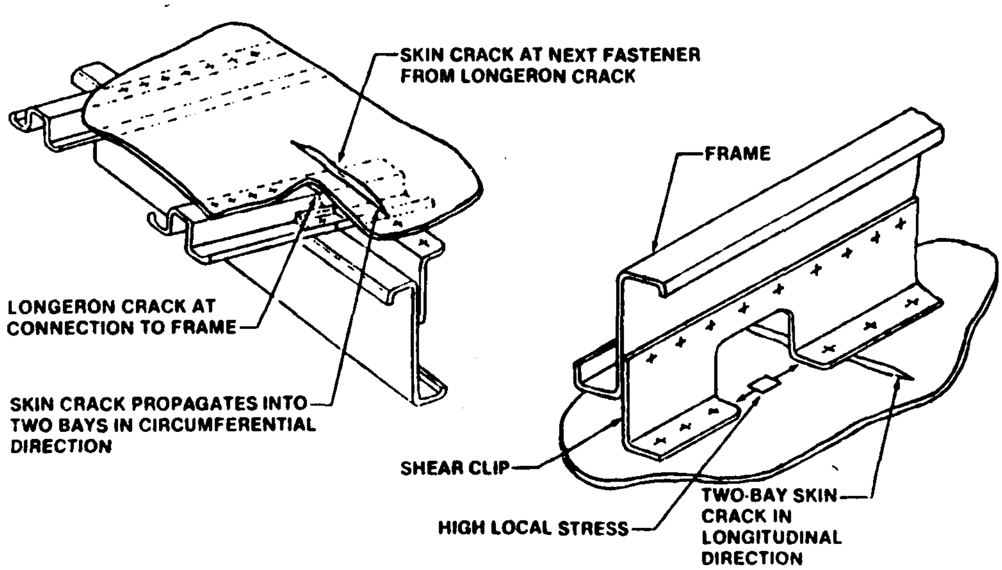


TWO BAY DAMAGE WITH BROKEN FRAME AND LONGERON (INSIDE VIEW)

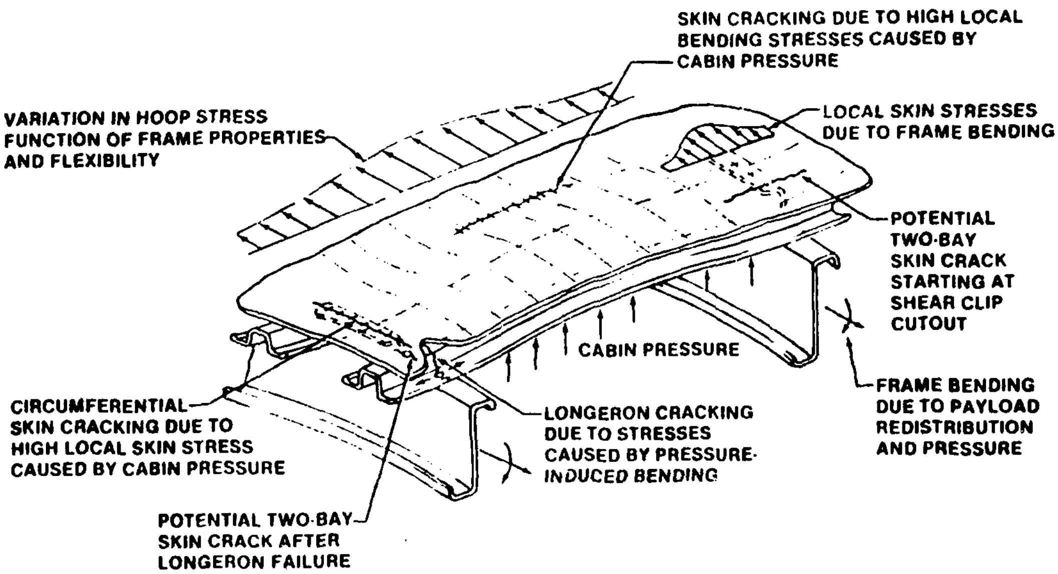


TWO BAY SKIN CRACK SHOWING FLAPPING (EXTERNAL VIEW)

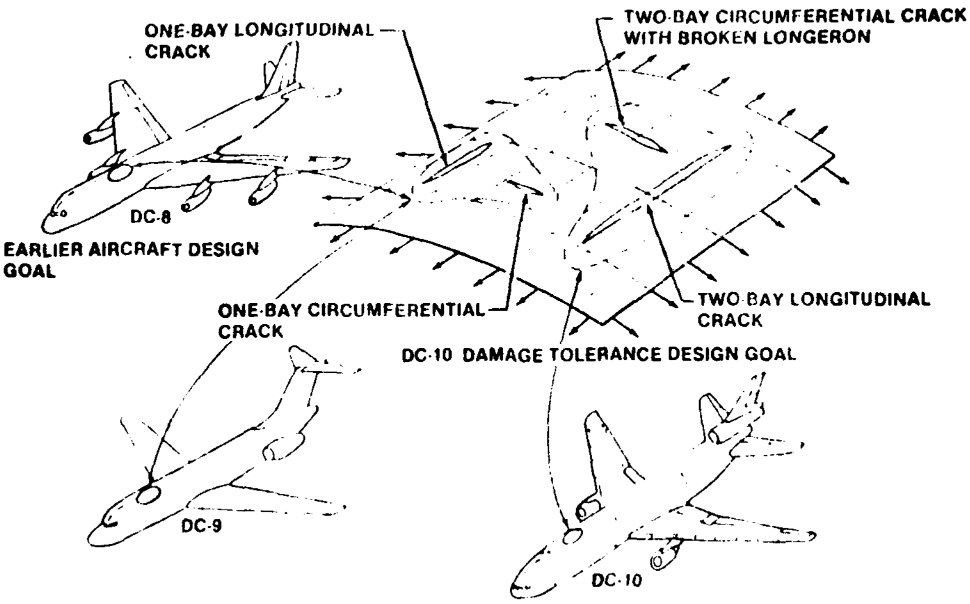
POSSIBLE SKIN FATIGUE CRACKING SCENARIOS IN CIRCUMFERENTIAL AND LONGITUDINAL DIRECTIONS



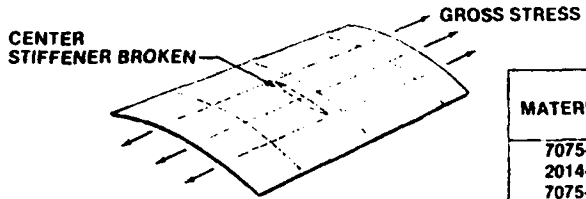
FATIGUE SENSITIVE AREAS IN FUSELAGE BASIC SHELL STRUCTURE



DAMAGE TOLERANCE DESIGN GOALS FOR FUSELAGE SKIN



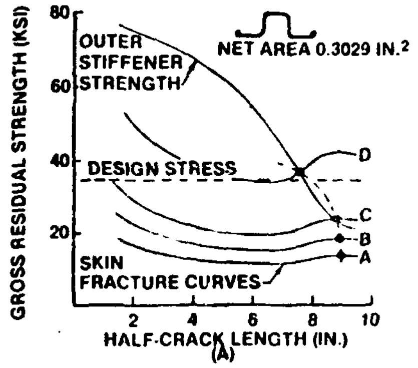
RESULTS OF FINITE ELEMENT ANALYSIS FOR CANDIDATE SKIN MATERIALS - TWO BAY CRACK WITH BROKEN LONGERON



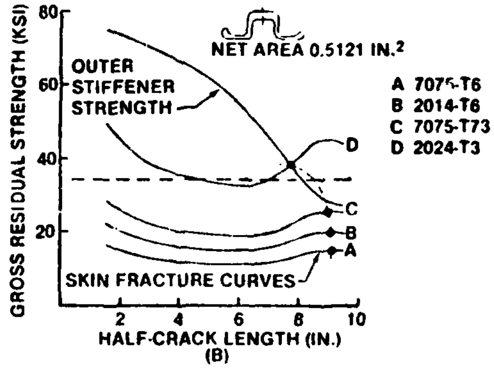
STIFFENER SPACING 8 INCHES

MATERIAL	CALCULATED ALLOWABLE (KSI)	
	SMALL HAT	LARGE HAT
7075-T6	13.60	14.72
2014-T6	18.40	19.52
7075-T73	23.52	25.28
2024-T3	36.48	38.72

LIMIT DESIGN STRESS REQUIRED
FOR THIS DAMAGE — 34 KSI

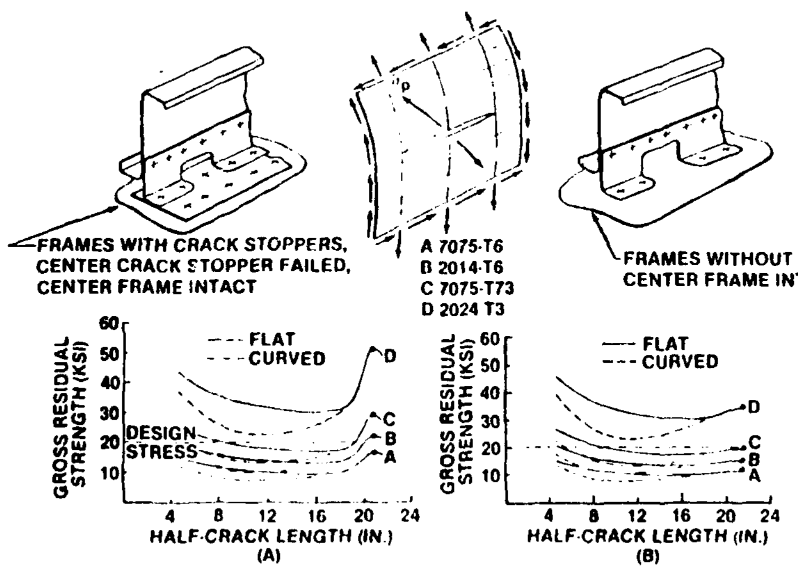


(A)

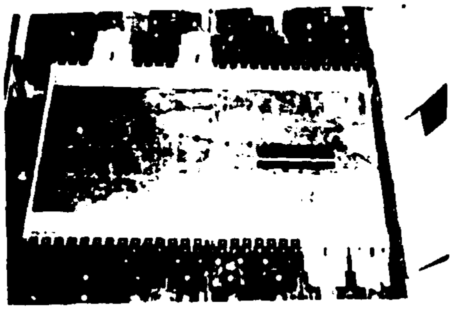


(B)

FINITE ELEMENT RESULTS - LONGITUDINAL CRACK CASE

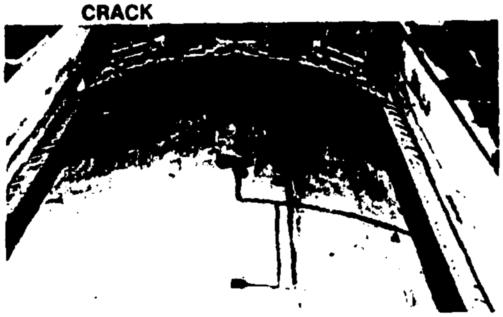


DEVELOPMENT TEST PANELS FOR LARGE DAMAGE SIMULATION



(A) FLAT PANELS — LONGITUDINAL CRACK

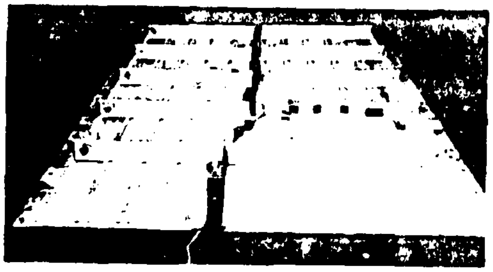
(B) FLAT PANELS — CIRCUMFERENTIAL CRACK



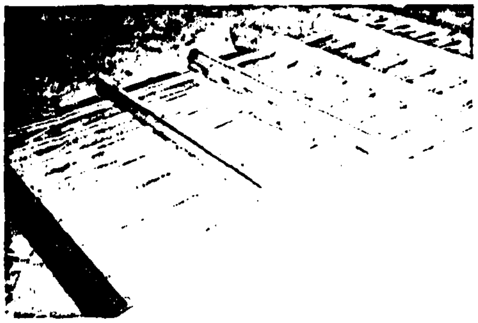
(C) CURVED PANELS

(D) VACUUM TEST MACHINE — CURVED PANELS

TYPICAL EXAMPLES OF PANELS CONTAINING LARGE DAMAGE LOADED TO FAILURE



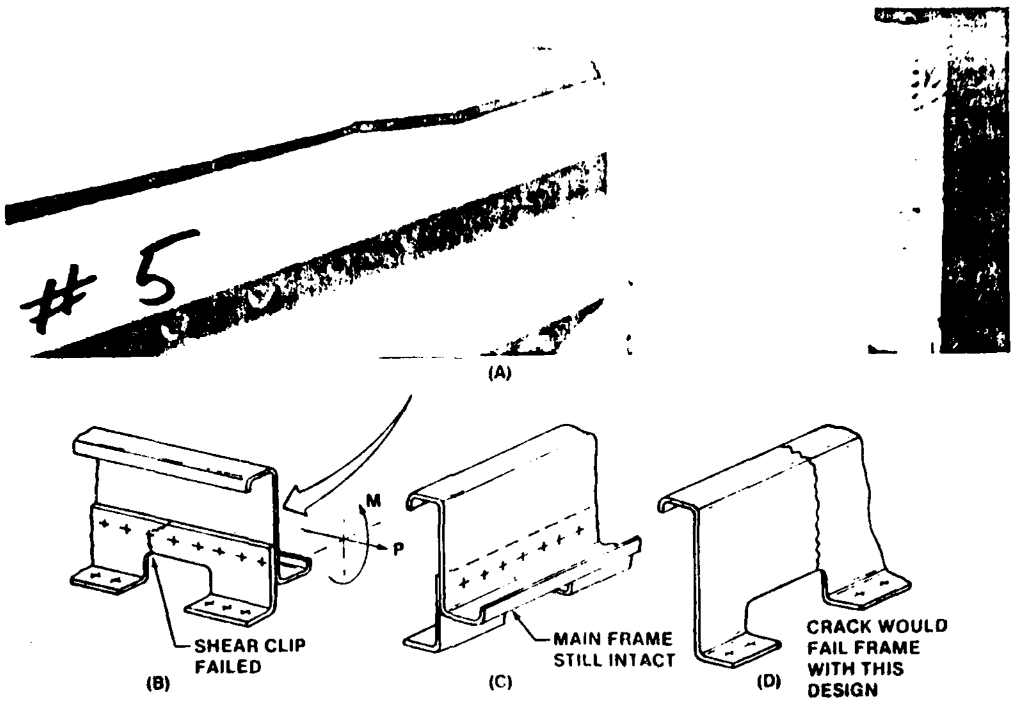
(A) FLAT PANEL SIMULATING LONGITUDINAL DAMAGE



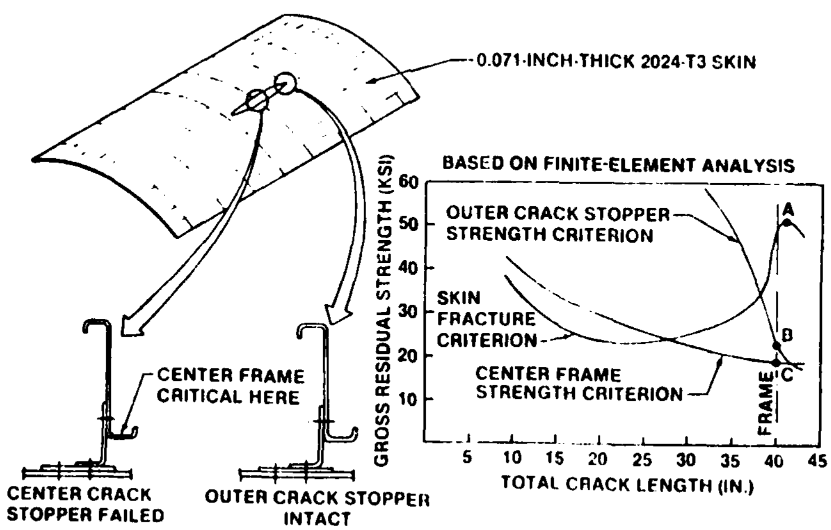
(B) FLAT PANEL SIMULATING CIRCUMFERENTIAL DAMAGE

(C) CURVED PANEL AFTER FAILURE FROM LONGITUDINAL DAMAGE

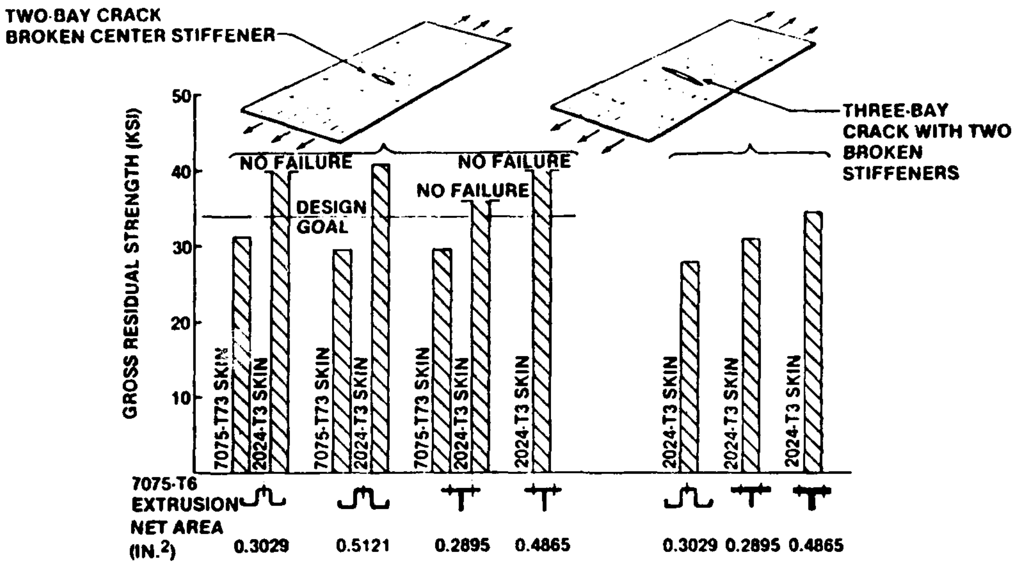
FLAT PANEL AFTER ARREST OF TWO BAY LONGITUDINAL CRACK



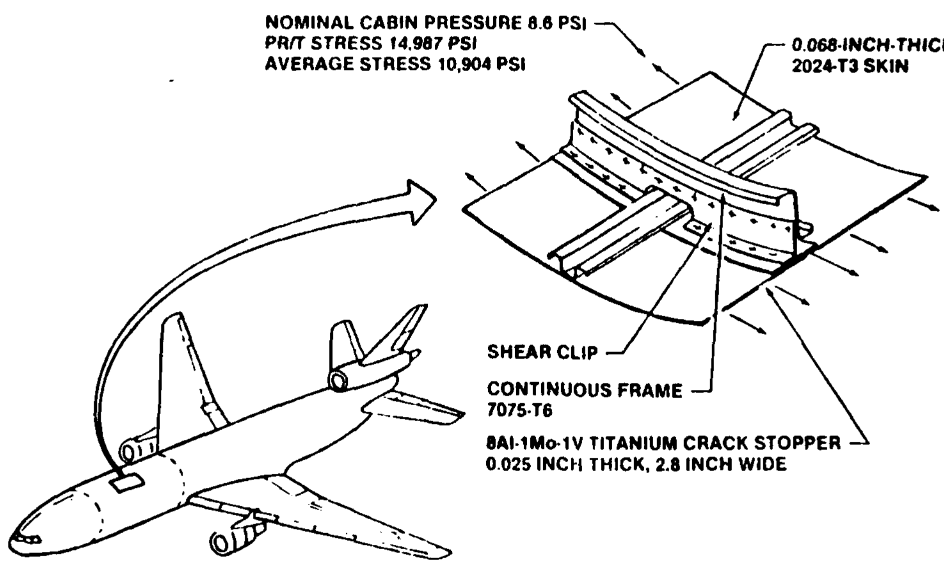
RESIDUAL STRENGTH FOR LONGITUDINAL CRACK IS LIMITED BY STIFFENER STRENGTH



TEST RESULTS FOR TWO BAY CIRCUMFERENTIAL CRACK WITH BROKEN CENTRAL LONGERON



MINIMUM GAUGE CONSTRUCTION FOR DC-10 FUSELAGE



Conclusions

FOR LONG LIFE PRESSURIZED FUSELAGE STRUCTURE WITH LARGE DAMAGE TOLERANCE CAPABILITY

- KEEP PR/t STRESS LEVELS LOW (15KSI)
- BE CAREFULL ABOUT CUTOUTS IN FRAME MEMBERS FOR AXIAL STIFFENERS - DESIGN THE CUTOUT IN A SEPARATE SHEAR CLIP LEAVING A CONTINUOUS FRAME MEMBER
- USE CRACK STOPPERS AT THE FRAME LOCATIONS AROUND THE ENTIRE CIRCUMFERENCE (TITANIUM)
- ACCOUNT FOR OUT OF PLANE BENDING STRESSES DUE TO SHELL CURVATURE NEAR CUTOUTS SUCH AS DOORS AND WINDOWS
- CONSIDER THE POSSIBILITY OF MULTI-SITE DAMAGE AHEAD OF THE LEAD CRACK
- USE DAMAGE RESISTANT MATERIALS (2024-T3)

F-16C FULL SCALE AIRFRAME DURABILITY TEST

PRESENTED AT

1987 USAF ASIP/ENSIP CONFERENCE, SAN ANTONIO

BY

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DURABILITY AND DAMAGE TOLERANCE ENGINEER

F-16 SPO
AERONAUTICAL SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OH

INTRODUCTION

0 BENEFITS OF TESTING

0 RATIONAL FOR F-16C TEST

0 COMPARISON OF F-16A/B VS F-16C/D

0 TEST PROGRAM

0 TECHNICAL FEATURES

0 IMPROVEMENTS OVER F-16A (FSD) TEST

0 NAVY PARTICIPATION

0 CURRENT STATUS

BENEFITS OF TESTING

O CONDUCT STRAIN SURVEY - YIELDS STRAINS FOR A KNOWN SPECTRUM LOAD CONDITION. IN CONTRAST:

- -- LOADS FLIGHT TEST YIELDS CALIBRATED LOADS FOR A KNOWN MANEUVER
- -- STATIC TEST STRAIN SURVEY YIELDS STRAINS FOR A KNOWN STATIC DESIGN CONDITION

O OBTAIN FRACTOGRAPHY

- -- CRACK SIZE VS. FLIGHT HOURS
- -- CRITICAL CRACK SIZE
- -- CRACK ARRESTMENT PROPERTIES OF THE AIRFRAME
- -- CRACK GROWTH RETARDATION/ACCELERATION

O EVALUATE POTENTIAL REPAIRS AND DESIGN CHANGES

BENEFITS OF TESTING (CONTINUED)

O UPDATE STRUCTURAL ANALYSIS MODELS

- -- FINITE ELEMENT MODELS
- -- CRACK GROWTH MODELS
- -- INDIVIDUAL AIRPLANE TRACKING MODELS

O DEMONSTRATE ECONOMIC LIFE

- -- AIRFRAME CRACKING BECOMES TOO WIDESPREAD TO ECONOMICALLY REPAIR

O IMPROVE FLEET MANAGEMENT

- -- INCREASES PREDICTIVE CAPABILITY
- -- INCREASES CONFIDENCE IN SAFETY AND ECONOMIC LIFE CALCULATIONS
- -- INCREASES PROBABILITY THAT INSPECTIONS AND MAINTENANCE WILL BE SCHEDULED FOR MAXIMUM BENEFIT

RATIONAL FOR SECOND F-16 TEST

- CHANGES SINCE THE FSD STAGE HAVE CREATED A NEED FOR A DURABILITY TEST THAT UTILIZES THE CURRENT CONFIGURATION AND ENVIRONMENT

  -- STRUCTURE - ACCOMODATE SYSTEMS, IMPROVE STRUCTURE
  -- WEIGHT - INCREASE IN SYSTEMS AND STRUCTURAL WEIGHT
  -- LOADS ENVIRONMENT - CHANGE IN USAGE AND MANEUVER ACTIVITY

- SERVICE LIFE BECAME GOAL FOR THE F-16C/D PROGRAM, THUS A GREATER POTENTIAL FOR PROBLEMS EXISTS

A/B VS. C/D PROGRAM COMPARISON

PROGRAM	A/B	C/D
MODEL TESTED	FSD	BLOCK 30
ENGINE	F100	F100/F110
MISSION MIX (HOURS)	55% A/A 20% A/G	28% A/A 57% A/G
SIMULATED LOADS ENVIRONMENT	DESIGN	FLIGHT/FLR MEASURED AND OPERATIONAL DATA
BASIC FLIGHT DESIGN G. W.	22,500 #	26,910 # (19.6% INC.)
MAX TAKEOFF G. W.	33,000 #	37,500 # (13.6% INC.)
USAF FLEET	785	1944
WORLDWIDE FLEET (INCL. USAF)	1514	2267

TEST PROGRAM

O TEST ARTICLE

-- STRUCTURALLY COMPLETE BLOCK 30 AIRFRAME
-- DUMMY LANDING GEARS, HORIZONTAL TAILS, AND ENGINE

O TEST HARDWARE

-- 115 LOAD RAMS, 90 BI-DIRECTIONAL
-- 235 LOAD DISTRIBUTIONS AVAILABLE, CAN BE COMBINED LINEARLY
-- 1024 CHANNELS STRAIN GAGES

O TEST USAGE

-- 3 LIVES OF 8,000 FLIGHT HOURS EACH
-- 28% A/A, 57% A/G
-- 5840 FLIGHTS/LIFE
-- TEST SPECTRUM CONSISTS OF A 500 HOUR BLOCK OF RANDOMLY ORDERED FLIGHTS REPEATED AS NECESSARY
-- AVERAGE CYCLING RATE - 30 LOAD POINTS/MIN, 168 LOAD POINTS/FLIGHT HOUR ==> 10 FLIGHT HOURS/CLOCK HOUR

TEST PROGRAM (CONTINUED)

O COMPUTERS

-- LOADS CONTROL: DEC PDP 11-84 AND MASSCOMP

--- PROVIDES COMMAND SIGNAL
--- MONITORS FEEDBACK
--- MONITORS LOAD AND POSITION LIMITS
--- CALIBRATES LOAD SYSTEM
--- PERFORMS ADAPTIVE CONTROL (MASSCOMP)

-- DATA ACQUISITION: DEC PDP 11-84

--- VERIFIES LOAD RAM COMMAND SIGNALS
--- RECORDS STRAINS AND LOADS
--- CALIBRATES STRAIN GAGES

-- LOAD CONTROL AND DATA ACQUISITION SOFTWARE DEVELOPED BY GENERAL DYNAMICS

TECHNICAL FEATURES

- APPLIED LOADS: AIR, INERTIA, ENGINE THRUST, GROUND, STORE
- COCKPIT AND FUEL TANK (WING AND FUSELAGE) PRESSURIZATION
- L. E. FLAP AND T. E. FLAPERONS ACTUATED UNDER LOAD
- RUDDER LOADED BUT NOT ACTUATED
- SOFTWARE INTEGRATION OF THE ERROR (FEEDBACK MINUS COMMAND)
  - ADJUSTS COMMAND SIGNAL "ON THE FLY"
  - PROVIDES SMOOTH CORRECTION SIGNAL WHICH ALLOWS FASTER CORRECTIONS
  - PERMITS FASTER LOADING RATES
  - PROVIDES QUICKER ACCEPTANCE
  - RESULTS IN FASTER CYCLING
  - PROVIDES BETTER TEST ARTICLE BALANCE

TECHNICAL FEATURES (CONTINUED)

O ADAPTIVE CONTROL/PHASE COMPENSATION

-- SOFTWARE IS CURRENTLY IN DEVELOPMENT
-- COMPUTER RECORDS RAM RESPONSE FOR EACH LOAD POINT
-- COMPUTER PRE-ADJUSTS COMMAND SIGNAL TO ACCOUNT FOR EACH RAM'S PARTICULAR RESPONSE
-- COMPUTER ATTEMPTS TO KEEP RAMS "IN STEP"
-- SHORTER ACCEPTANCE TIME FOR A GIVEN LOADING RATE
-- PERMITS FASTER CYCLING RATES

IMPROVEMENTS OVER FSD TEST

- SPECTRUM BASED ON OPERATIONAL USAGE
- ENGINE THRUST LOADS
- MOVING WING CONTROL SURFACES
- MORE LOAD POINTS/500 HOUR BLOCK
  - "A" TEST: 72,120
  - "C" TEST: 84,251
- SOFTWARE INTEGRATION OF THE ERROR SIGNAL
- TEST ARTICLE POSITIONING
  - "A" TEST POSITIONED USING 6 LOADS RAMS
  - "C" TEST POSITIONED BY DISTRIBUTING BALANCING LOADS AMONG ALL LOADS RAMS (IN DEVELOPMENT)

NAVY PARTICIPATION

0 NAVY PURCHASING 26 F-16C/D'S

0 NAVY'S "DESIGN USAGE" SIGNIFICANTLY DIFFERENT THAN USAF'S

0 NAVY DOES NOT BELIEVE THAT 2 LIVES OF USAF USAGE IS SUFFICIENT FOR NAVY PURPOSES

0 THEREFORE, NAVY FUNDING THIRD LIFE OF TEST - USAF USAGE

0 ADVANTAGES

-- NAVY DOES NOT HAVE TO FUND A SEPARATE TEST
-- AIR FORCE GETS EXTRA TESTING WITHOUT USING ADDITIONAL FUNDS

0 DISADVANTAGES

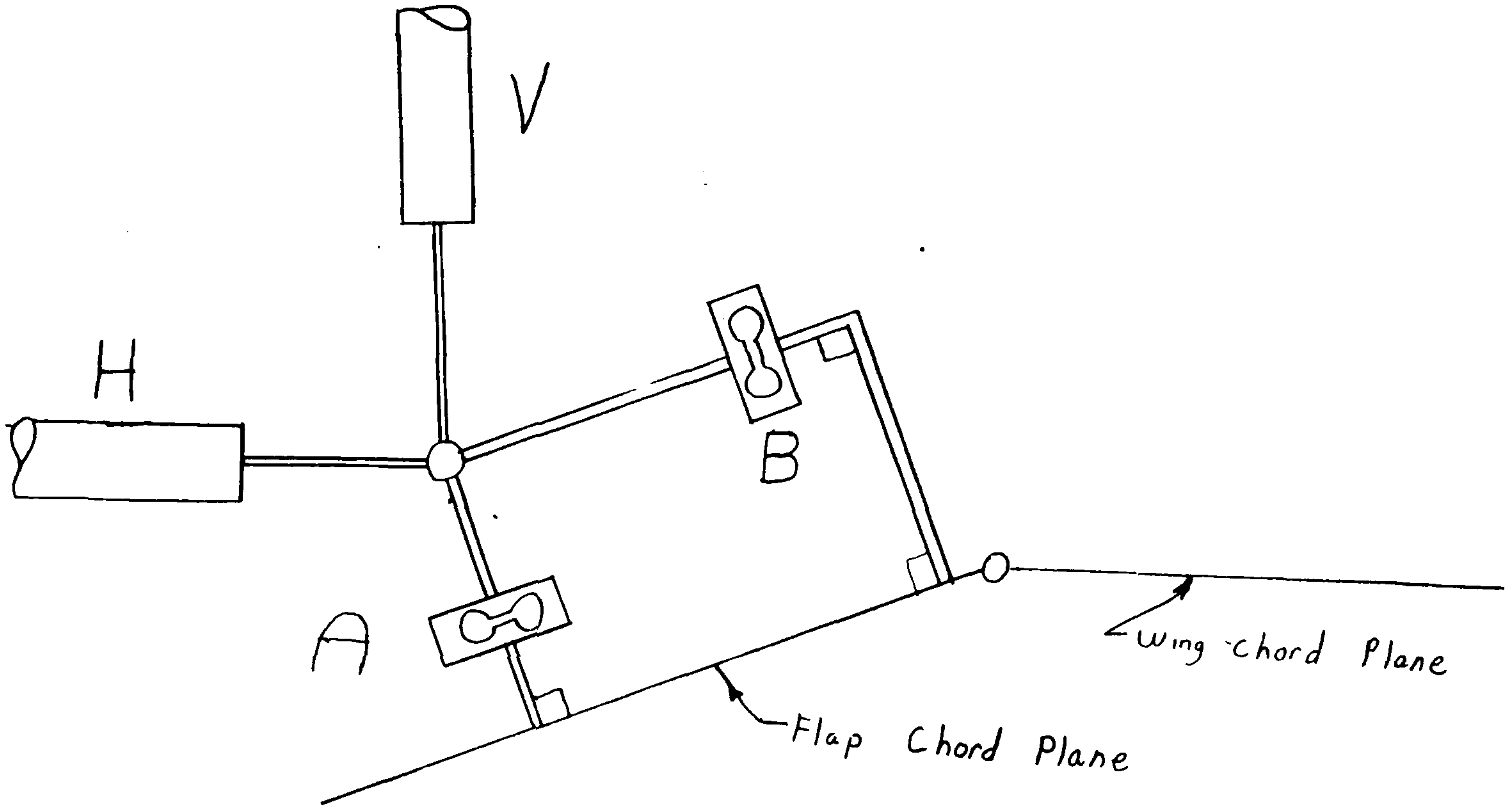
-- NOT TESTED TO NAVY USAGE
-- TEAR DOWN INSPECTION AND FINAL RESULTS DELAYED

CURRENT STATUS

- 800 HOURS AS OF 27 NOV 87
- 5 DAY - 2 SHIFT OPERATION
- AVERAGING 60 HOURS PER WORK SHIFT
- PEAK RATES OF 14 FLIGHT HOURS/CLOCK HOUR
- EXPECTED COMPLETION DATES
  - FIRST LIFE: MAY 88
  - SECOND LIFE: JAN 89
  - THIRD LIFE: JAN 90
  - TEAR DOWN INSPECTION: MAY 90

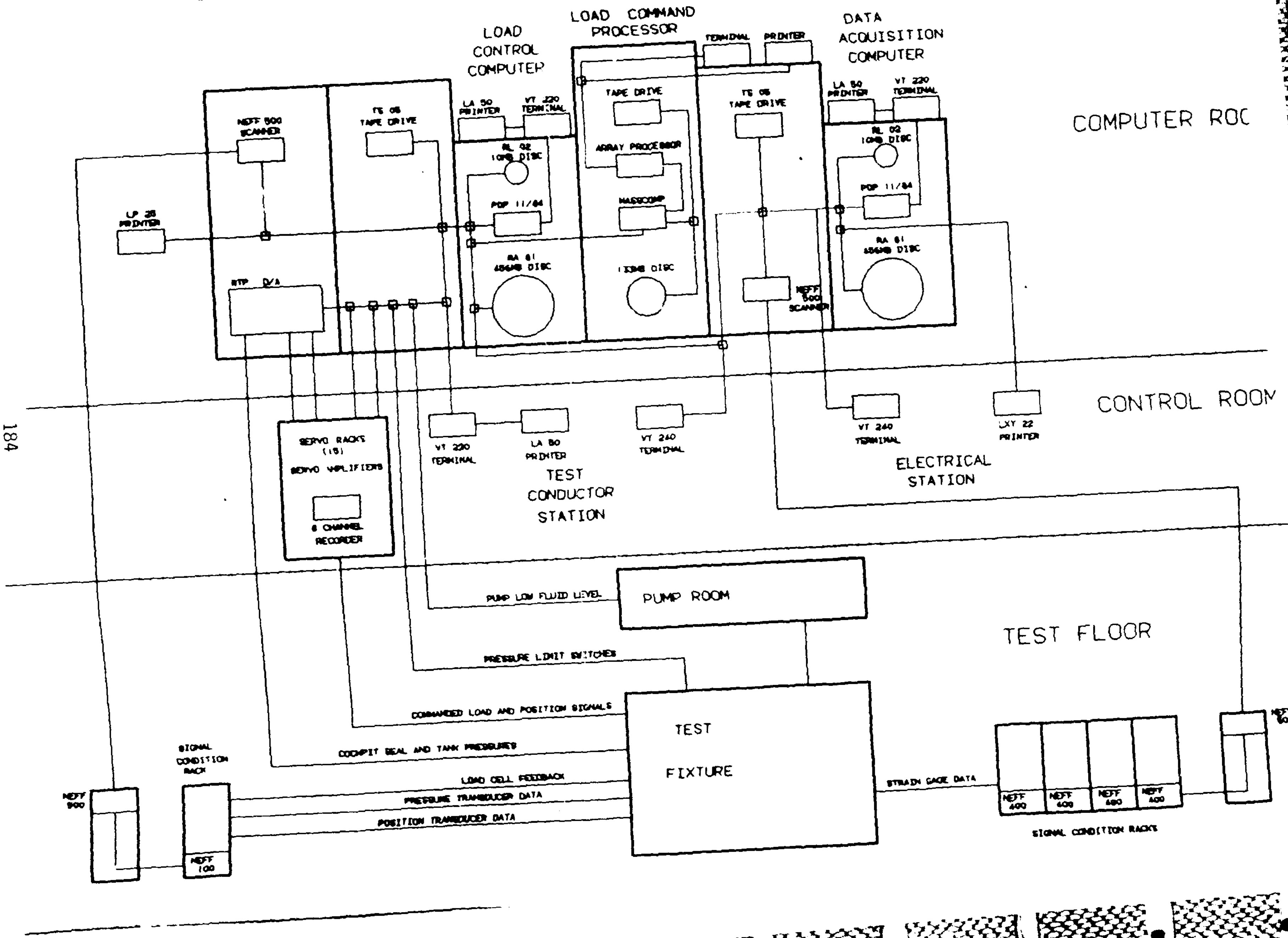
CONCLUSION

- 0 IMPROVE FLEET MANAGEMENT FOR THE F-16C/D
  - -- UPDATE STRUCTURAL ANALYSIS AND FLEET MANAGEMENT TOOLS
- 0 DEMONSTRATE ECONOMIC LIFE OF C/D AIRFRAME
- 0 IMPROVE FULL-SCALE DURABILITY TESTING TECHNIQUES
  - -- MORE REALISTIC SIMULATION - WHEN IS A TEST "GOOD ENOUGH"
- 0 DEMONSTRATE BENEFITS OF JOINT SERVICE PARTICIPATION
  - -- MORE TEST FOR THE MONEY
  - -- A LEARNING EXPERIENCE

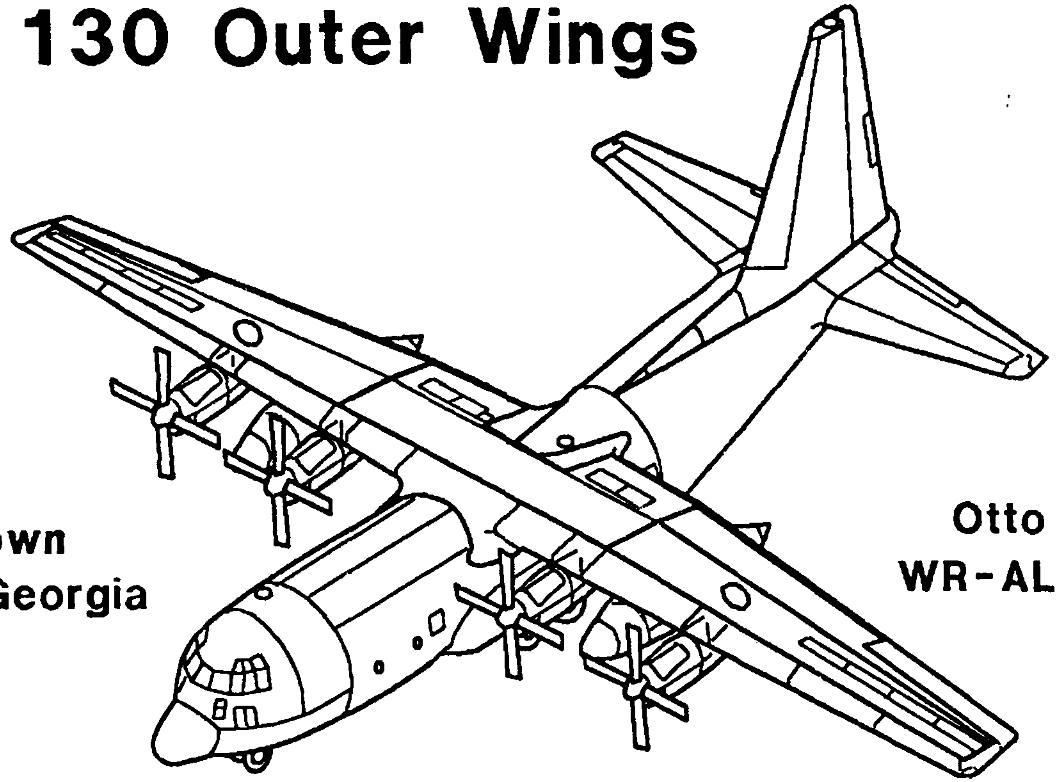


Load Cell 'A' always perpendicular to flap chord plane
Load Cell 'B' always parallel to flap chord plane
Error of 'A' drives ram 'V'
Error of 'B' drives ram 'H'
'B' is always commanded to Ø

DURABILITY TEST HARDWARE
LOCATION AND INTERCONNECTION DIAGRAM



Residual Strength Testing of C-130 Outer Wings



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WR-ALC/MMSRD

GA-7363-1

C-130 Outer Wing Residual Strength Test

Overview

- Background
- Test Program
- Test Results
- Aircraft Management
- Summary

GA-7363-2

Background

- All USAF C-130B/E series outer wings are being replaced. These wings were manufactured twenty to thirty years ago, have been subjected to severe military service approaching 20,000 flight hours, and were constructed of 7075-T6 aluminum alloys. Some of these wings have multiple repairs resulting from corrosion, cracking, and many years of service. Protection for these aging wings was being provided by a series of flight restrictions and non-destructive inspections.

C-130 Outer Wing Residual Strength Test

Background

- C-130B/E Series Outer Wing Replacement Program
  - Some Wings Had Multiple Repairs
  - Flight Restrictions Imposed on Many Aircraft
  - Nondestructive Inspections (NDI) Required

GA-7363-3

Test Program

The objectives of this test program were to:

(1) Evaluate the NDI presently being used on these airplanes. Air Force procedures and equipment were to be used to inspect each outer wing prior to testing.

(2) Assess typical B/E series outer wings as to the length and location of cracks that exist.

(3) Experimentally determine the residual strength of these outer wings in their repaired and/or cracked configuration.

(4) Correlate the analytically determined residual strength with the experimentally determined values in order to substantiate the analytical methods in use or modify them as required.

C-130 Outer Wing Residual Strength Test

Test Program

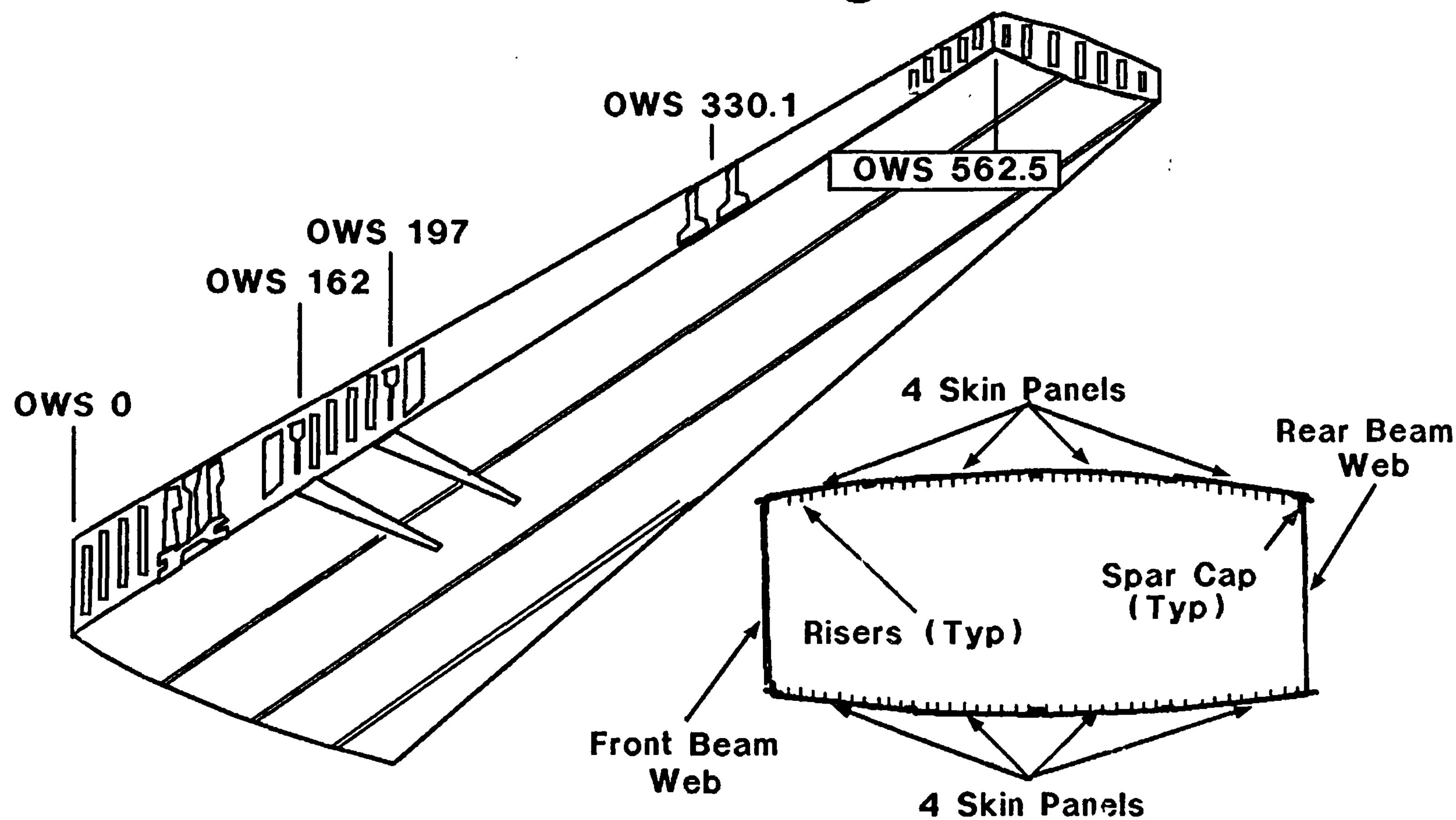
- **Objectives:**
  - Evaluate NDI
  - Assess Typical B/E Series Wings
  - Determine Residual Strength of Repaired/Cracked Wings
  - Substantiate Structural Analyses

GA-7363-4

C-130 Outer Wing Structure

- The outer wing box is a one-cell box beam.
- Front and rear beams are made of upper and lower spar caps, shear webs and web stiffeners.
- Upper and lower surfaces are integrally stiffened extruded panels which are machined to final dimensions. These panels are joined by lap splices with single rows of fasteners.
- The basic material of these early configuration outer wings which were tested is 7075-T6511 aluminum alloy.

C-130 Outer Wing Structure



GA-7363-25

Test Program

Five outer wings were tested in this program:

(1) HC-130H (AF65-0973), selected as a typical outer wing, was assigned to Kirtland AFB. Upper and lower front spar caps had been replaced, and 10,877 flight hours had been accumulated. This wing was operating with Level II restrictions.

(2) C-130E (AF61-2367), selected because no major components had been replaced, was assigned to Van Nuys, CA. It had accumulated 17,538 flight hours and was operating under Level II restrictions.

(3) C-130E (AF62-1846), selected as a "best case" outer wing, was assigned to NGB at Minneapolis, St. Paul. All surface panels and spar caps were replaced after 13,018 flight hours. This outer wing, which had accumulated a total of 15,338 flight hours, was operating unrestricted.

(4) C-130E (AF62-1859), selected as a "worst case" outer wing, was assigned to Clark AB. The lower front spar cap had been replaced, and 17,441 flight hours had been accumulated. This box had many repairs – several of them being concentrated in the dry bay lower surface. Eleven cracks were found in our pretest inspection, nine of them in the dry bay lower surface. This outer wing was operating under Level I restrictions.

(5) AC-130A (AF54-1630), selected as a typical "A" model outer wing, was assigned to Eglin AFB. All upper and lower surface wing panels were replaced after 9,045 flight hours. This outer wing accrued a total of 13,332 flight hours. Although there were two repairs in the lower surface dry bay and ten repairs in the upper surface dry bay, no significant cracks were detected in the pretest inspection. This outer wing was operating under Level II restrictions.

C-130 Outer Wing Residual Strength Test

Test Program

- Test Wings
  - HC-130H - Typical
  - C-130E - All Original Components
  - C-130E - Best Case
  - C-130E - Worst Case
  - AC-130A - Typical

GA-7363-5

Flight Restrictions

Level I

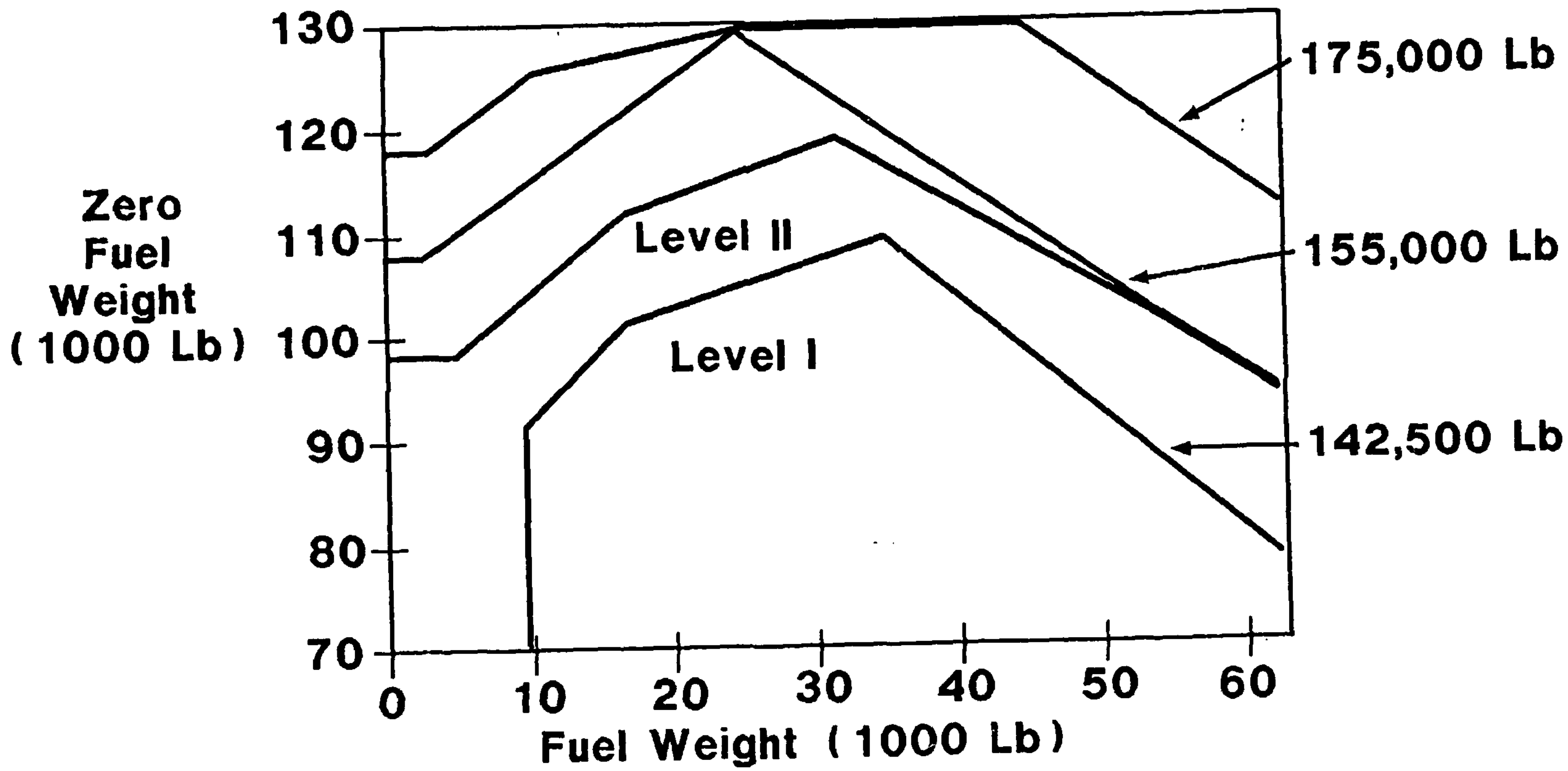
- Maximum Airspeed 225 KIAS
- 9,000 Lb Minimum Landing Fuel in Main Wing Tanks
- 107,500 Lb Maximum Zero Fuel Weight
- Max. Gross Weight of 142,500 Lb (E Models)
- Not To Exceed 0 – 2 G Load Factor in Wings Level Maneuvers
- Bank Angle Restrictions
- Avoid Turbulence
- Avoid Low Level Flight. Airspeed Limited to 190 KIAS Below 2,000 Feet
- Avoid Abrupt Control Inputs
- Special Operations Missions Are Prohibited

Level II – Same as Level I Except:

- 119,500 Lb Maximum Zero Fuel Weight
- Maximum Airspeed of 225 KIAS Does Not Apply
- Maximum Gross Weight of 155,000 Lb (E Models)

C-130 Outer Wing Residual Strength Test

Flight Restrictions



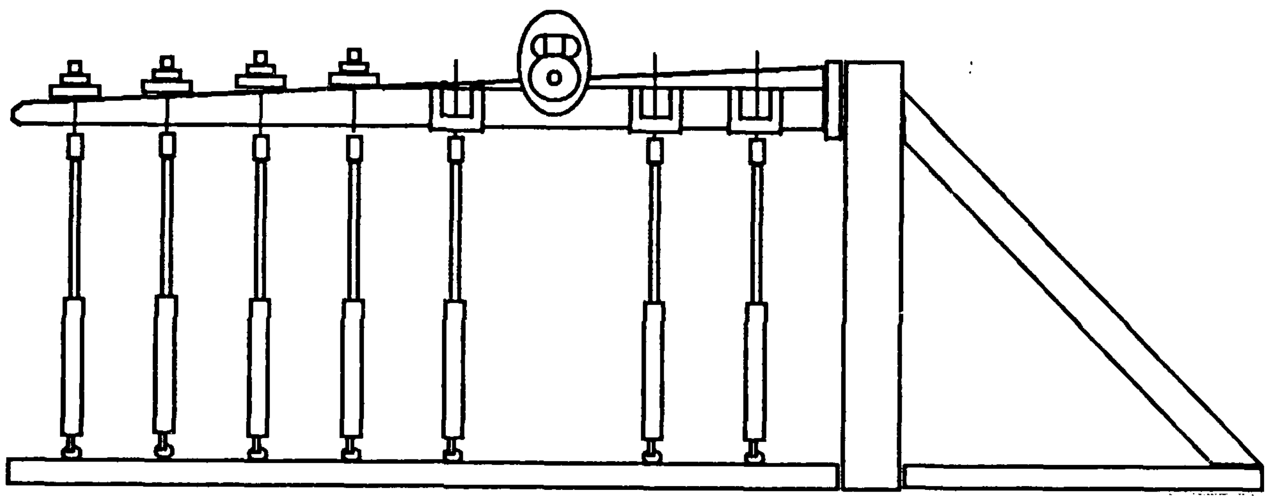
GA-7363-6

Test Set-Up

- Each outer wing was cantilevered from a test fixture with the attachment at the in-board end simulating attachment to the center wing.
- The outer wings were mounted to the test fixture upside down in order to simplify the application of upbending flight load cases.
- Each wing was hydraulically loaded to failure as deflection and strain gages were monitored and recorded.

C-130 Outer Wing Residual Strength Test

Test Set-up



GA-7363-7

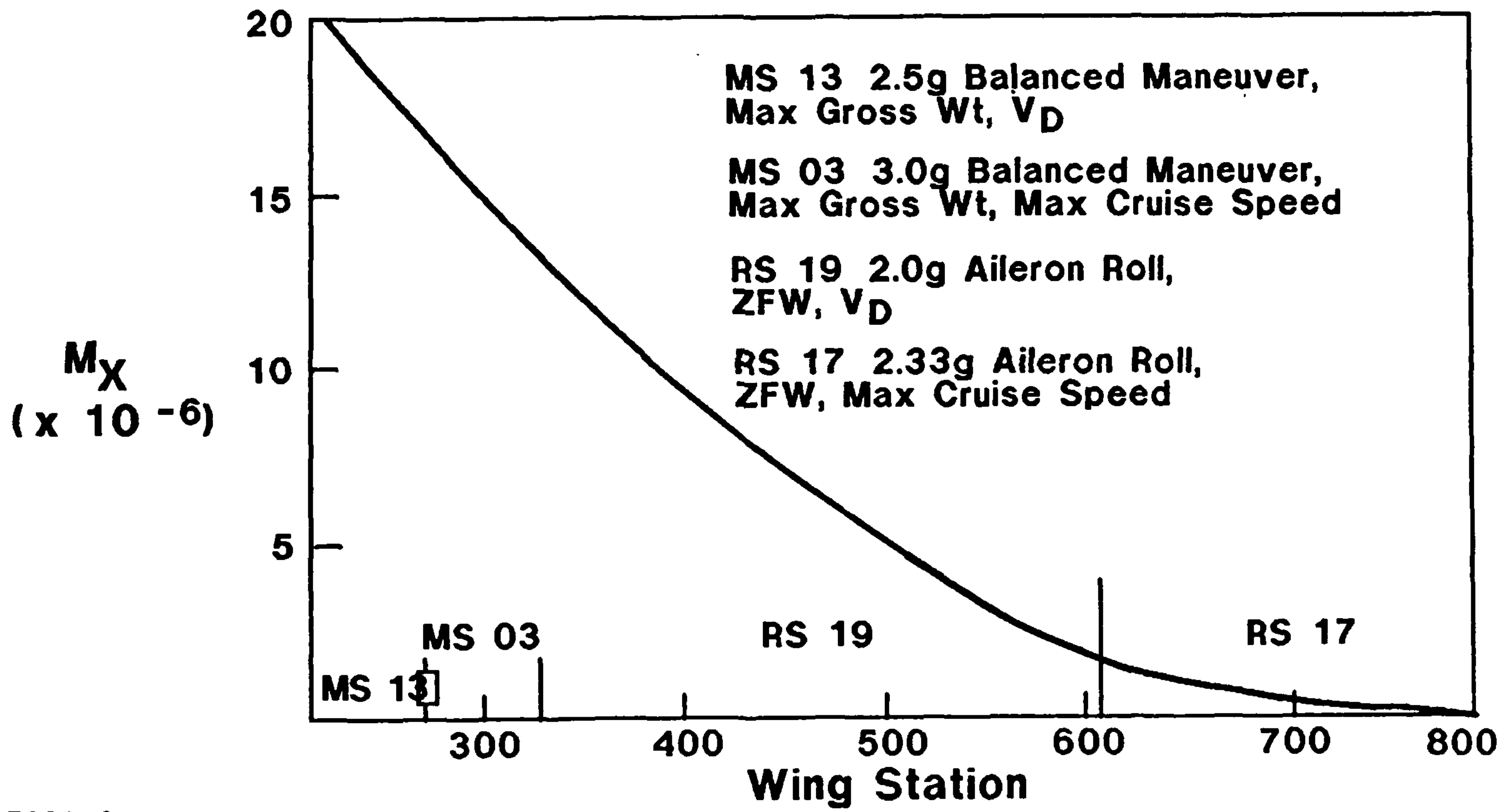
Outer Wing Maximum Upbending Case

- The test loading condition represents the envelope of maximum wing bending moments (M_x) for the entire span of the outer wing.

- Torsion (M_y) is a representation of typical torsions for these envelope conditions.

C-130 Outer Wing Residual Strength Test

Outer Wing Maximum Upbending Case (Limit Load)



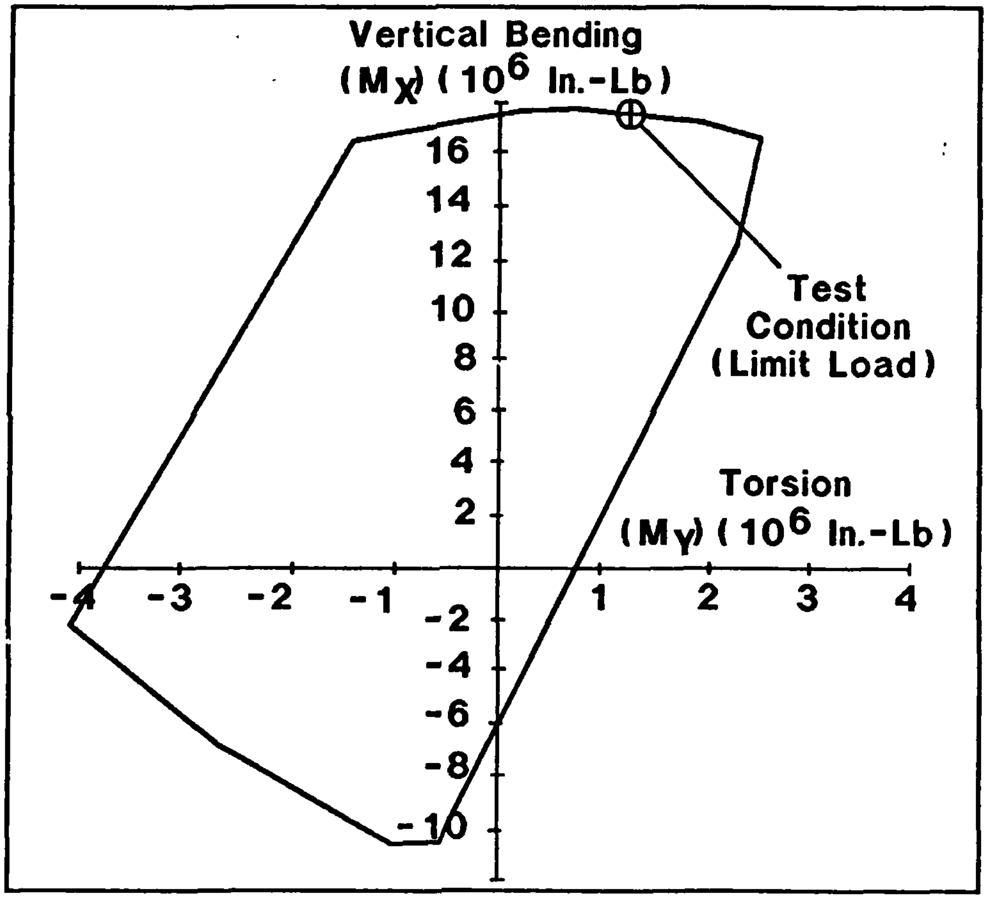
GA-7363-8

Test Loads Case – O.W.S. 34

- The test condition represents maximum vertical bending moment (M_x) and a typical torsion (M_y) condition.

C-130 Outer Wing Residual Strength Test

Test Loads Case - OWS 34



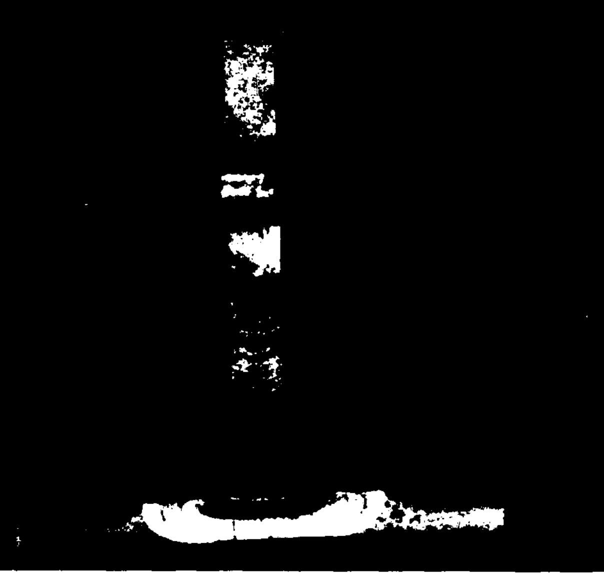
GA-7363-9

Results of Test 1

- Non-destructive inspections performed per the Air Force Technical Orders did not detect any cracks.
- The wing failed in tension through the lower surface behind the outboard engine.
- Examination of the fracture surface revealed a fatigue crack approximately 1.5 inches long. This crack originated at a fastener hole at the outboard end of a structural repair. The failure had propagated from this crack. This repair was not designed to the requirements of the C-130 Structural Repair Manual (T.O. 1C-130A-3).

C-130 Outer Wing Residual Strength Test

Test Results



- Test 1: HC-130H (Typical)
- NDI Findings
- Failing Load: 97.6 Design Limit Load
- Undetected Crack at Failure Origin

GA-7363-10

Test 1 Material Properties

- All tensile properties for the lower surface wing panels were within the specification values for 7075-T651 extrusion.

- The average fracture toughness value obtained from lower surface coupon tests (76.1 ksi $\sqrt{\text{in.}}$) is certainly acceptable; however, the 63.5 and 64.1 ksi $\sqrt{\text{in.}}$ values from panel 3 were below what was expected.

- Evaluation of the chemical composition of the material indicated that each panel was made from 7075 aluminum alloy with very low impurity levels.

C-130 Outer Wing Residual Strength Test

Test 1 Material Properties

- Tensile: Exceeded All Specification Minimums
- Fracture Toughness: $Kc_{AVE} = 76.1\ ksi\sqrt{In.}$
  $Kc_{Range} = 63.5$ to $89\ ksi\sqrt{In.}$
- Chemistry: 7075-T6

GA-7363-11

Results of Test 2

- Inspection of this wing was increased in an effort to define the condition of this wing prior to test. A few small cracks were detected, but none that should affect the test results. Several crack indications in the test area were determined to be false indications by microscopic examination after the failure.

- The wing failed in tension through the lower surface behind the outboard engine.

- Examination of the fracture surface revealed a fatigue crack at plugged holes for a screw and attaching nut plate which had been used for installing an earlier configuration engine exhaust heat shield. The failure had propagated from this crack.

C-130 Outer Wing Residual Strength Test

Test Results

- Test 2: C-130E (No Parts Replaced)
- NDI Findings
- Failing Load: 126% Design Limit Load
- Undetected Crack at Failure Origin

GA-7363-12

Test 2 Material Properties

- All tensile properties for the lower surface wing panels were within the specification values for 7075-T651 extrusion.

- The average fracture toughness value obtained from lower surface coupon tests (52.6 ksi $\sqrt{\text{in.}}$) was much lower than expected as was the entire range of values.

- Evaluation of the chemical composition of the material indicated that each panel was made from 7075 aluminum alloy with very low impurity levels.

C-130 Outer Wing Residual Strength Test

Test 2 Material Properties

- Tensile: Exceeded All Specification Minimums
- Fracture Toughness: $Kc_{AVE} = 52.6\ ksi\sqrt{In.}$
  $Kc_{Range} = 48\ to\ 62\ ksi\sqrt{In.}$
- Chemistry: 7075-T6

GA-7363-13

Results of Test 3

- This outer wing box was selected for test as a "best case." All surface panels and spar caps had been replaced, and only 2320 flight hours had been accumulated since the replacement.

- There were no repairs on the surface panels and no cracks were found by the inspections.

- The wing failed at approximately 170% of design limit load. A compression failure of the upper surface occurred at O.W.S. 236.

C-130 Outer Wing Residual Strength Test

Test Results

- Test 3: C-130E (Best Case)
- NDI Findings
- Failing Load: 170% Design Limit Load
- Compression Failure of Upper Surface

GA-7363-14

Test 3 Material Properties

- All tensile properties for the lower surface wing panels were within the specification values for 7075-T651 extrusion.

- The average fracture toughness value obtained from lower surface coupon tests (69.9 ksi $\sqrt{\text{in.}}$) was essentially as expected. The range of fracture toughness values was also about what was expected.

- Evaluation of the chemical composition of the material indicated that each panel was made from 7075 aluminum alloy with very low impurity levels.

C-130 Outer Wing Residual Strength Test

Test 3 Material Properties

- Tensile: Exceeded All Specification Minimums
- Fracture Toughness: $Kc_{AVE} = 69.9$ ksi$\sqrt{\text{In.}}$
  $Kc_{Range} = 63$ to 78 ksi$\sqrt{\text{In.}}$
- Chemistry: 7075-T6

GA-7363-15

Results of Test 4

- This outer wing box was selected for test as a "worst case." Only the lower front spar cap had been replaced in 17,441 flight hours. Multiple repairs existed.
- Eleven cracks were found in the pretest inspection, nine of them in the dry bay lower surface.
- The wing failed in tension through the lower surface behind the outboard engine.
- Examination of the fracture surface revealed that the fracture progressed through two known *cracks and several other small undetected* cracks.

C-130 Outer Wing Residual Strength Test

Test Results

- Test 4: C-130E (Worst Case)
- NDI Findings
- Failing Load: 103.1% Design Limit Load
- Tension Failure Through Known Cracks

Test 4 Material Properties

- All tensile properties for the lower surface wing panels were within the specification values for 7075-T651 extrusion.

- The average fracture toughness value obtained from lower surface coupon tests (68.1 ksi $\sqrt{\text{in.}}$) was slightly lower than the average value expected. The range of fracture toughness values was greater than expected.

- Evaluation of the chemical composition of the material indicated that each panel was made from 7075 aluminum alloy with very low impurity levels.

C-130 Outer Wing Residual Strength Test

Test 4 Material Properties

- Tensile: Exceeded All Specification Minimums
- Fracture Toughness: $Kc_{AVE} = 68.1$ ksi-$\sqrt{\text{In.}}$
  $Kc_{Range} = 54$ to 79 ksi-$\sqrt{\text{In.}}$
- Chemistry: 7075-T6

GA-7363-17

Results of Test 5

- This outer wing box was selected as a typical example of a C-130A outer wing. It contained two repairs on the dry bay lower surface and ten repairs on the dry bay upper surface.

- No significant cracks were detected in the pretest inspection.

- The wing failed in compression of the upper surface at O.W.S. 50.

C-130 Outer Wing Residual Strength Test

Test Results

- Test 5: AC-130A (Typical)
- NDI Findings
- Failing Load: 128% Design Limit Load
- Compression Failure of Upper Surface

GA-7363-18

Test 5 Material Properties

- All tensile properties for the lower surface wing panels were within the specification values for 7075-T651 extrusion.
- The average fracture toughness value obtained from lower surface coupon tests (65.0 ksi $\sqrt{\text{in.}}$) was lower than the average value expected, and the range of fracture toughness values was greater than expected.
- Evaluation of the chemical composition of the material indicated that each panel was made from 7075 aluminum alloy with very low impurity levels.

C-130 Outer Wing Residual Strength Test

Test 5 Material Properties

- Tensile: Exceeded All Specification Minimums
- Fracture Toughness: $Kc_{AVE} = 65.0\ ksi\sqrt{In.}$
  $Kc_{Range} = 56$ to $80\ ksi\sqrt{In.}$
- Chemistry: 7075-T6

GA-7363-19

C-130 Wing Static Tests

- This pictorial summary of the outer wing residual strength tests demonstrates the large variability that can be expected from these outer wings. This is due in part to:

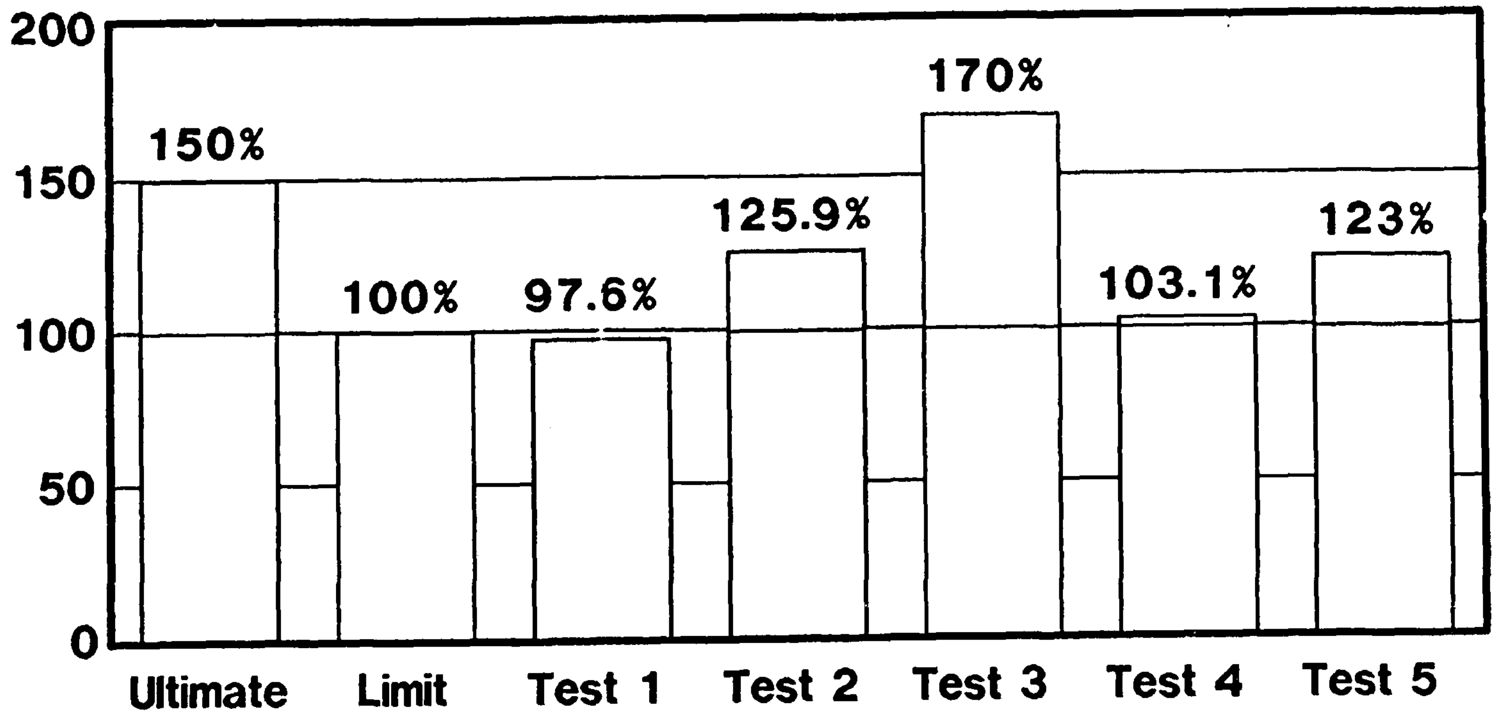
  Variations in the size, location, and design of structural repairs

  Variations in the fracture toughness of basic extrusion material from which the wing planks were manufactured

  Variations in the amount and severity of the utilization of individual airplanes

C-130 Outer Wing Residual Strength Test

C-130 Wing Static Tests



GA-7363-20

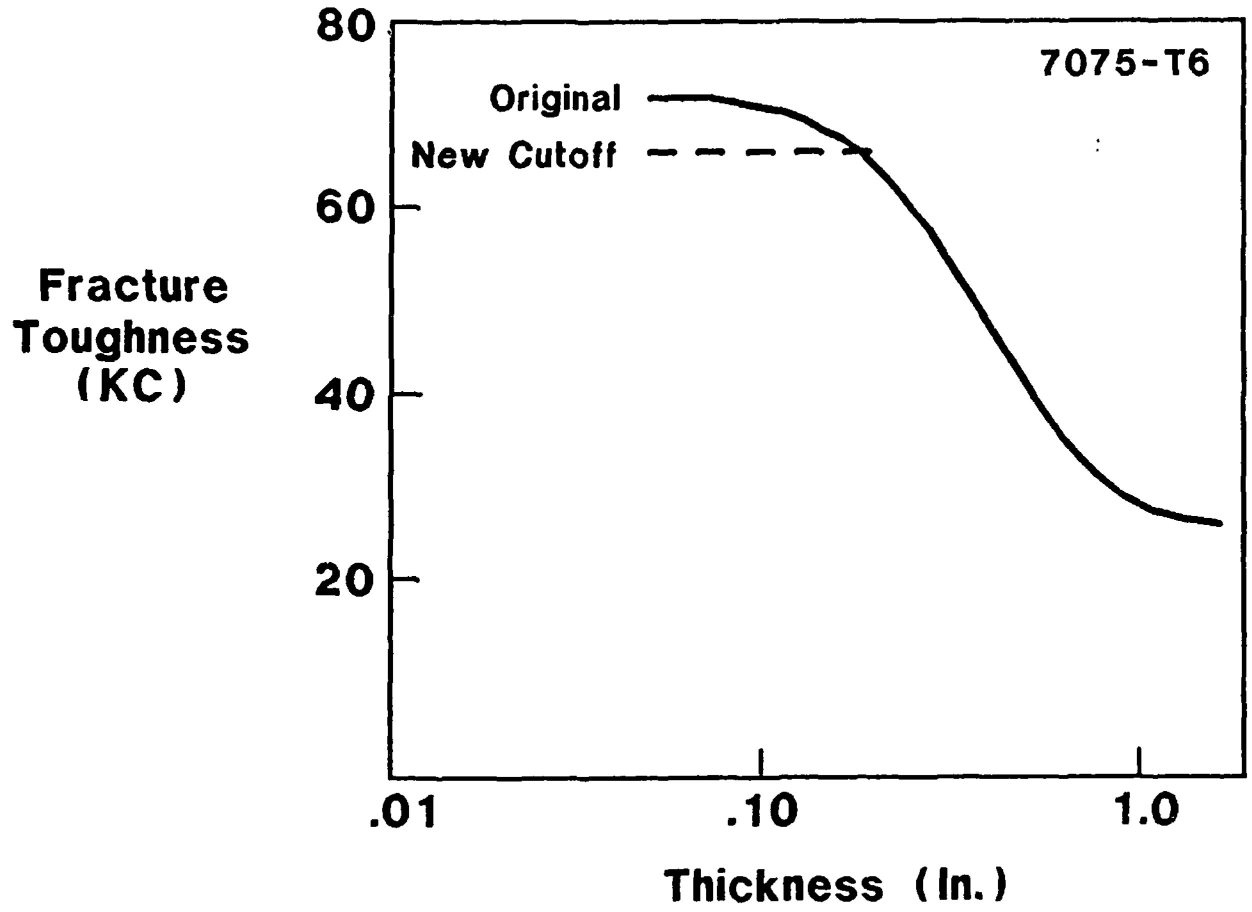
Fracture Toughness Evaluation

- Two coupons were taken from each outer wing lower surface panel of each of the five test wings and tested for fracture toughness.

- An average value of 66 ksi $\sqrt{in.}$ was used for all subsequent analyses. A value of 70 ksi $\sqrt{in.}$ was used previously, which represented all forms of 7075-T6 aluminum alloy.

C-130 Outer Wing Residual Strength Test

Fracture Toughness Evaluation



GA-7363-21

Test Conclusions

- The possibility of missing cracks using the existing NDI procedures was demonstrated by the first two tests. Each of these wings contained cracks at the inboard or outboard ends of repairs which should have been detected by the inspection program that was conducted. In both tests, the wing failure originated at these existing cracks.

- The tension failures, tests 1, 2, and 4, originated at cracks initiating at the ends of repairs that did not conform to T.O. 1C-130A-3. Conforming repairs should exhibit better fatigue life and be more inspectable. Very few conforming repairs were found on the test wings.

- The fracture toughness variations between wing panels were greater than expected. Additional testing to define this variation for other forms and alloys should be considered.

C-130 Outer Wing Residual Strength Test

Test Results

- Critical Cracks Missed on Two Tests
- Repairs Deviate From T.O. 1C-130A-3
- Fracture Toughness Values Lower Than Published Handbook Average

GA-7363-22

Structural Management

- The following restrictions have been imposed:

  Repaired wings – Level 1

  Unrepaired wings with plugged heatshield holes – Level 2

  Unrepaired wings without plugged heatshield holes – unrestricted

  Rehabbed "A" model wings with less than 4500 flight hours – unrestricted

  Rehabbed "B" model wings with less than 5000 flight hours – unrestricted

  Rehabbed "E" model wings with less than 3500 flight hours – unrestricted

- The reduced fracture toughness values have been considered in a re-evaluation of all structural inspection intervals.

- Inspections have been developed and procedures updated for evaluating the inboard and outboard ends of repairs.

- All outer wings awaiting replacement were surveyed for type and extent of repairs. The effect of existing repairs on crack growth predictions and the resulting inspections was evaluated.

C-130 Outer Wing Residual Strength Test

Structural Management

- Restrictions Were Imposed on C-130 Aircraft With Repaired Outer Wings
- Initiated Further Kc Evaluation Effort
- Established Inspections for Repairs
- Evaluated Effect of Wing Repairs

GA-7363-23

Summary

- The USAF aircraft awaiting replacement wings are operating safely with flight restrictions and an updated inspection program.

- The results of these tests are being used to enhance operations of the C-130.

  - The effects of the reduced fracture toughness values for the wing panel material are being integrated into the damage tolerance analyses. Fracture toughness of the 7075-T73 extrusions used for the wing panels on the replacement and new-manufacture outer wings is being investigated.

  - Non-destructive test procedures are being revised in order to increase the probability of detecting existing cracks, and improved non-destructive inspection methods are being investigated.

C-130 Outer Wing Residual Strength Test

Summary

- Force Aircraft Are Operating Safely
- Test Results Are Used To Enhance ASIP
  - Kc Evaluation Continues
  - NDI Investigation Continues

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A UNIQUE C5A

STRUCTURAL MODIFICATION

ASIP MEETING

AT SA-ALC

DECEMBER 1-3, 1987

PROPOSED MODIFICATIONS

- THE TROOP COMPARTMENT FLOOR IS REMOVED, THEREFORE THE MODIFIED AIRCRAFT HAS NO PASSENGER CARRYING CAPABILITY OTHER THAN THE RELIEF CREW AND COURIER COMPARTMENTS.
- THE UPPER LOBE FRAMES ARE MODIFIED TO RESIST CABIN PRESSURE LOADS.
- THE WL 314 LONGERON IS BEEFED-UP TO BEAM PRESSURE LOADS TO THE DEEP FRAMES.
- THE PRESSURE BULKHEAD AT FUS STA 2101 IS DELETED AND A NEW FIXED PRESSURE BULKHEAD IS ADDED AT FUS STA 2167.
- A NEW MOVABLE PRESSURE BULKHEAD IS ADDED AT FUS STA 2167.
- THE PRESSURE DOOR IS ROTATED AFT TO ALIGN WITH THE NEW PRESSURE BULKHEAD.
- THE TORQUE DECK AND TORQUE DECK BEAMS ARE MODIFIED TO CLEAR THE CONTAINER AS IT IS LOADED.
- THE CENTER CARGO DOOR WHICH STOWS BENEATH THE TORQUE DECK IS MODIFIED TO CLEAR THE PATH OF THE CONTAINER. THE MODIFIED AIRCRAFT WILL HAVE NO AERIAL DELIVERY CAPABILITY.
- MODIFY AFT RAMP LOCKS AND BACK-UP STRUCTURE DUE TO INCREASED PRESSURE LOADS ON THE RELOCATED RAMP PRESSURE DOOR.

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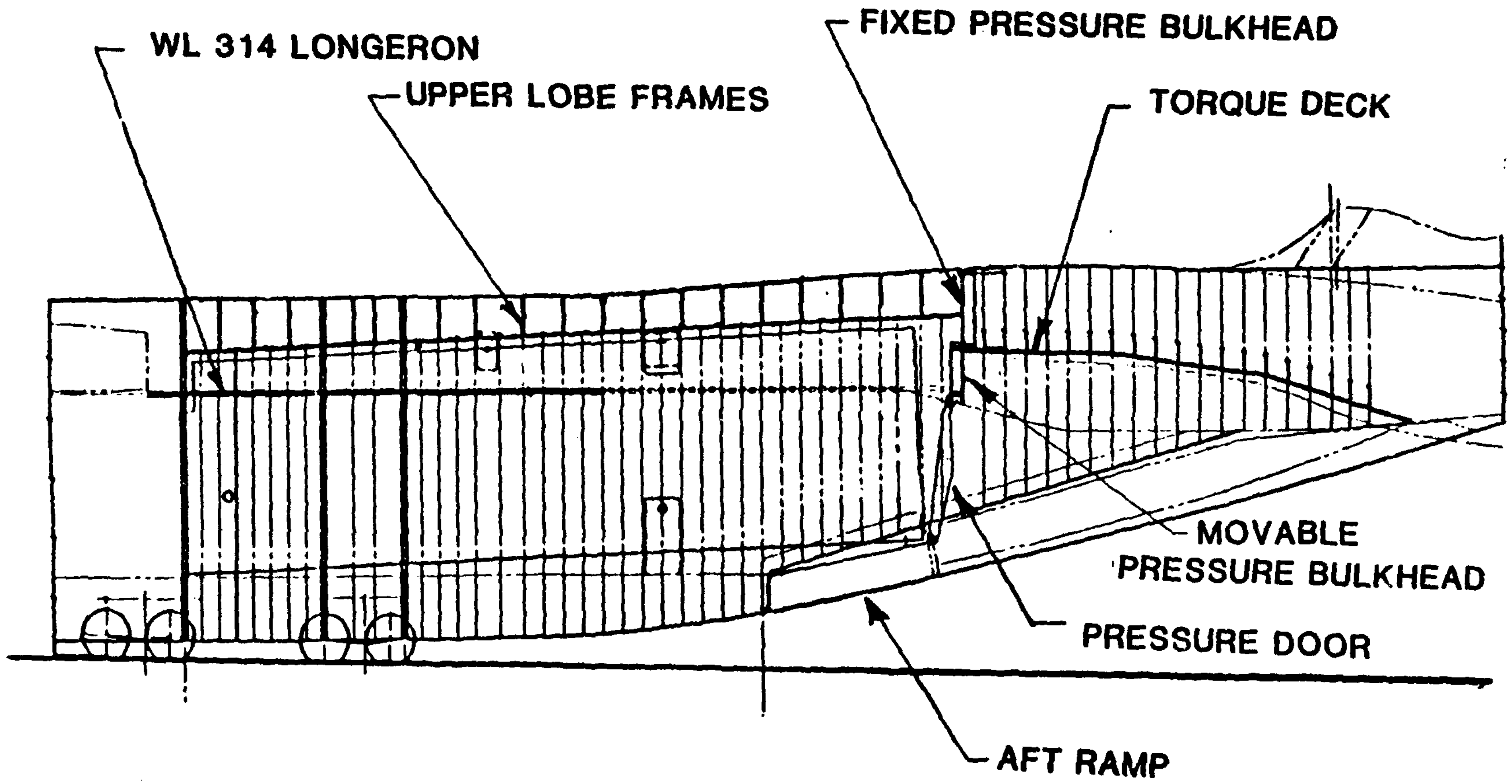
A UNIQUE C5A STRUCTURAL MODIFICATION

MODIFICATIONS



LOADING CONFIGURATION

- THIS SLIDE SHOWS GROUND TRANSPORT CONFIGURATION OF THE CONTAINER TRANSPORT SYSTEM.
- CONSISTS OF A TRACTOR, CONTAINER, AND AFT TRANSPORTER.
- THE CONTAINER ASSEMBLY IS 15 FT 8 IN HIGH, 17 FT 8 IN WIDE AND 74 FT 11 IN LONG.
- THE AFT TRANSPORTER IS ALSO AN ELEVATING UNIT AND HAS DRIVEN WHEELS WHICH ARE STEERABLE.

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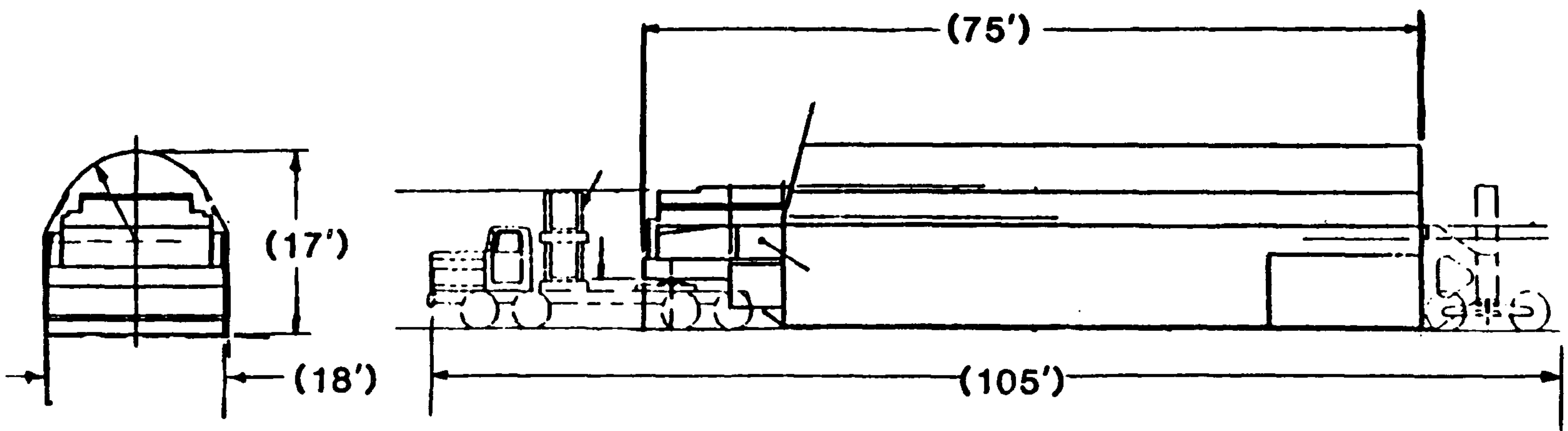
CONTAINER CONFIGURATION



CONTAINER LOADING

- THE AIRCRAFT IS IN THE FORWARD KNEEL POSITION.
- AFT RAMP IS SUPPORTED.
- THE CONTAINER IS PULLED INTO THE AIRCRAFT BY A WINCH ATTACHED TO THE TRACTOR FRONT BUMPER.
- THERE IS A NOMINAL 6 INCH CLEARANCE ALONG THE UPPER CENTERLINE OF THE AIRCRAFT DURING LOADING.
- A 2.5 INCH CLEARANCE IS MAINTAINED AT THE RAMP CREST DURING LOADING;.
- THE FORWARD AND AFT ELEVATING DEVICES ALTERNATIVELY RAISE AND LOWER THE FORWARD AND AFT ENDS OF THE CONTAINER TO MAINTAIN THESE CLEARANCES.

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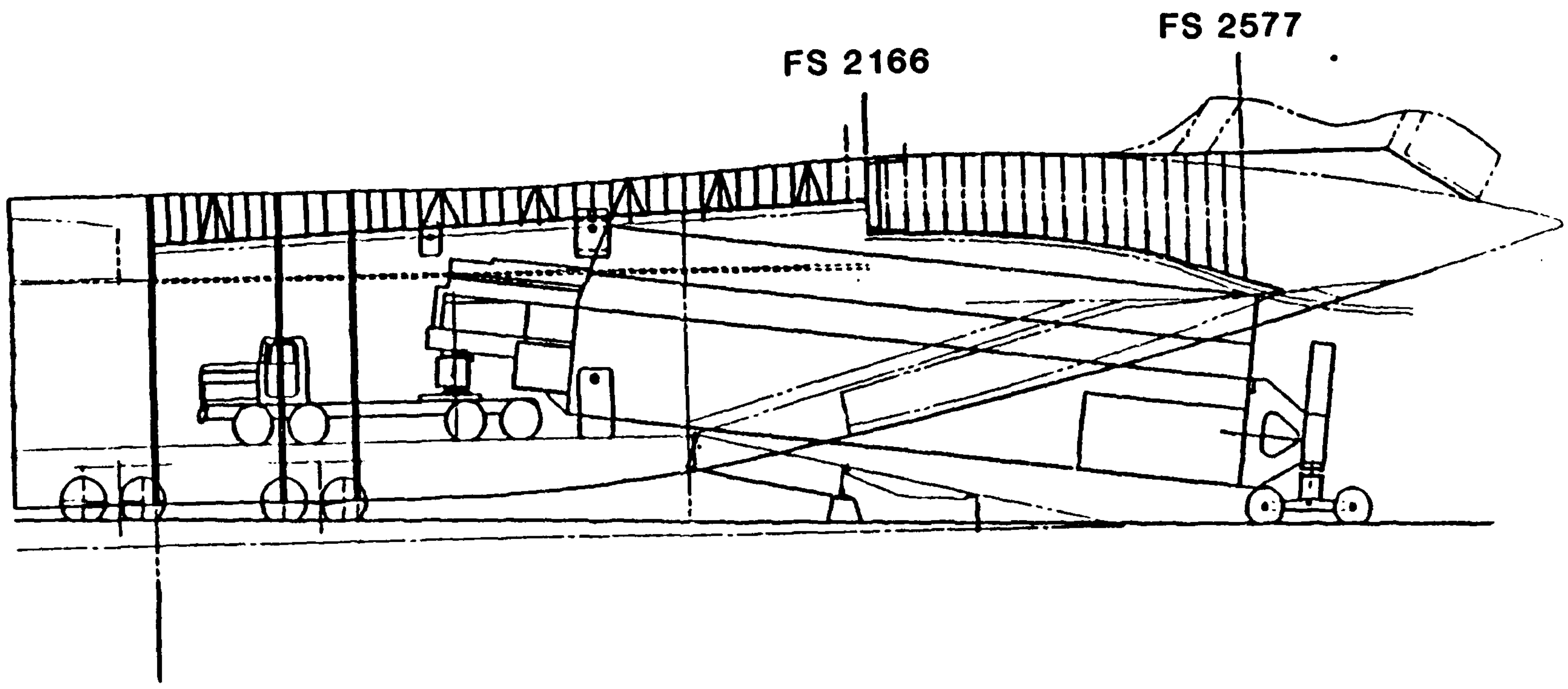
A UNIQUE C5A STRUCTURAL MODIFICATION

CONTAINER LOADING



C5 FINITE ELEMENT MODEL

TO SUPPORT THE MODIFICATION PROGRAM, THE C-5A NASTRAN MODEL WAS MODIFIED TO REFLECT THE MODIFIED AIRFRAME DESIGN. THIS REQUIRED EXTENSIVE REWORK OF THE BASELINE MODEL FROM FS 1383 TO FS 2617. THIS SLIDE IS A CUTAWAY VIEW OF THE AFT FUSELAGE OF THE MODEL SHOWING THE STRUCTURAL CHANGES WHICH INCLUDE REMOVAL OF THE TROOP DECK AND CONSTRUCTION OF THE DEEP FRAMES, BUILD-UP OF THE WL 314 LONGERON, RELOCATION OF THE FIXED PRESSURE BULKHEAD AND AFT PRESSURE DOOR, ADDITION OF THE MOVEABLE PRESSURE BULKHEAD, AND MODIFICATION OF THE TORQUE DECK.

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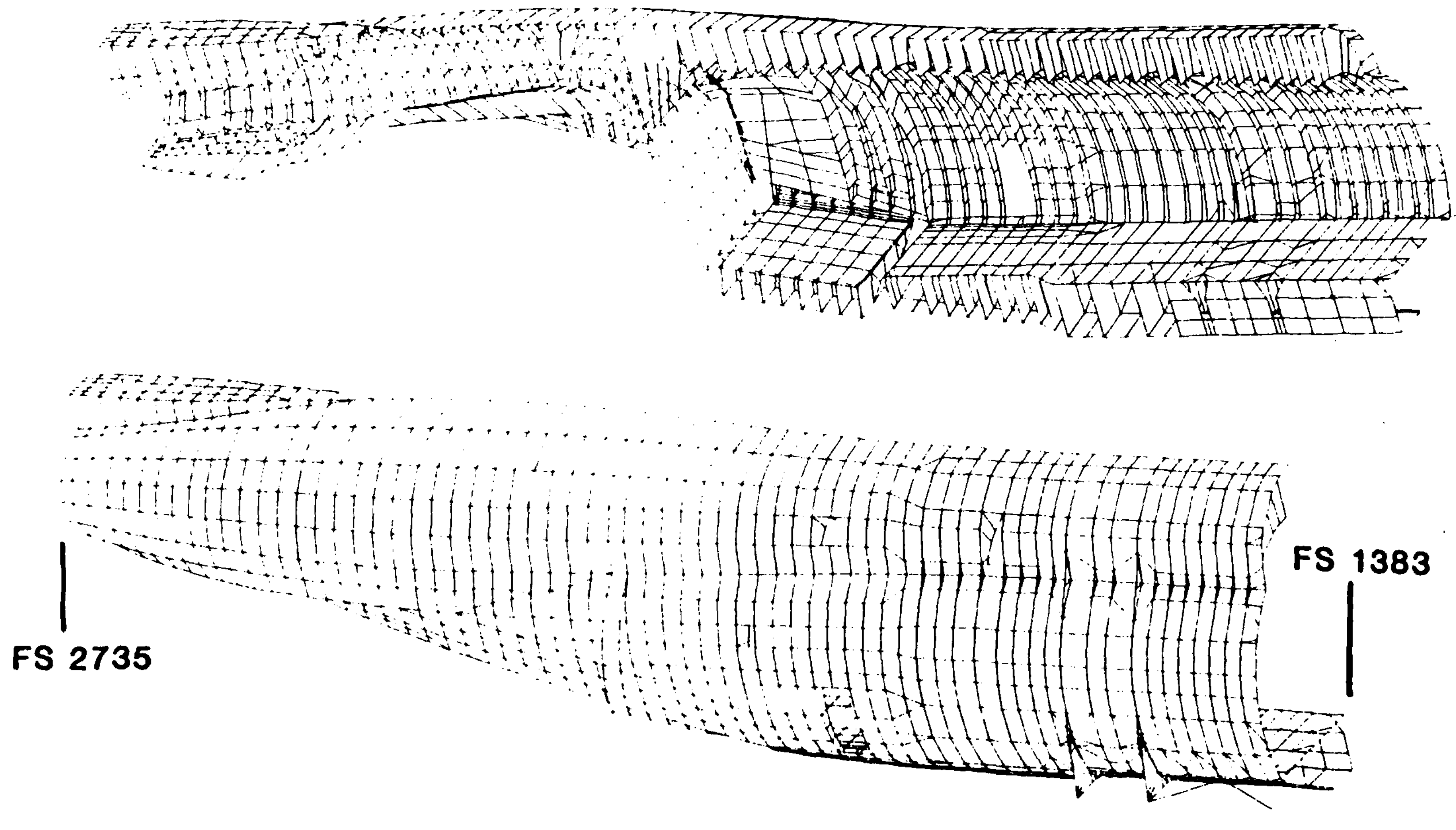
A UNIQUE C5A STRUCTURAL MODIFICATION

NASTRAN MODEL



DETAILED FINITE ELEMENT MODELS

IN ADDITION TO THE MODIFIED AIRFRAME MODEL, SEVERAL DETAIL FINITE ELEMENT MODELS WERE CONSTRUCTED TO CALCULATE STRESS DISTRIBUTONS AND FASTENER LOADS FOR DURABILITY AND DAMAGE TOLERANCE ANALYSIS. A TYPICAL INTERMEDIATE FRAME INSTALLATION AND DEEP FRAME INSTALLATION AT WL 314 WERE TWO OF THE AREAS MODELED IN DETAIL. A PORTION OF EACH OF THOSE MODELS IS SHOWN HERE.

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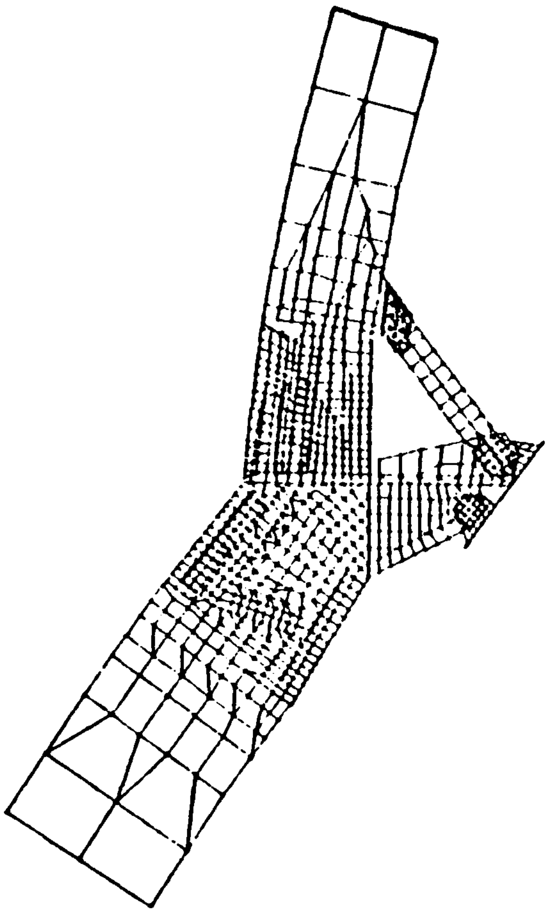
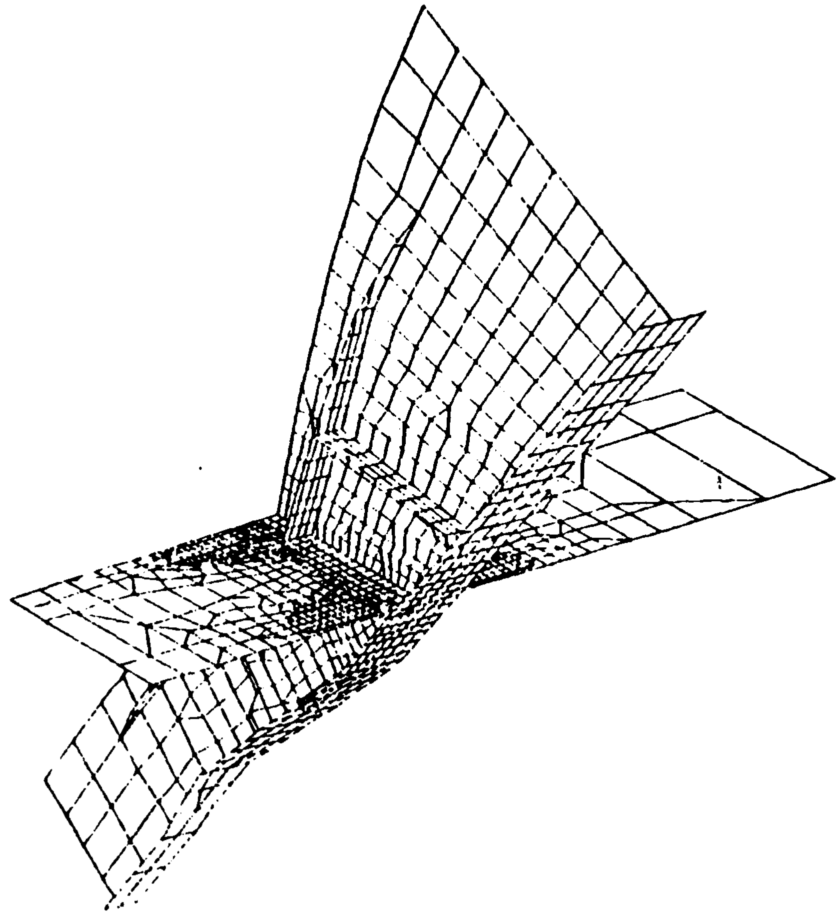
A UNIQUE C5A STRUCTURAL MODIFICATION

DETAILED FINITE ELEMENT MODELS



INTERMEDIATE FRAME



DEEP FRAME

C-5 MODIFIED AIRFRAME/CONTAINER FEM INTEGRATION

A FINITE ELEMENT ANALYSIS WAS USED TO ANSWER TWO BASIC QUESTIONS ABOUT THE MODIFICATION PROJECT. FIRST, IS THE TIE-DOWN ARRANGEMENT ADEQUATE CONSIDERING CARGO FLOOR DEFLECTIONS AND FLEXIBILITIES OF THE AIRFRAME, CONTAINER, AND TIE-DOWN SYSTEM? SECOND, WILL THE RELATIVE DEFLECTIONS OF THE AIRFRAME AND CONTAINER CAUSE INTERFERENCE BETWEEN THE TWO STRUCTURES?

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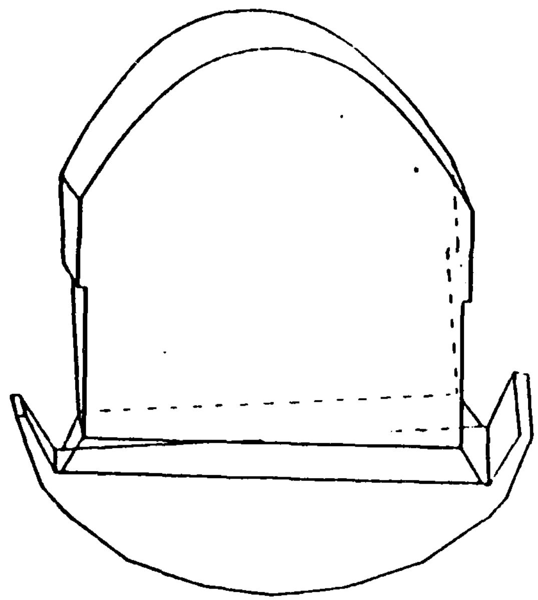
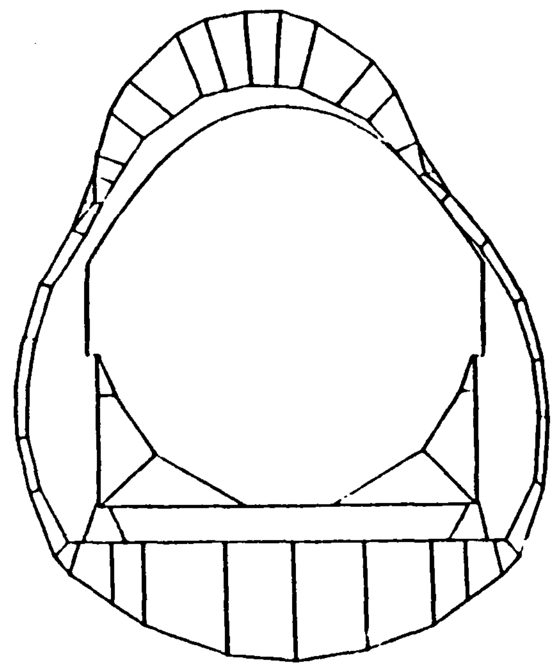
A UNIQUE C5A STRUCTURAL MODIFICATION

FLEXIBILITY STUDIES



CARGO TIE DOWN STUDIES



RELATIVE DEFLECTION STUDIES

TEST ARTICLE FINITE ELEMENT MODEL

A FINITE ELEMENT MODEL OF THE X990 TEST ARTICLE, INCLUDING THE TEST JIG AND TORQUE DECK TRANSITION AREA, WAS CONSTRUCTED TO SUPPORT THE TEST PROGRAM. THE MODEL WAS USED EARLY IN THE PROGRAM TO CALCULATE VERTICAL TAIL DEFLECTIONS NECESSARY TO DETERMINE JACK STROKE LENGTHS AND ANGULAR MOVEMENT. THE MODEL WILL BE USED TO GENERATE PREDICTED STRESSES FOR STRAIN SURVEYS AND STATIC TESTS.

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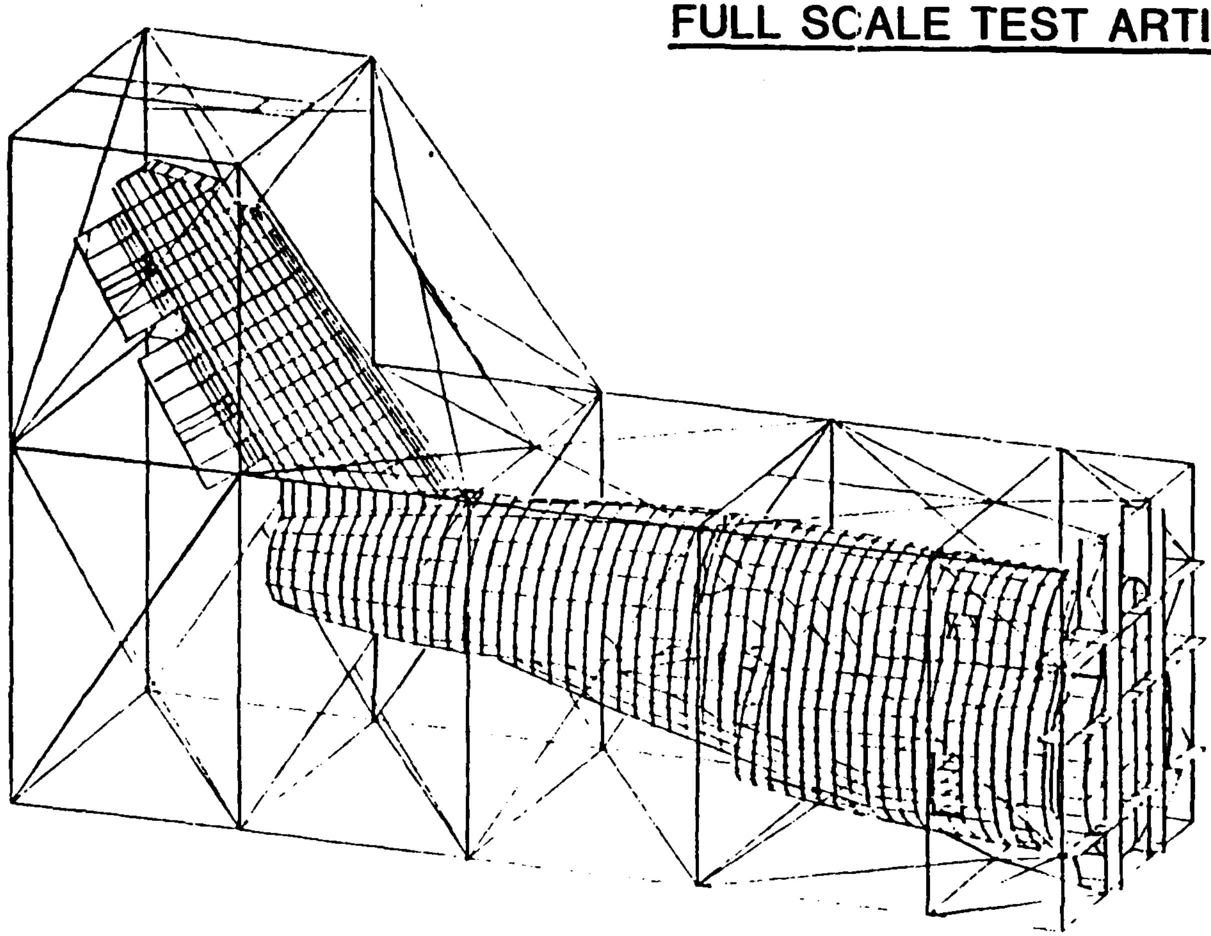
A UNIQUE C5A STRUCTURAL MODIFICATION

FULL SCALE TEST ARTICLE



LOAD REDISTRIBUTION EFFECTS ON REDESIGN

A PRINCIPAL LOAD PATH CHANGE OCCURS BETWEEN THE C5A/B AND THE C5 MODIFICATION REDESIGN. IN ORDER TO PROVIDE CLEARANCE FOR THE PAYLOAD CONTAINER THE WL 314 TROOP FLOOR AFT OF THE WING HAD TO BE DELETED.

ON THE C5 A/B THE WL 314 TROOP DECK FLOOR PROVIDES AN EFFICIENT 'TENSION TIE' ACROSS THE AIRCRAFT TO REACT THE CABIN PRESSURE HORIZONTAL FORCES AT WL 314 DUE TO THE FUSELAGE SHELL RADIUS CHANGE. ON THE MODIFICATION, THESE HORIZONTAL FORCES ARE REACTED AS A BEAM ON MULTIPLE SUPPORTS, THE BEAM BEING THE WL 314 LONGERON AND THE SUPPORTS ARE THE DEEP FRAMES ON 40" FORE-AFT SPACINGS.

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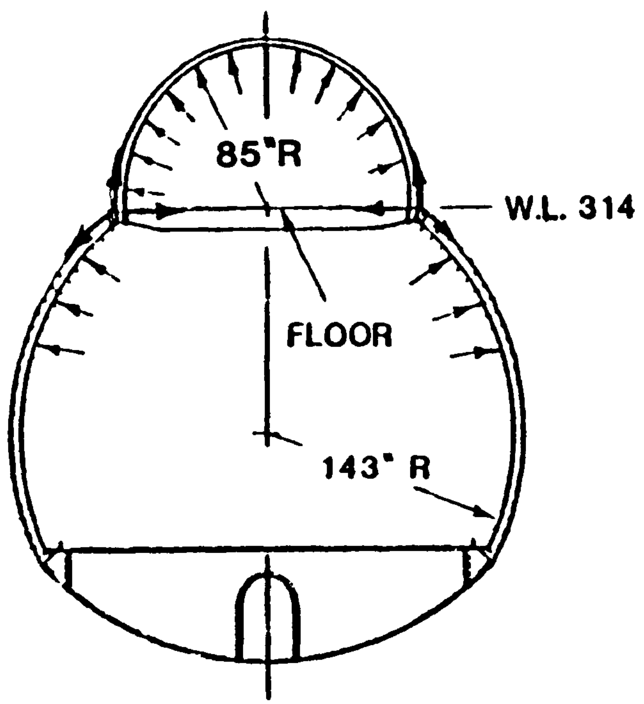
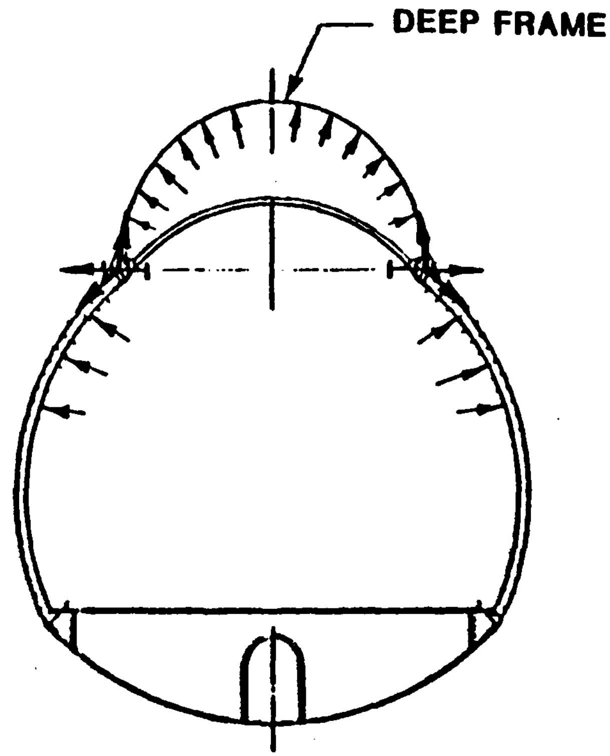
A UNIQUE C5A STRUCTURAL MODIFICATION

LOAD REDISTRIBUTION



C5-A



C-5A MOD. CONFIGURATION

WATERLINE 314.00 LONGERON

- THE DEPTH OF THE WL 314.00 LONGERON IS INCREASED IN ORDER TO BEAM THE SKIN PRESSURE KICK LOADS FORMERLY REACTED BY THE TROOP COMPARTMENT FLOOR OVER TO THE REINFORCED FRAMES.

- DUE TO THE CLOSE PROXIMITY OF THE CONTAINER TO THE AIRCRAFT CONTOURS, THE LONGERON IS LOCATED EXTERNALLY FROM FS 1347 TO FS 1824 AS SHOWN IN SECTION A-A.

- THE LONGERON IS COVERED BY THE WING FILLET FAIRING.

- THE LONGERON CONFIGURATION IS AS SHOWN IN SECTION B-B FROM FS 1824 AFT TO FS 2209.

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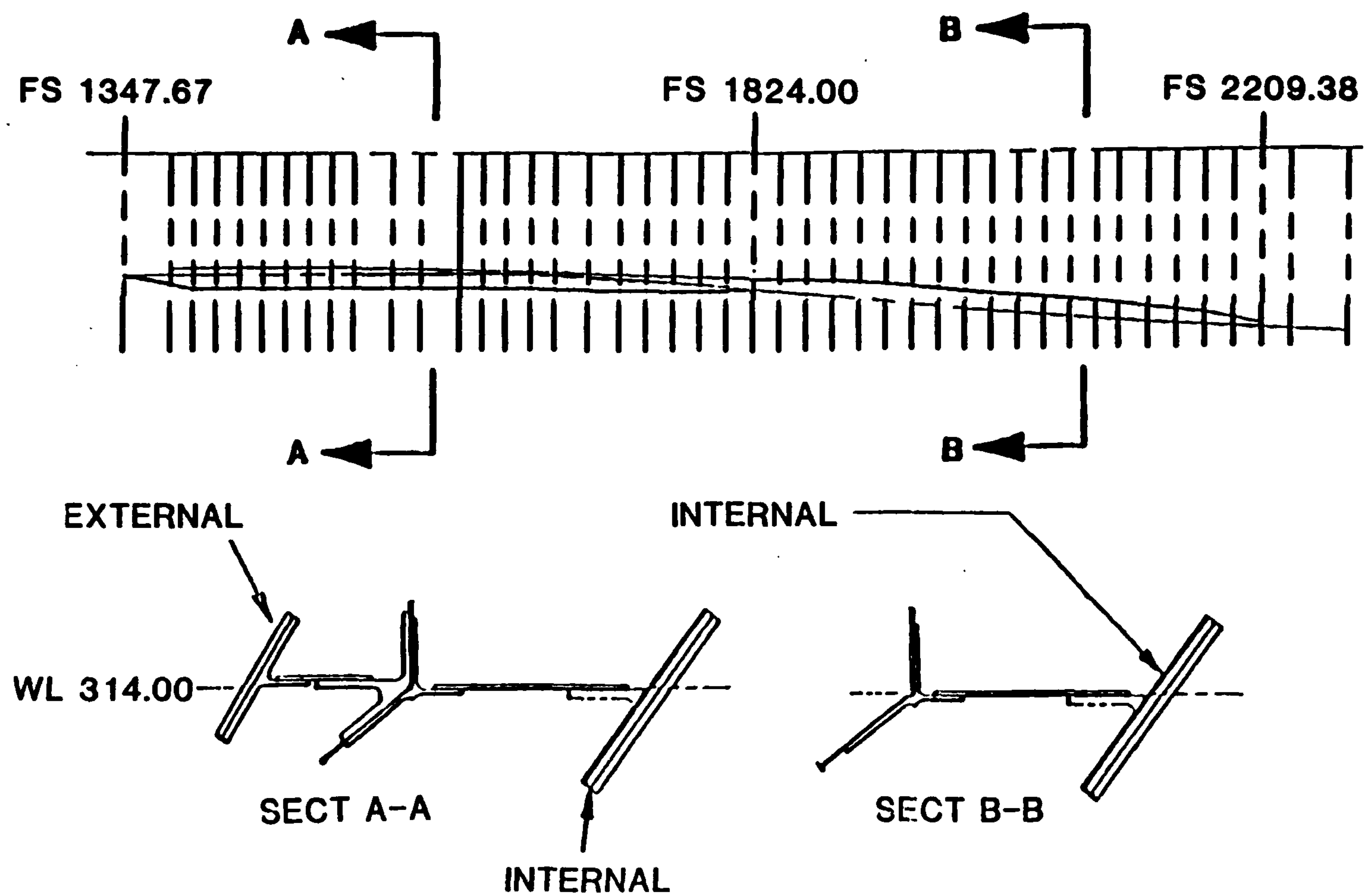
W.L. 314.00 LONGERON



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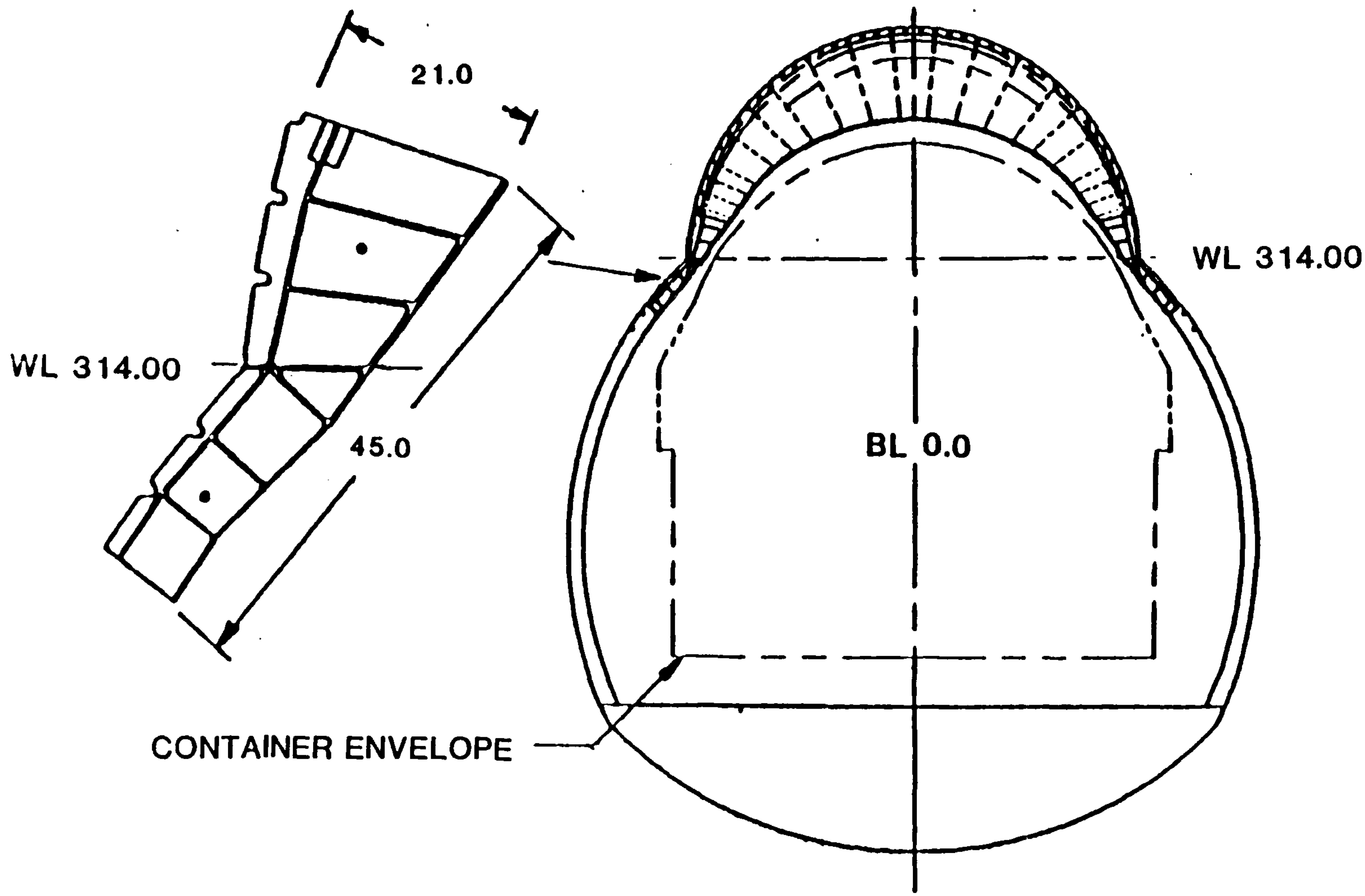
A UNIQUE C5A STRUCTURAL MODIFICATION

FRAME REINFORCEMENT



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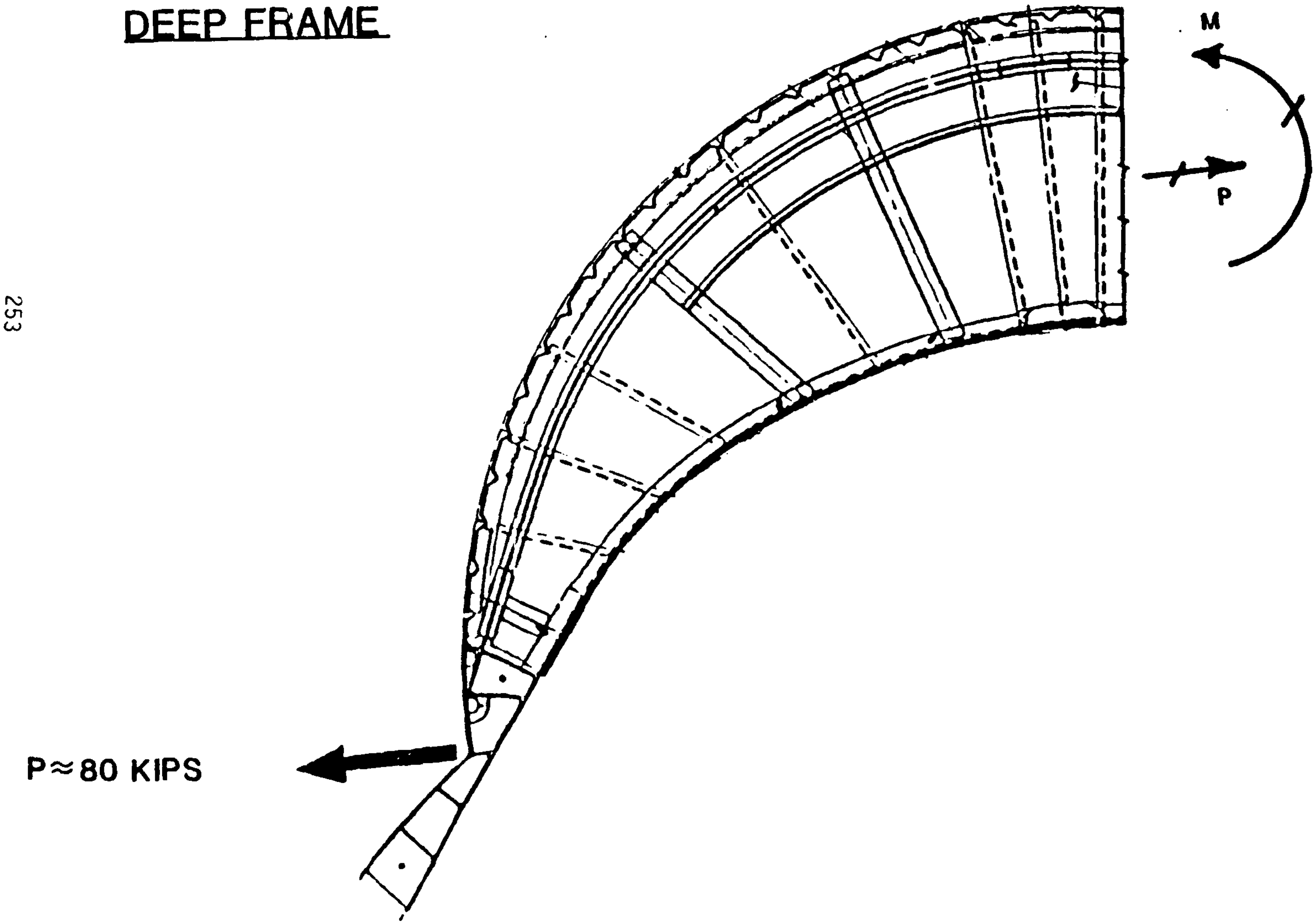
A UNIQUE C5A STRUCTURAL MODIFICATION

DEEP FRAME



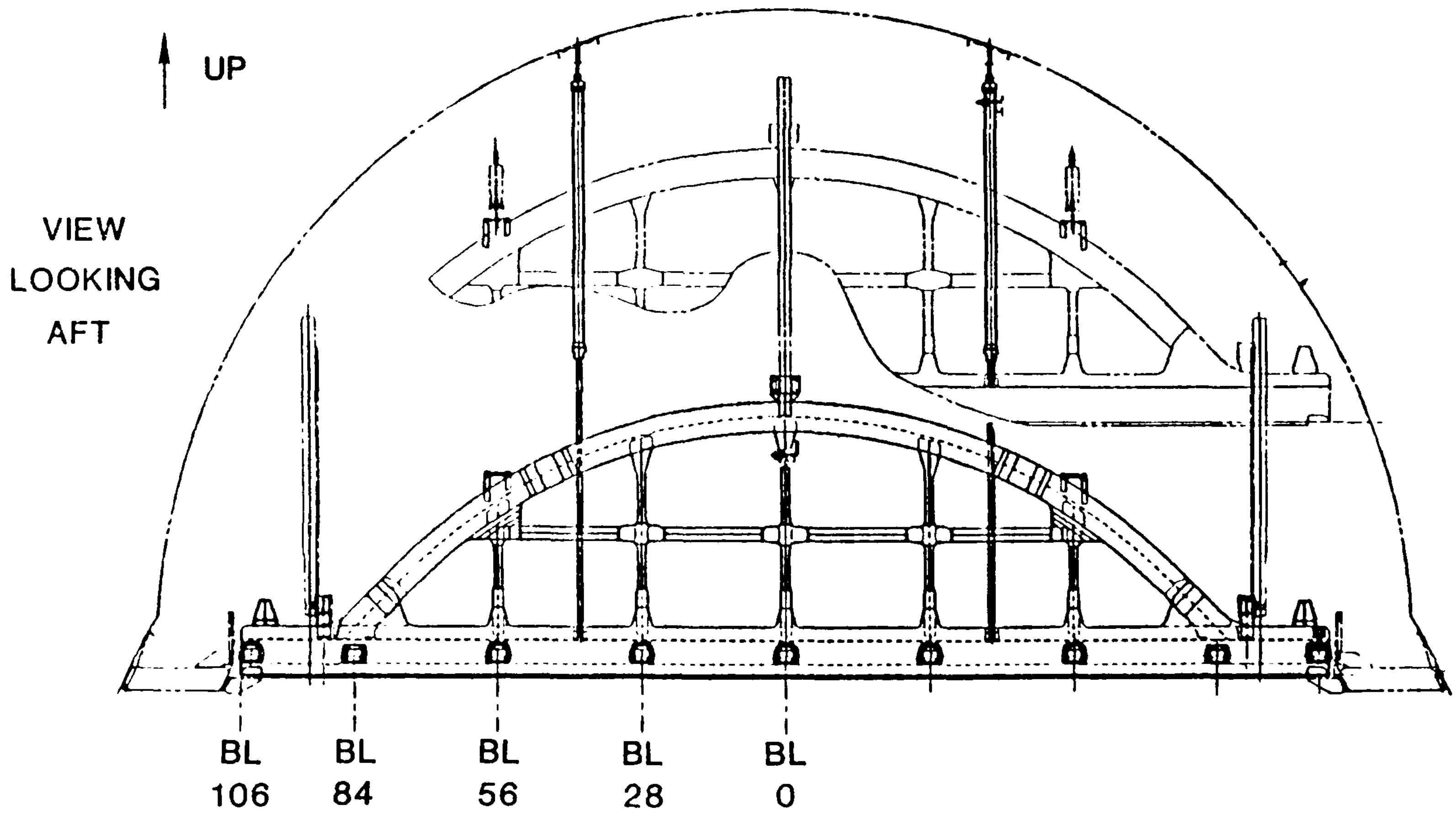
MOVABLE PRESSURE BULKHEAD

- HORIZONTAL BOX BEAM IS 17 IN. DEEP @ WL 305.6
- THE PRESSURE DOOR HAS NINE FINGERS THAT BEAR AGAINST ROLLERS ON THE PRESSURE BULKHEAD.
- THE HORIZONTAL BOX BEAM IS LOADED PRINCIPALLY BY THE NINE PRESSURE DOOR ROLLER SUPPORTS.
- THE MODIFICATION DESIGN ROLLERS HAVE BEEN LOWERED .40 IN. TO ACCOUNT FOR THE INCREASED DEFLECTION BETWEEN THE FINGER FITTINGS AND THE ROLLERS.
- CURVED I - BEAM IS 16.1 IN. DEEP AND R = 106.1 IN.
- VERTICAL I - BEAMS ARE 15.1 IN. DEEP @ BLO, 28 L&R, 56 L&R
- PRESSURE DIAGPRAGM BETWEEN THE CURVED I - BEAM AND THE HORIZONTAL BOX BEAM IS ON THE AFT SIDE OF THE BULKHEAD.

Lockheed
-Georgia Company

A UNIQUE C5A STRUCTURAL MODIFICATION

MOVABLE PRESSURE BULKHEAD



FIXED PRESSURE BULKHEAD

- THE FIXED PRESSURE BULKHEAD IS CRITICAL FOR 17.4 PSI INTERNAL PRESSURE. THE REACTION POINTS FROM THE MOVABLE PRESSURE BULKHEAD ARE THE DARKENED AREAS.

- THE FIXED PRESSURE BULKHEAD ALSO PROVIDES SHEAR REDISTRIBUTION FOR THE TORQUE DECK.

- THE PRESSURE RELIEF DOORS ARE RELOCATED BUT UNCHANGED.

- THE FRONTAL AREA IS 18,604 SQ. IN.
  THE BURST PRESSURE LOADS P_b IS:

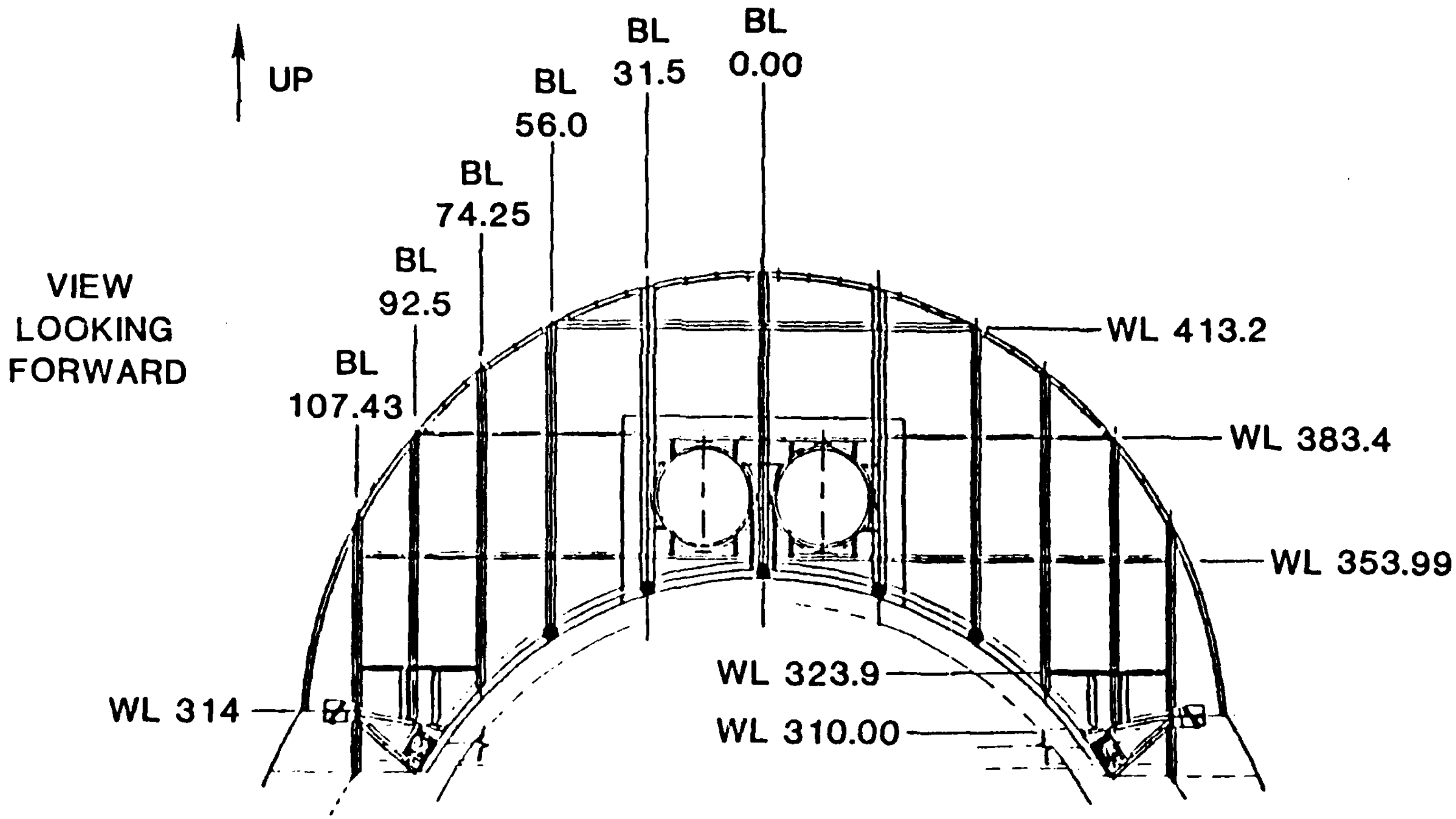
  $$P_b = 18604(17.4) = 323710 \text{ LBS. (@ 17.4 PSI PRESSURE)}$$

- VERTICAL I - BEAMS ARE LOCATED AT BL 0, AND AT L&R BL 31.5, 56.0, 74.25, 92.5, 107.25 AND ARE 12.0 IN. DEEP.

Lockheed -Georgia Company

A UNIQUE C5A STRUCTURAL MODIFICATION

FIXED PRESSURE BULKHEAD



TORQUE DECK MODIFICATION

- THE TORQUE DECK AFT OF THE PRESSURE BULKHEAD, WHICH FORMERLY WAS A FLAT HONEYCOMB PANEL SUPPORTED BY STRAIGHT BEAMS, IS MODIFIED TO PROVIDE CLEARANCE FOR THE CONTAINER. DURING LOADING OPERATIONS.

- A NEW CURVED BEAM IS INSTALLED WHICH SPLICES INTO STUBS OF THE EXISTING STRAIGHT BEAMS.

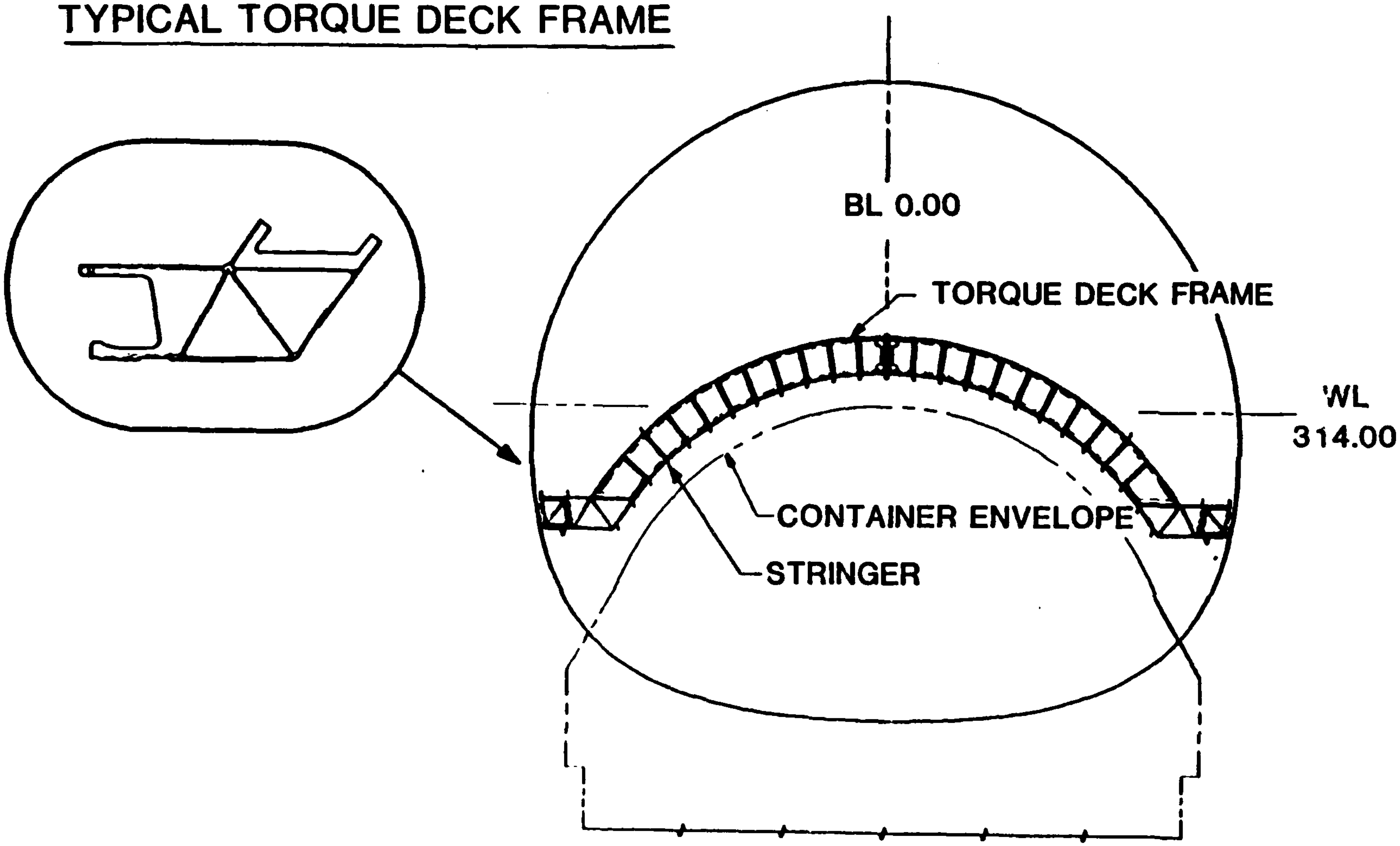
- A MINIMUM CLEARANCE OF 6 INCHES IS MAINTAINED.

- EXTRUDED FORE AND AFT STRINGERS AND A SHEET METAL DECK IS INSTALLED TO THE LOWER CAPS OF THE CURVED BEAMS.

Lockheed -Georgia Company

A UNIQUE C5A STRUCTURAL MODIFICATION

TYPICAL TORQUE DECK FRAME



CARGO DOOR SWING ARMS

- THE SWING ARM PROGRAMS THE CENTER CARGO DOOR HALF TO SCISSOR PARTIALLY CLOSE DURING OPENING OF THE SIDE CARGO DOOR SO THAT THE INBOARD FORWARD CORNER WILL BLEAR THE GROUND WHEN THE AIRCRAFT IS IN THE AFT KNEEL POSITION.

- THERE ARE TWO SWING ARMS. THE FORWARD ONE IS SHOWN.

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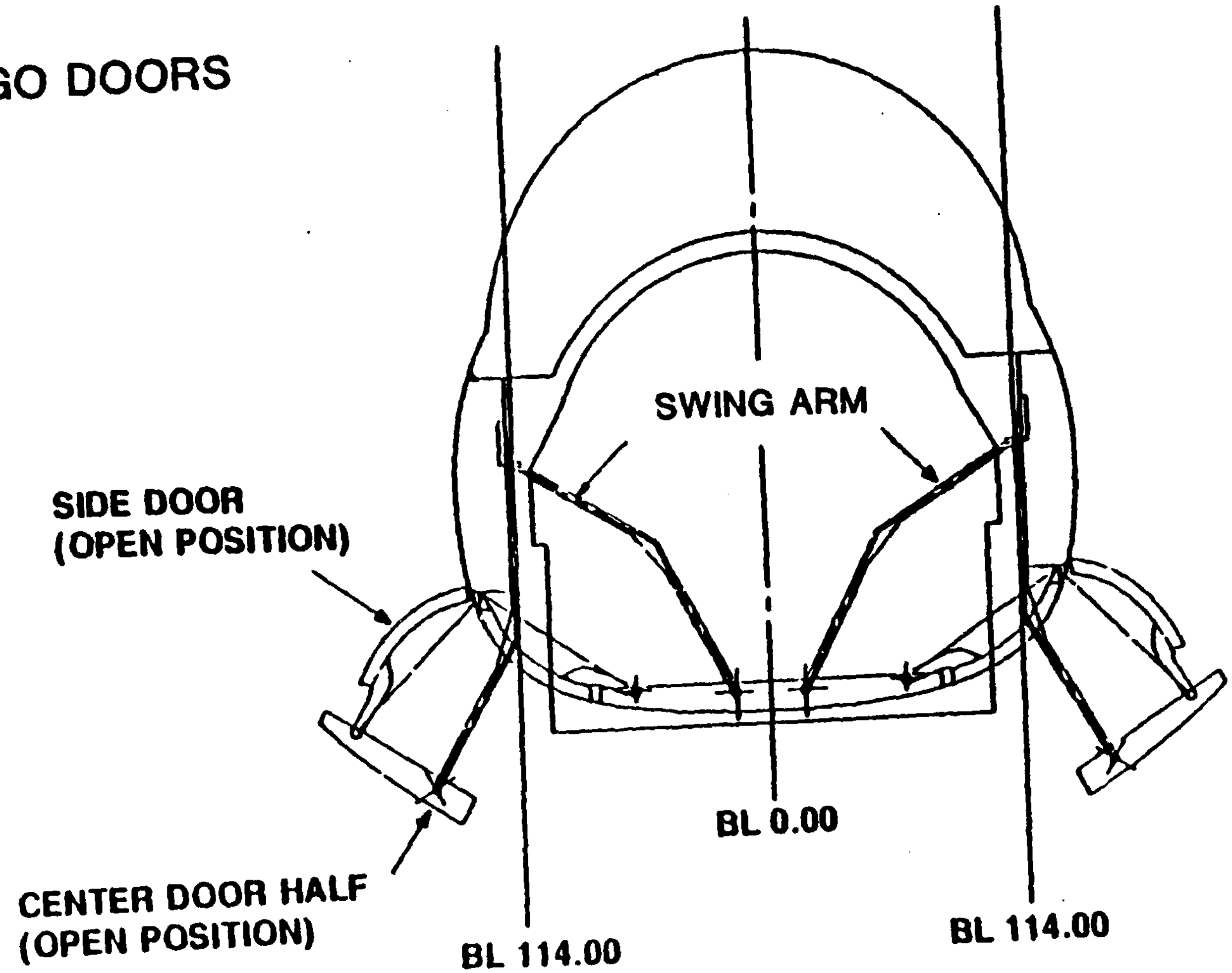
A UNIQUE C5A STRUCTURAL MODIFICATION

CARGO DOORS



CONCLUSIONS

- THE PRINCIPAL METHODS OF STRUCTURAL ANALYSES UTILIZED THE FEM PLUS DETAIL ANALYSES CONSIDERING ECCENTRICITIES, JOINT (SPLICE) ANALYSES AND ACTUAL DESIGN DETAILS.

- THE TESTING ON X990 WILL EXPERIMENTALLY SUBSTANTIATE THE STRUCTURAL ANALYSIS AND WILL VERIFY THE MODIFIED STRUCTURE FATIGUE LIFE.

Lockheed
-Georgia Company

A UNIQUE C5A STRUCTURAL MODIFICATION

CONCLUSIONS

- REDESIGN ANALYTICALLY SATISFIES THE CONTRACT REQUIREMENTS OF STRENGTH – FORMAL REPORTS SUBMITTED AND APPROVED.
- X990 FULL SCALE TEST WILL EXPERIMENTALLY SUBSTANTIATE THE STRUCTURAL ANALYSIS.
- STRUCTURAL TECHNOLOGY WAS APPLIED TO INCREASE THE CAPABILITY OF AN EXISTING AIRCRAFT.

IMPROVED STRUCTURAL INTEGRITY USING Al-Li ALLOY 8090

Al-Li F-15 WING SKIN PROGRAM - OVERVIEW

Steve Forness	MCAIR
Sam Pollack	AFWAL/FIBAA
Joe Burns	AFWAL/FIBEC

ASIP1
30-NOV-87

Export Authority; 22CFR125.4(B)(13)

PROGRAM TASKS

- o MATERIAL CHARACTERIZATION
- o ANALYSIS (FINITE ELEMENT MODELING)
- o MANUFACTURING AND INSTALLATION
- o FLIGHT TEST
- o FLIGHT TEST STRAIN ANALYSIS

ASIP3
30-NOV-87

ff

MATERIAL CHARACTERIZATION TEST MATRIX

	-L-	-LT-	-ST-	TOTAL
TENSION	12	12	12	36
COMPRESSION	18	18	18	51
BEARING	9	9	-	16
Da/Dn	4	4	-	8
KIC	6	6	4	16
S .vs. N	10	11	-	21

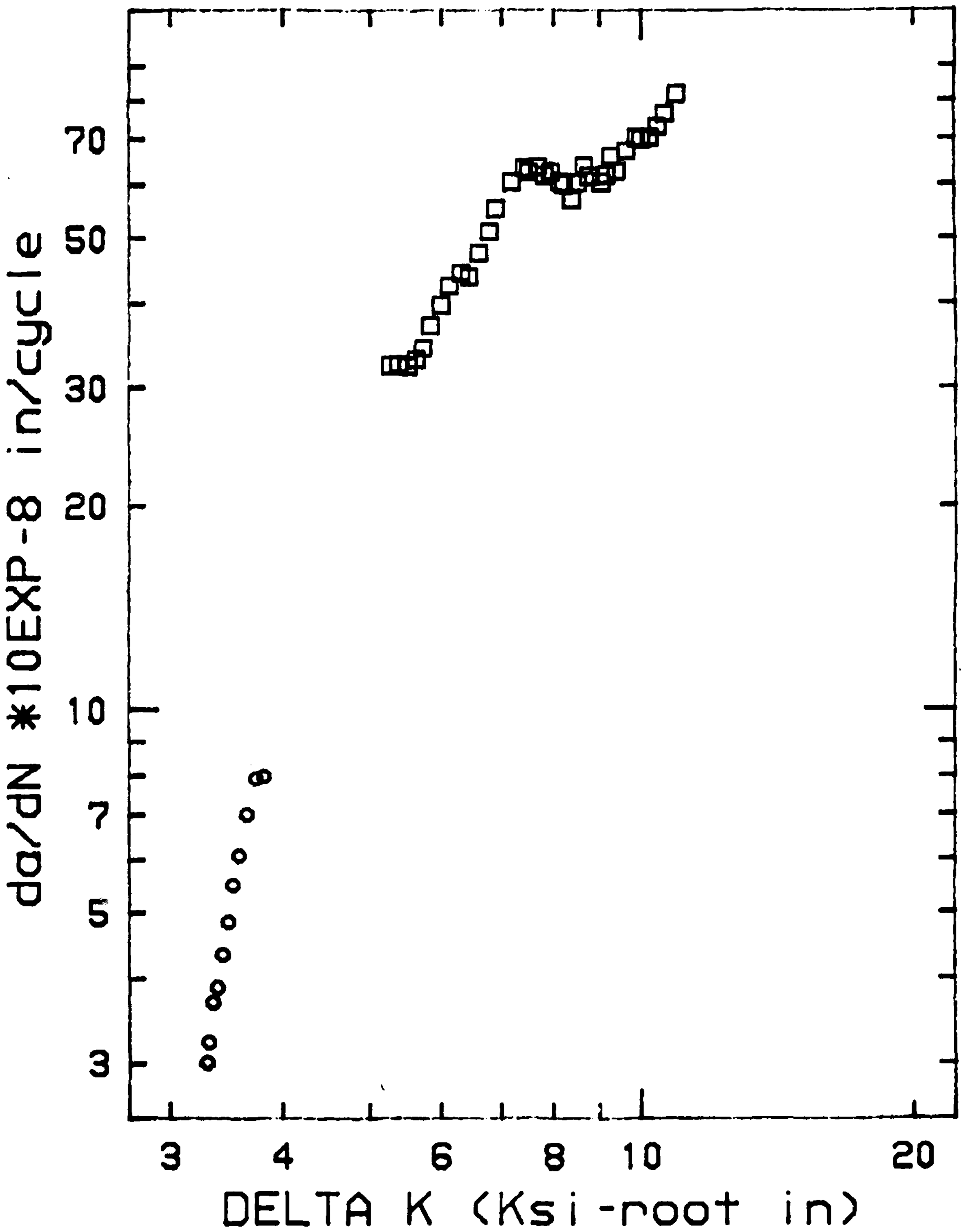
ASIP4
30-NOV-87

MATERIAL CHARACTERIZATION AVERAGE RESULTS

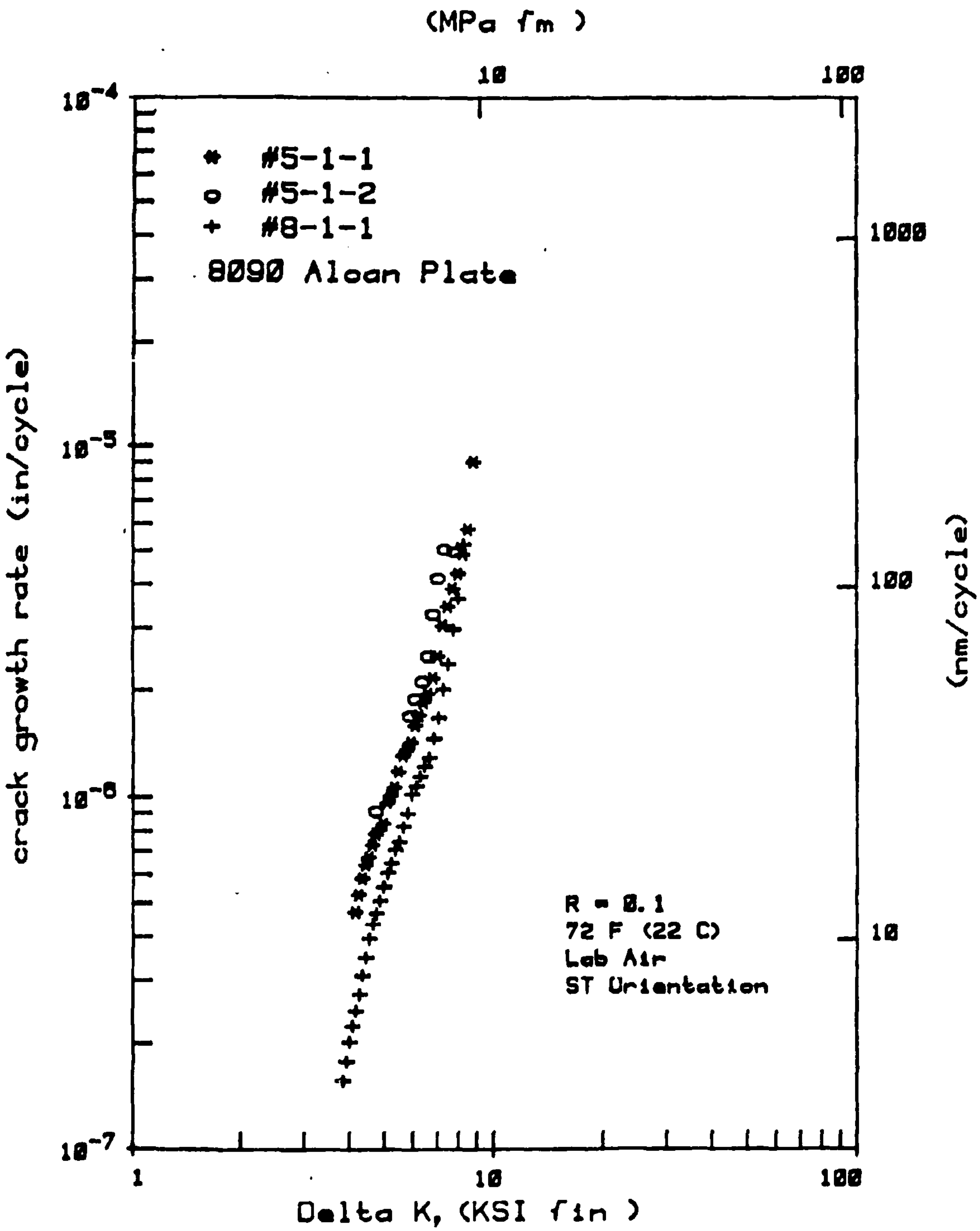
	-L-	-LT-	-ST-
TENSION (Fty)	61.7	55.8	53.0
TENSION (Ftu)	72.1	70.7	68.3
COMPRESSION	58.3	57.1	60.3
BEARING (E/D=1.5)	87.5	86.8	-
KIC	23.2	24.7	-

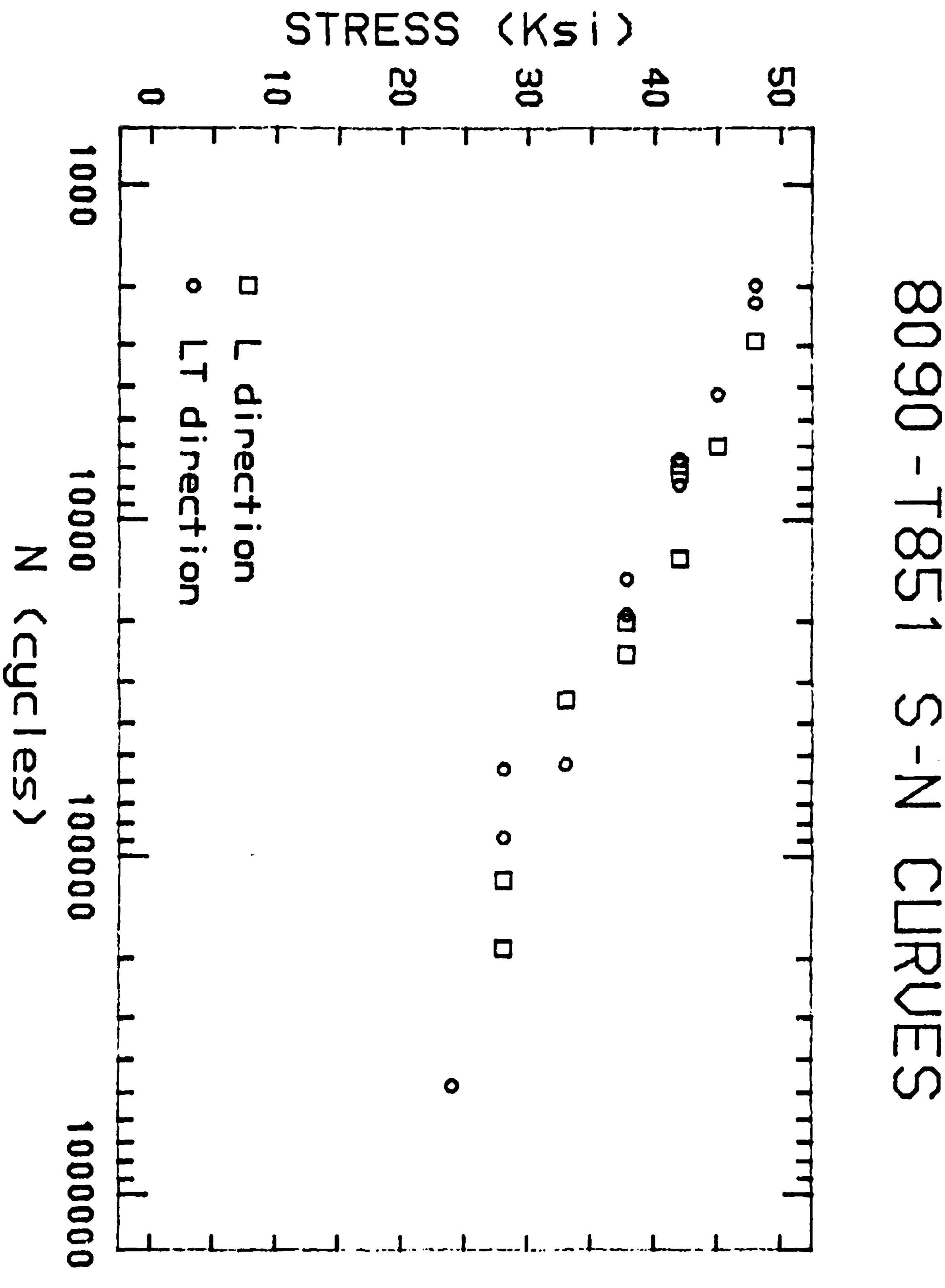
ASIP2
30-NOV-87

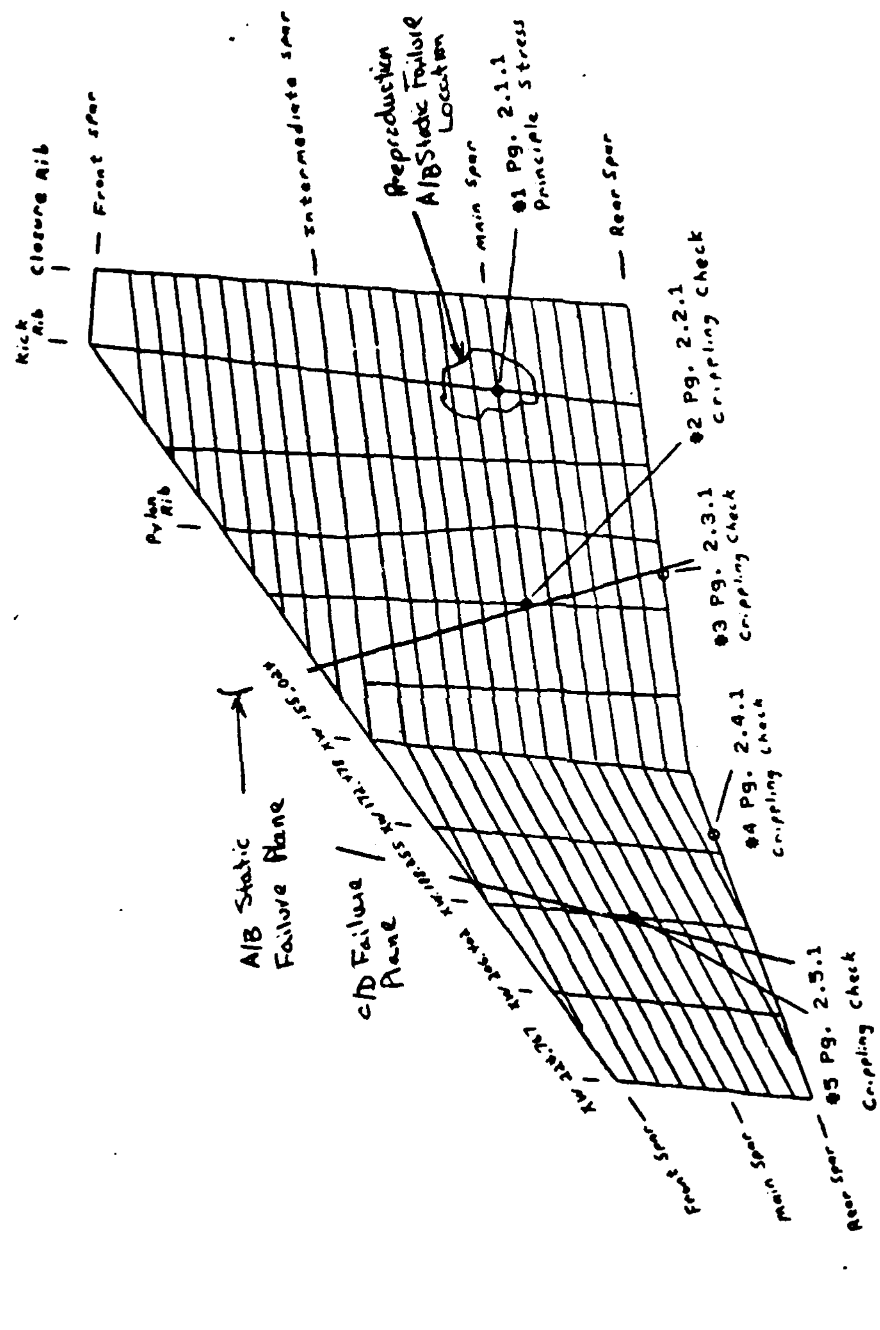
8090-T851



da/dN *10EXP-8 in/cycle
70
50
30
20
10
7
5
3
3
4
6
8
10
20
DELTA K (Ksi-root in)


(MPa √m)
10
100
10^{-4}
10^{-5}
10^{-6}
10^{-7}
* #5-1-1
o #5-1-2
+ #8-1-1
8090 Aloan Plate
1000
100
10
crack growth rate (in/cycle)
(nm/cycle)
R = 0.1
72 F (22 C)
Lab Air
ST Orientation
1
10
100
Delta K, (KSI √in)


8090-T851 S-N CURVES
STRESS (Ksi)
0
10
20
30
40
50
1000
10000
100000
1000000
10000000
L direction
LT direction
N (cycles)

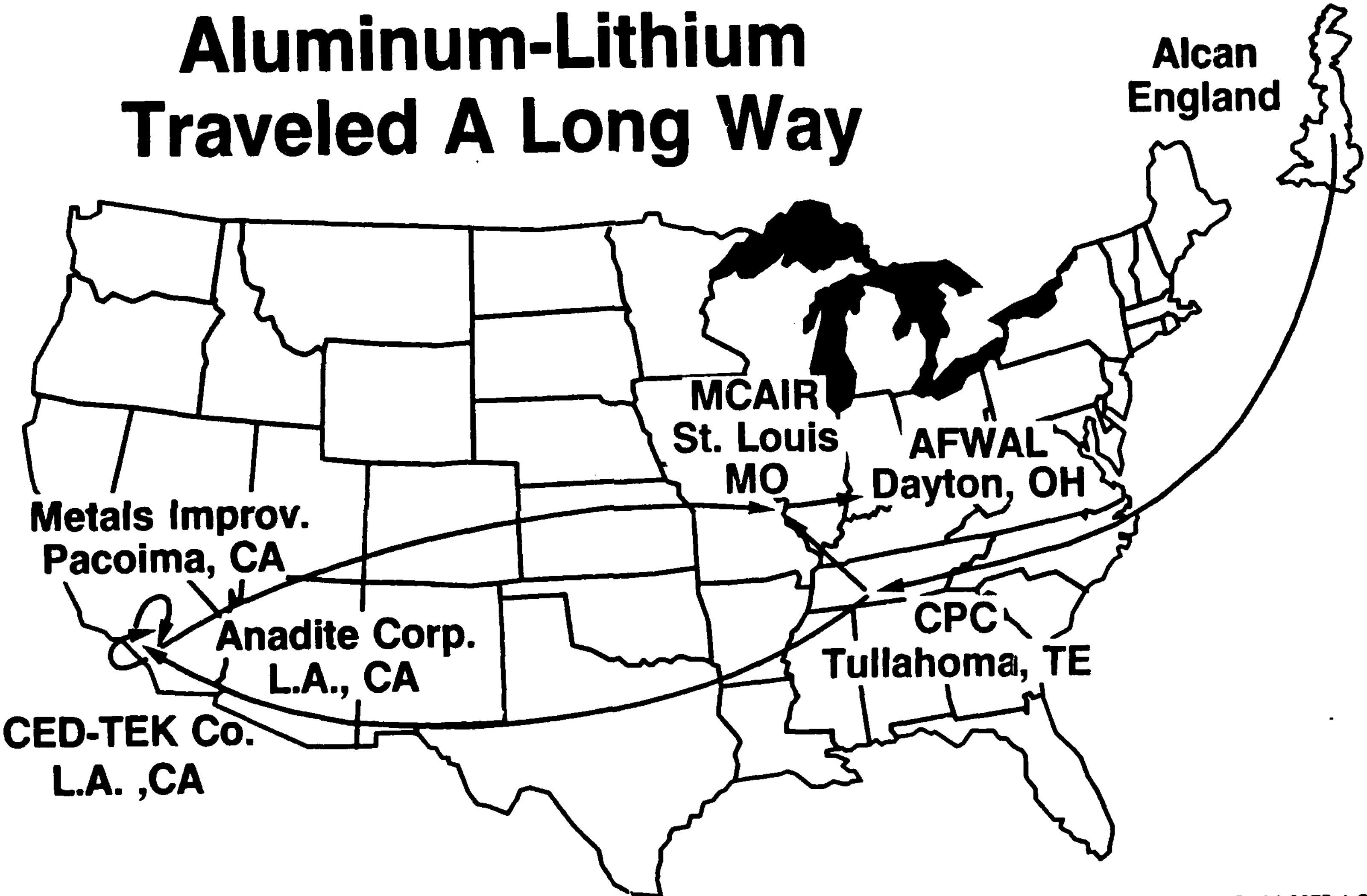


Kick Rib
Closure Rib
Front Spar
Intermediate spar
Preproduction A/B Static Failure Location
Main Spar
#1 Pg. 2.1.1 Principle Stress
Rear Spar
#2 Pg. 2.2.1 Crippling Check
Pylon Rib
#3 Pg. 2.3.1 Crippling Check
A/B Static Failure Plane
XW 155.024
XW 172.478
XW 188.555
C/D Failure Plane
XW 206.404
XW 224.767
#4 Pg. 2.4.1 Crippling Check
#5 Pg. 2.5.1 Crippling Check
Front Spar
Main Spar
Rear Spar

MARGINS OF SAFETY

LOCATION	SKIN/UNDERSTRUCTURE CONFIGURATION	
	A/B PROD.	ALLI on A/B
PRIN. STRESS	.03	.12
MAIN SPAR (I)	.19	.66
REAR SPAR (I)	.02	.42
REAR SPAR (O)	.22	.31
MAIN SPAR (O)	.01	.11

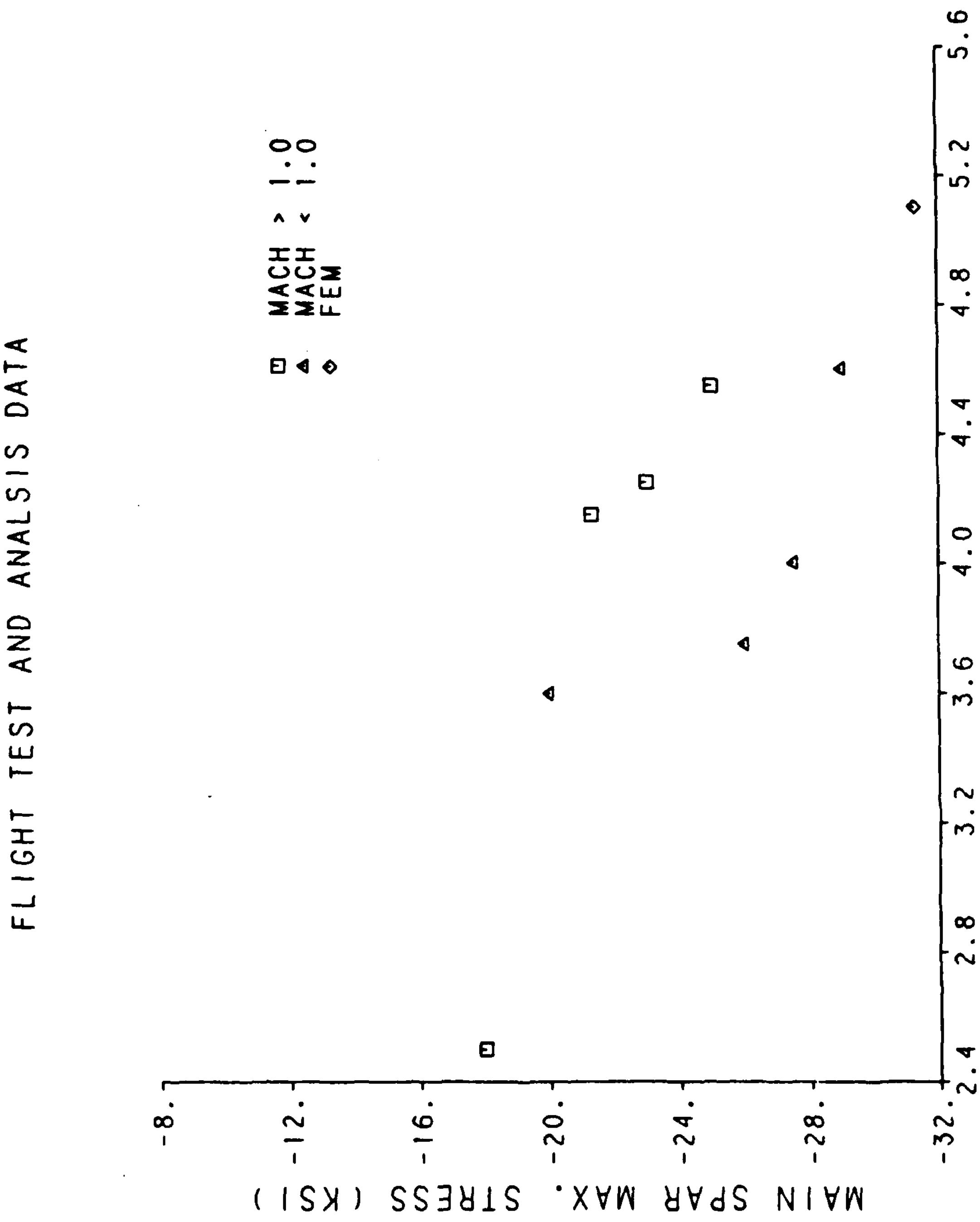
Aluminum-Lithium Traveled A Long Way



GP61-0975-1-G

FLIGHT TEST MANUVERS

MACH NO.	ALTITUDE	G'S(MAX.NZ)
1.6	30	5.6
1.2	30	6.5
1.05	30	5.8
.90	20	6.5
.80	15	7.2
1.05	30	6.0
1.20	30	6.1
1.20	30	7.0
.90	20	6.6


FLIGHT TEST AND ANALSIS DATA
MACH > 1.0
MACH < 1.0
FEM
MAIN SPAR MAX. STRESS (KSI)
-8.
-12.
-16.
-20.
-24.
-28.
-32.
2.4
2.8
3.2
3.6
4.0
4.4
4.8
5.2
5.6

ELEMENT TEST RESULTS

This data was obtained under the MCAIR Al-Li Wing Skin program for the F-15 STOL technology demonstrator, Contract F33615-80-C-3227. The tests were performed on ALCAN's production stock of 8090 Al-Li plate (1.8" thick) in the T851 condition. AFWAL/FIBEC and MCAIR did the actual testing.

COMPRESSION (L AND LT)

SPECIMEN #	F_{cy} (ksi)	F_{cu} (ksi)	MOD (msi)
1	56.3	80.6	11.9
2	56.8	78.0	11.8
3	57.3	77.6	11.6
4	58.4	79.0	11.8
5	58.9	77.7	12.2
6	57.4	87.1	11.4
7	57.6	77.1	12.1
8	58.4	78.3	11.8
9	57.6	78.6	12.1
10	59.7	79.6	11.9
11	59.7	81.3	11.5
12	59.7	81.4	11.9
13	59.7	82.4	11.7
14	59.0	81.5	11.4
15	59.7	81.5	11.9
16	59.0	80.9	11.8
17	59.7	81.5	11.6
18	58.9	81.9	11.3
19	57.9	77.3	11.0
20	57.0	79.9	12.6
21	57.6	77.4	11.4
22	59.0	78.5	12.0
23	58.1	78.2	12.1
24	58.2	81.0	11.6
25	58.6	79.8	11.8
26	59.4	78.7	11.8
27	58.1	77.9	12.0
28	58.4	77.9	11.9
29	58.4	80.2	11.7
30	58.4	79.4	12.2
31	58.7	78.7	11.9
32	58.9	78.4	12.1
33	58.4	78.6	11.8
34	58.2	80.9	11.7
35	59.0	78.0	11.8
36	58.5	78.8	11.6

TENSILE (L and LT)

SPECIMEN #	F_{ty} (ksi)	F_{tu} (ksi)	MOD (msi)	
37	61.0	72.6	11.7	
38	60.5	71.4	11.3	
39	60.6	71.5	11.0	
40	61.2	71.8	11.2	
41	61.3	71.8	11.3	
42	60.0	71.3	11.1	
43	61.4	72.2	11.2	
44	60.3	71.2	11.3	
45	60.6	71.4	11.2	
46	55.4	70.6	11.2	
47	55.0	69.9	11.1	
48	55.6	70.6	11.1	
49	56.4	70.9	10.7	
50	56.0	72.6	8.3	
51	56.8	71.3	10.5	ALL VALUES ARE
52	61.7	72.4	10.8	ESTIMATED FROM THE
53	56.3	70.9	10.0	STRIP CHART
54	62.4	72.4	10.9	
55	55.2	70.6	10.8	
56	55.6	70.4	11.3	
57	56.4	70.6	11.4	
58	55.8	69.9	10.9	
59	56.3	70.5	11.0	
60	56.1	71.1	11.0	

COMPRESSION (ST)

PLATE #	F_{cy} (ksi)	MOD (msi)
1	60.1	12.5
1	60.8	12.5
1	60.6	12.5
1	60.7	12.6
1	60.4	12.5
1	60.9	12.1
1	60.7	12.4
2	59.9	12.5
2	59.8	13.0
2	60.0	12.8
2	60.2	11.9
2	59.7	12.8
2	59.6	12.1
2	60.2	12.6
2	59.6	11.8

TENSILE (ST)

PLATE #	F_{ty} (ksi)	F_{tu} (ksi)	MOD (msi)	% elongation
1	53.0	61.0	12.2	2.0
1	53.5	65.5	12.1	- 2.0
1	53.5	70.0	12.1	3.0
1	53.5	63.5	11.8	2.0
2	52.5	69.5	12.3	3.0
2	52.5	68.5	11.3	3.0
2	52.5	69.5	11.7	3.0
2	53.0	67.5	12.1	3.0
2	51.5	68.0	11.9	3.0
2	53.0	68.0	12.2	2.0
2	53.0	69.0	12.8	4.0
2	52.5	68.0	11.7	4.0

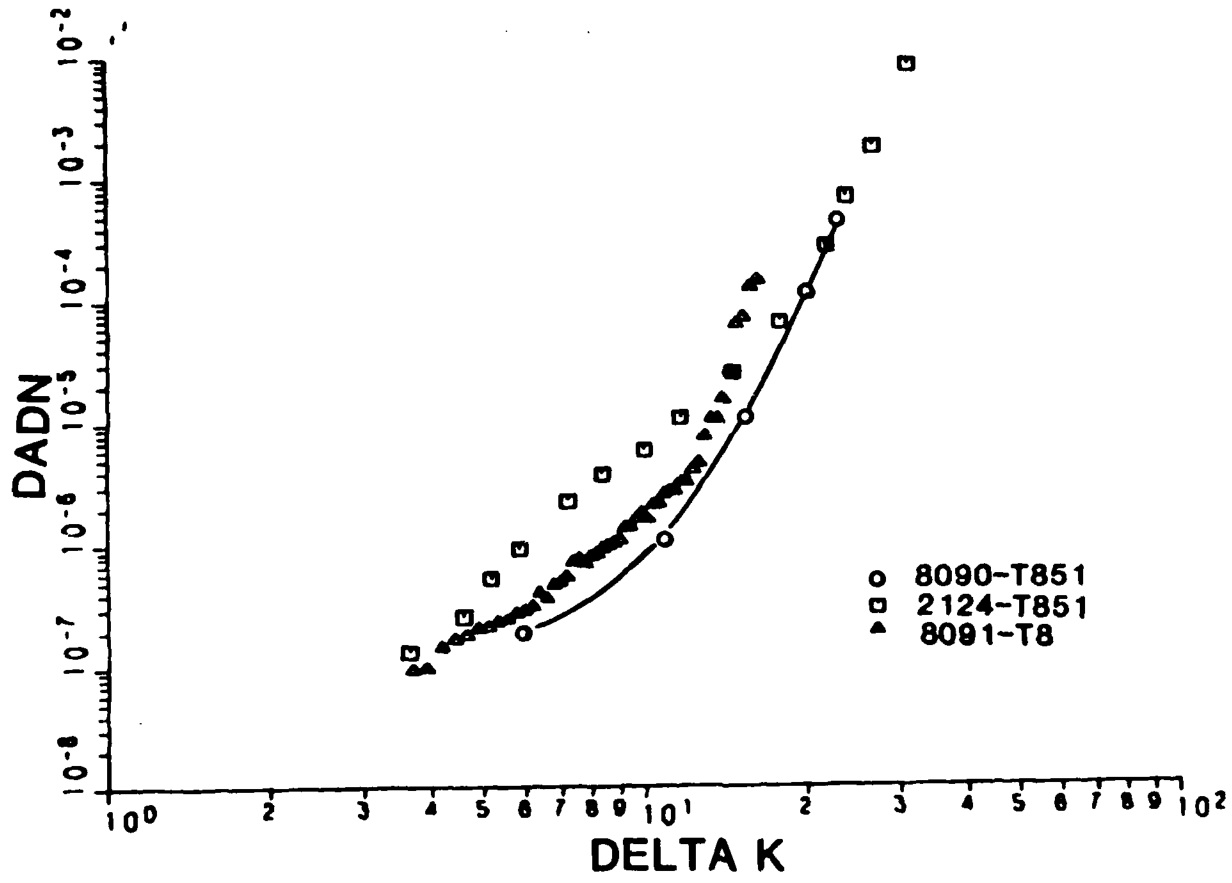
BOLT BEARING (L and LT)

PLATE #	F_{bry} (ksi) e/d = 1.5
1	87.0
1	87.3
1	87.5
1	86.5
1	86.9
1	86.2
1	87.2
2	87.2
2	88.1
2	87.9
2	87.2
2	88.0
2	86.4
2	86.9
2	87.2
2	86.3

FRACTURE TOUGHNESS (tested at ALCAN)

PLATE #	K_{1C} (ksi in)	ROLLING DIRECTION (L or LT)
1	23.0	L
1	23.2	LT
2	23.1	L
2	23.3	LT

ALUMINUM LITHIUM DADN VS DELTA K -LT-



Lessons Learned from the T-46A
Durability and Damage Tolerance Program

by

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To be presented at:

1987 USAF Aircraft/Engine Structural Integrity Program
(ASIP/ENSIP) Conference

-Snyopsis-

This paper presents the T-46A Durability and Damage Tolerance Program accomplishments through full scale engineering development. The T-46A design criteria, analysis, development test, and the full scale durability test are discussed.

A detail discussion of the findings of the Pre-Production Design Verification (PDV) tests, full scale durability test and proposed design changes to be incorporated in the production articles are described. Special manufacturing processes such as application of cold work, installation of interference fasteners are also presented.

Lessons learned from the program are discussed in detail. It is believed that use of the lessons learned discussed in this paper for future structural integrity programs will significantly reduce durability and damage tolerance technical risk.

Lessons Learned from the T-46A
Durability and Damage Tolerance Program

I. Introduction

Military Standard 1530A "Aircraft Structural Integrity Program (ASIP) Airplane Requirements" (Ref 1) defines the overall requirements necessary to achieve structural integrity of USAF airplanes. To partially satisfy ASIP requirements, a Durability and Damage Tolerance Program was established in the T-46A Full Scale Engineering Design Development (Refs 2, 3). This was started in May 1982 and was concluded in March 1987.

II. Design Criteria

The service life requirement of T-46A was 20,000 flight hours. In order to insure structural safety and achieve low cost maintenance and operational readiness throughout the design service life of the aircraft, the durability and damage tolerance design requirement was specified that the structure will neither reach functional impairment nor safety limit within two design lifetimes.

III. Analysis

In the preliminary design, allowable stress levels for sizing were selected to satisfy the structural design criteria. Critical areas were further selected for detail durability and damage tolerance analysis. These areas were selected on the basis of preliminary stress analysis, geometrical configuration and materials.

The original randomized flight-by-flight spectrum was developed with mission profiles defined in the Next Generation Trainer (NGT) Request for Proposal (RFP). A Durability and Damage Tolerance Analysis was performed to determine the analytical structural lives. After the T-46A Critical Design Review (CDR) in August 1983, the original design spectrum was revised to include the anticipated operational air speeds. The Air Training Command (ATC), USAF, revised the air speeds in the original mission profiles which were much lower than the aircraft performance capability and previous trainer service experience. Durability and Damage Tolerance Analysis were rechecked with the revised flight by flight randomized spectrum to determine the impact of this change on the structural design. The reanalysis led to a design change for the upper skin from 7075-T6 to 2024-T3 and reinforcement of spar and stringer on the horizontal stabilizer.

IV. Design Development Test

The design development test program included coupon tests, structural configuration tests and pre-production design verification tests.

a. Coupon Test Program

Coupons representing various parts of the forward fuselage, aft fuselage, empennage and wing were tested under constant amplitude and flight-by-flight spectrum to obtain crack growth data, truncation levels for design development testing and validation of the analytical crack growth methodology.

A total of 90 coupons representing five materials and four airframe components were manufactured for the original design flight by flight spectrum test. Three basic structural configurations were represented by open hole, filled hole, and various degrees of load transfer. Design gross stresses for the respective airframe components were used as references for the selection of maximum spectrum test stresses. After a truncation study of analysis and coupon tests, the truncation range of 3000 psi was selected. Spectrum coupon test results showed that the crack growth analysis was in good agreement with test results (Ref 4).

An additional 30 coupons were manufactured for the revised spectrum testing. The results indicated that wing spar flanges required design modification.

b. Structural Configuration Test Program

Three structural configuration; wing splice front spar, wing splice/skin and lower wing cover, were selected for durability, damage tolerance and residual strength tesing.

(1) Durability Testing

The original design flight by flight spectrum was applied to three structural configurations for two lifetimes of durability testing. Inspection conducted at end of two lifetime testing showed no signs of crack initiation in any of these three specimens.

(2) Damage Tolerance Testing

Damage tolerance testing were intiated after two lifetimes of durability testing. A thorough structural inspection of these specimens were conducted. Fasteners were removed for hole inspections and flaw insertions. Four locations on each specimen were inserted with artificial flaws with a jeweler's saw and an Exactor knife. At end of one lifetime damage tolerance testing, no sign of crack growth or damage were found on specimens. The wing splice/skin specimen were further tested for an additional two lifetimes of damage tolerance but no crack growth was found.

(3) Residual Strength Testing

Only the lower wing cover underwent residual strength testing after the completion of one lifetime damage tolerance testing. No damage or flaw growth was found. This same flaw was then extended to 8.42" and was tested to failure. The failure load reached 135% of design limit load.

In conclusion, the results of these tests showed that the design exceeded the specification design requirements (ref.5).

c. Pre-Production Design Verification (PDV) Test Program

The objectives of PDV tests were to verify whether the design sastifys the durability and damage tolerance requirements. Tests were conducted on selected critical compmnents using the earliest available production-type parts. Three major structural components were programmed for testing to the revised flight by flight spectrum for two lifetime of durability and one lifetime of damage tolerance testing.

(1) Empennage/Fuselage Attachment

This component completed two lifetime of durability and one lifetime damage tolerance testing with the revised flight by flight randomized spectrum. A thorough inspection was conducted after two lifetime durability testing and found two cracks in the fuselage access door doubler. Five artificial corner flaws of .05" were induced with saw cut at the beginning of damage tolerance testing. The locations were selected on the basis of damage tolerance analysis and the accessibility for crack growth monitoring. At 25% lifetime of damage tolerance testing, the .05" induced flaw was increased to .10" for an additional 45% of a lifetime testing,no significant crack growth data was recorded. This same flaw was increased to .175" and testing for another 30% of a lifetime. At the completion of one lifetime damage tolerance testing, no crack growth was found. Follow on residual strength testing was performed to 100% limit load. The empennage/fuselage design satisfied the durability and damage tolerance requirements for the revised spectrum (ref 6).

(2) Engine Thrust Fitting (Engine Support Structure)

This test consisted of flight-by-flight radomized spectrum loading for two lifetime durability and one lifetime damage tolerance testing. Five flawed locations on the test specimen were imposed after two lifetime durability testing. Because of very small measured crack growth at 25% lifetime, the three .05" flaws were increased to .100" and monitored to the end of damage tolerance testing. The inspection noted very small crack growth in any of the induced flaw locations. The design met the durability and damage tolerance design requirements with the application of the revised spectrum (ref 7).

(3) Wing/Fuselage Attachment/Main Landing Gear Support Structure

The test article was a structually complete fuselage from FS 252 to FS 282 and wing from RWS 117 to LWS 117; including the landing gear backup structure. The wing leading edge, wing trailing edge, aileron and flap were not included. The wing hand forged frames at FS 252 and FS 282 were machined to the die forged dimensions. The test fuselage structure was supported by steel bulkheads at FS 210 and FS 298 in the fixture areas of the fuselage. The test area included two main frames (FS 252 and FS 282), landing gear ribs and pertinent stabilizing structure main members of the fuselage structure between FS 252 and FS 282.

The test article was planned to be subjected to three lifetimes of testing (two lifetime durability and one lifetime damage tolerance). The loading spectrum of 1000 hours each would be repeated 56.5 times. Fourteen marker band sequences of 250 hours would be applied at the end of each20% lifetime interval. The load spectrum consisted of 1263 unique flight loading conditions and 20 landing load conditions truncated to exclude conditions from one g to and including 2.25g. The testing was started on 31 May 85. On 29 Jun 85, at 11.58% of the first lifetime, the specimen was undergoing the application of 5.75g condition, the test shut down automatically. A visual inspection revealed a crack in the right hand front spar frame 252 in the nacelle outboard wall portion.

This failure investigation was jointly conducted by the Fairchild Republic Company and an Air Force Structures Review Team. The main tasks of the failure investigation included the examination of fracture surfaces of the failure area, NDI inspection of fastener holes on left side of 252 and 282 frames, verification of stress analysis with strain survey, verification of the da/dN data by coupon tests from the test article. A minimum initial repair was accomplished in order to complete a 3g detail strain survey. Additional fracture analysis for multiple hole configurations were performed as well as an indepth review of the finite element stress model.

The additional strain survey revealed that the stresses at the failure location were higher than predicted. The review of the stress analysis and internal load finite element model revealed an error. This error caused an unconservative calculation in the percentage of wing root moment distributed to the nacelle wall portion of the frame. As a result, the local region of the nacelle wall, including the outboard flange was under designed. To correct the deficiency, the 252 frame in the nacelle area was beefed up in cross sectional area. The new frame was incorporated in the durability test article and the first production aircraft (P-1).

The front spar frame 252 of the PDV test article was repaired and testing was resumed later. As the test progressed, several events occurred ; fastener head cracking, front spar cracking and WS 78 upper wing skin cracking. Each event was analyzed for an appropriate action to be incorporated in the durability test article and production articles. Larger size fasteners were installed in the durability and production articles. Revised chem mill skin thickness and increased local frame flange thickness on both front and rear spar of production articles were also determined.

At 103% lifetime the front spar at WS 96 R/H lower surface cracked and was repaired with internal steel straps. The test was resumed. No major events occurred until 186% lifetime when the L/H side frame 282 at WS 63 stub wing cracked in the lower cap and vertical web.

The failure was again jointly investigated by the Fairchild Republic Company and an Air Force Structures Review Team. The failure investigation included the evaluation of test loads, stress analysis, fracture analyses, materials evaluation, fractographic and metallurgical examinations.

In the previous analysis, both 7175-T73652 hand and die forging crack growth rate in the low ΔK regions were estimated with a very limited test

data. To eliminate the uncertainty, coupon specimens for hand forging machined from the test article and die forging machined from disposed part were tested to generate da/dN in the low ΔK regions. The results showed die forging had a lower threshold stress intensity factor ΔK_{th} (~1.5 KSI$\sqrt{in}$) and slight higher da/dN.

Additional coupon spectrum testing of the 7175-T73652 die forgings were also performed. The results indicated that the existing rear spar required a design change or modification to cold work holes to ensure adequate durability structural integrity. The design change for the production articles proposed three options; cold work holes, cold work holes with steel strap reinforcement, or increased flange thickness to reduce stress levels.

In accordance with the test plan,one lifetime of damage tolerance testing was planned to be conducted after the completion of two lifetimes of durability testing on this specimen. Testing was terminated because additional repairs would result in an unrepresentative test article. Subsequent to the test termination a detail teardown inspection was accomplished but no additional cracks were found.

V. Full Scale Durability Testing

A complete full scale airframe was planned to be tested for two lifetimes of flight-by-flight spectrum. The test article was the fourth complete airframe constructed and was representative of the production aircraft. The test article had the redesigned die forging frame at FS252. As much as schedule allowed, structural modifications identified by the pre-production design verification tests were incorporated on the full scale article.

Full scale durability testing was started on 22 Jul 86 and one lifetime was complected on 21 Jan 87. At 25% of the first lifetime, additional strain gages were installed in the FS 252 and 282 armpit areas and the left rear wing spar lower cap. This generated detail stress distribution for correlation with the wing/fuselage PDV tests and for comparison with the applied front and rear spars load distribution. Because of the failure of the PDV test article at 1.86 lifetimes, selected fastener holes of the full scale article were inspected with low frequency eddy current techniques in addition to the regular inspection program specified in the full scale durability test plan at every 25% lifetime interval.

During the first lifetime schedule inspection, some broken fasteners, a damaged outboard wing rib (at WS 201 due to cylinder overload), a failed wing rib at wing station 63.5, and two upper wing skin cracks about 2" long were found. The inspection at 100% of one lifetime revealed no cracks in the front and rear spars.

The Air Force Structures Review Team recommended that selected fastener holes be cold worked, the upper wing skin be replaced, and redesign ribs be installed at wing stations 63.5 and 80.0 for the second lifetime durability testing. On 13 Mar 87,the T-46A program was concluded. The second lifetime testing was not started.

VI. Lessons Learned

Although some of the design changes and recommendations couldn't be implemented, several important lessons learned deserve to be mentioned. These should be very useful for planning future structural integrity programs.

a. Design Spectrum

The design spectrum should be defined as early as possible. The importance of the mission profiles should be emphasized to the using command so that they are aware of the impact that changes will have on the structural design and possibly performance. Once the prelminary design has been completed, a change in the design spectrum requires a complete assessment of design development and verification test programs. Design changes should be incorporated on all full scale test articles.

b. Materials Selection/Geometric Configuration

Materials types or forms used for the development test article should be the same as those used in the production article. If substitution is required due to the schedule constraint, an assessment should be performed to determine the impact on structural integrity. In any event, the same geometrical configuration should be maintained. This also avoids duplicate efforts on stress analysis, test interpretation, drawings identification and manufacturing tooling.

c. Crack Growth Data

Basic facture data utilized in the design analysis should be obtained from existing sources or developed as part of the contract. Data not available from the existing sources should be generated in the early stage of the development test program. Specifically, crack growth data for the low ΔK region should be generated for the proposed materials (types, forms, etc). Estimations of crack growth rate for this region should be exercised with care.

d. Finite Element Model

The early failure of the wing/fuselage attachment pre-production design verification test article led to an extensive and time consuming review of the finite element model. This effort concluded that inappropriate stiffness assumptions were made in the vertical nacelle flange areas and the resulting stress values were underestimated. Analysis indicated that a portion of the frame required redesign. This redesign adversely affected the original full scale engineering development program. It was concluded that a special effort to review the finite element model should be completed before the release of final engineering drawings. A detailed review of the model with the aid of computer graphic capability should be employed. The review should include verification of the modeling philosophy and accuracy. The engineering drawings should be compared with the model. The model load condition distribution and the boundary condition assumptions should be checked. The deflection and stress distribution

thoroughly checked to insure proper load distribution. The finite element model should be continuously updated to reflect the current structural configuration.

e. *Crack Growth Methodology Verification*

Crack growth analysis methodology should be verified with the coupon test program. The test stress levels should be similar to the design stress levels in the airframe structure. Due to the spectrum sensitivity, any spectrum change should be verified with the coupon test program. Any modification or design changes resulting from a spectrum change should be included in the test articles and production articles as soon as practical.

f. Test Schedule

The purpose of development tests are to uncover any possible design deficiency, and generate test information to be applied to the full scale and production articles. The schedule should be conservative with appropriate recognition of test down time for repairs and inspections. The overall test planning should allow completion of the development testing and tear down inspection to permit incorporation of changes into the full scale test article. Any unrealistic schedule will affect the overall program schedule.

VII. Conclusion

The T-46A Durability and Damage Tolerance Program was established under the T-46A Structural Integrity Program to ensure that engineering development would result in an airframe that satisfied the durability and damage tolerance requirements specified in the Air Vehicle Specification. Ample information was generated from the engineering design, analysis and development test program which were incorporated in the full scale durability test and production article. Several important lessons learned are discussed. Incorporation of the lessons learned discussed above for future programs will significantly reduce durability and damage tolerance technical risk.

VIII. Acknowledgement

The author wishes to thank Mr Stephen Powers, Chief Systems Engineer, ASD/AFY, Mr Raymond Veldman, Lead Structures Engineer, ASD/AFE ,Mr John Hopkins,ASD/AFE and Mr Bob Kidd,ASD/YCE for their helpful discussions and suggestions during the course of the preparation of this paper.

References

1. Military Standard, Aircraft Structural Integrity Program, Airplane Requirements MIL-STD-1530A, 11 Dec 75.

2. T-46A Aircraft Structural Integrity Program Master Plan. Rev E, 22 Sep 86.

3. Military Specification Airplane Damage Tolerance Requirements, MIL-A-83444 (USAF) 2 Jul 74.

4. Final Report Revised T-46A Coupon Test Program, FRC Report GT 210R0104, Amend 1, 15 May 84.

5. Final Report Design Development Structural Configuration Test of Wing Splice and Wing Lower Cover, FRC Report GT 210R0200 Rev A, 31 July 84.

6. Test Report T-46A Preproduction Design Verification Test No. 3, Empennage/Fuselage Attachments, FRC Report GT 210R0330, 29 Aug 86.

7. Test Report T-46A Preproduction Design Verification Test No. 5, Engine Thrust Fitting/Engine Support Structure, FRC Report GT 210R0350, Aug 86.

LESSONS LEARNED FROM THE T-46A DURABILITY AND DAMAGE TOLERANCE PROGRAM

HSING C YEH
DEPUTY FOR C-17
AERONAUTICAL SYSTEMS DIVISION (AFSC)
UNITED STATES AIR FORCE
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

T-46A

AIRCRAFT STRUCTURAL INTEGRITY PROGRAM(ASIP)

MAY 1982-MARCH 1987

STRUCTURAL DESIGN CRITERIA

* SERVICE LIFE - 20,000 FLIGHT HOURS PER ONE LIFETIME

* DURABILITY - NO FUNCTIONAL IMPAIRMENT WITHIN TWO(2) DESIGN LIFETIMES

* DAMAGE TOLERANCE - NO STRUCTURAL FAILURE WITH THE APPLICATION OF P_{LT} WITHIN TWO(2) DESIGN LIFETIMES FOR IN-SERVICE NON-INSPECTABLE STRUCTURE

DURABILITY AND DAMAGE TOLERANCE CONTROL PLAN

* BASIC FRACTURE DATA
* FRACTURE CRITICAL PARTS LIST
* NDI PROGRAM
* MATERIAL PROCUREMENT AND MANUFACTURING PROCESS SPECIFICATION
* TRACEABILITY REQUIREMENTS
* ANALYSIS, DEVELOPMENT TESTING, AND FULL SCALE TESTING

SPECTRUM DEVELOPMENT

- ORIGINAL SPECTRUM
  - BASED ON SPECIFICATION DEFINED IN THE NEXT GENERATION TRAINER (NGT) REQUEST FOR PROPOSAL (RFP)
  - MAXIMUM SPEED 200 KEAS
- REVISED SPECTRUM
  - IMPLEMENTED AFTER CRITICAL DESIGN REVIEW (CDR) IN AUGUST 1983 AND REQUESTED BY ATC
  - MAXIMUM SPEED 275 KEAS AND 22 % OF ALL OCCURRENCES ABOVE 200 KEAS

DURABILITY AND DAMAGE TOLERANCE ANALYSIS

* PRELIMINARY ANALYSIS
  * SIZING STRUCTURE
  * SELECT STRESS LEVELS

* DETAIL ANALYSIS
  * ENSURE SATISFYING TWO LIFETIME DURABILITY AND DAMAGE TOLERANCE REQUIREMENTS

DESIGN DEVELOPMENT TEST

* COUPON TEST
* STRUCTURAL CONFIGURATION TEST
* PRE-PRODUCTION DESIGN VERIFICATION TEST

COUPON TEST PROGRAM

* CRACK GROWTH METHODOLOGY VALIDATION
* SPECTRUM SENSITIVITY
* TRUNCATION LEVEL STUDY

STRUCTURAL CONFIGURATION TEST

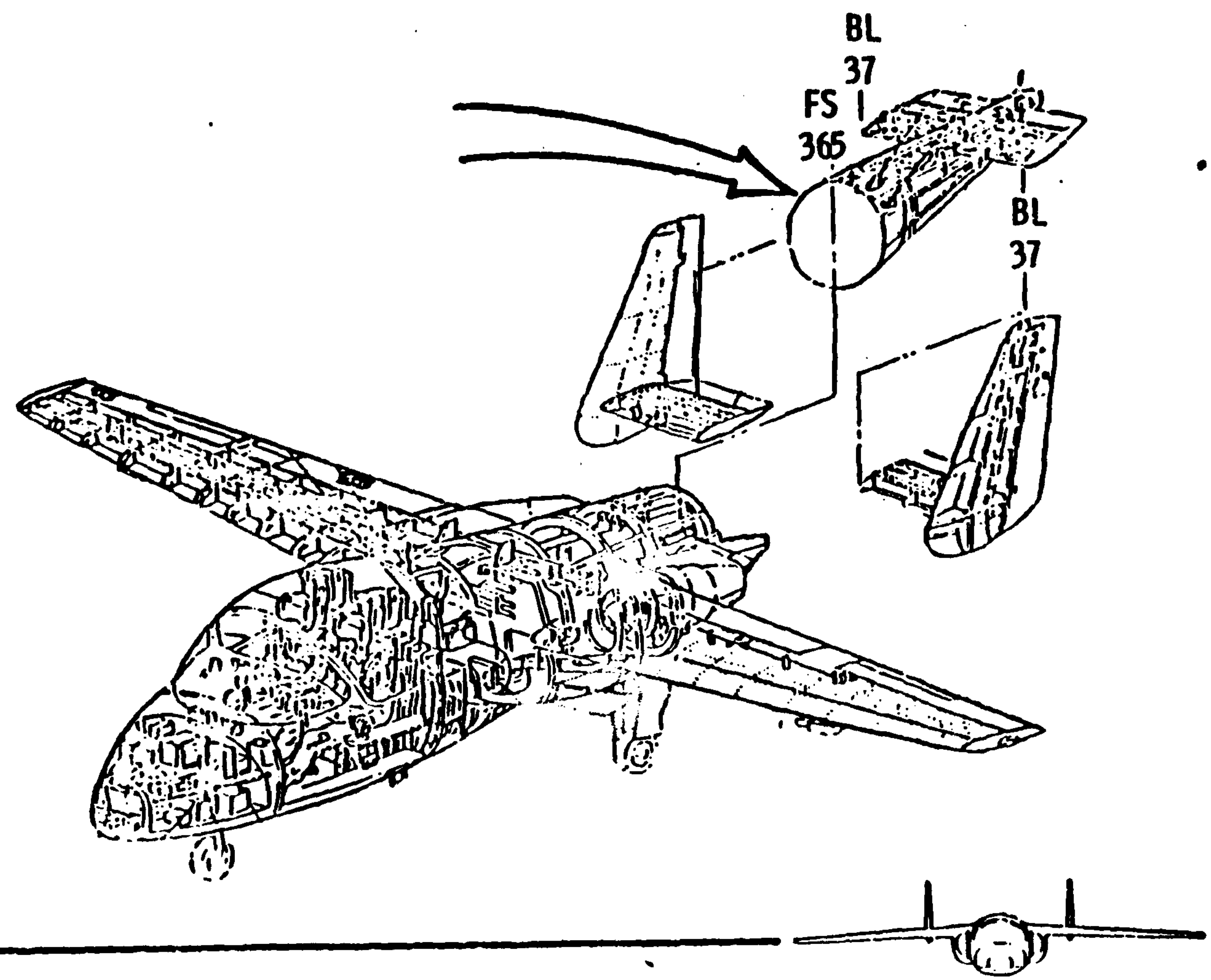
* DESIGN CONCEPT TEST
* DURABILITY AND DAMAGE TOLERANCE TEST
* RESIDUAL STRENGTH TEST

PRE-PRODUCTION DESIGN VERIFICATION TEST

* EMPENNAGE/FUSELAGE ATTACHMENT
  2 LT DURABILITY (APR.84-JAN.85)
  1 LT DAMAGE TOLERANCE (JUN.85-DEC.85)
* ENGINE THRUST FITTING
  2 LT DURABILITY (SEPT.84-MAR.85)
  1LT DAMAGE TOLERANCE (JUN.85-OCT.85)
* WING/FUSELAGE ATTACHMENT/MAIN LANDING GEAR SUPPORT STRUCTURE
  1.86 LT DURABILITY (JUN.85-JUN.86)

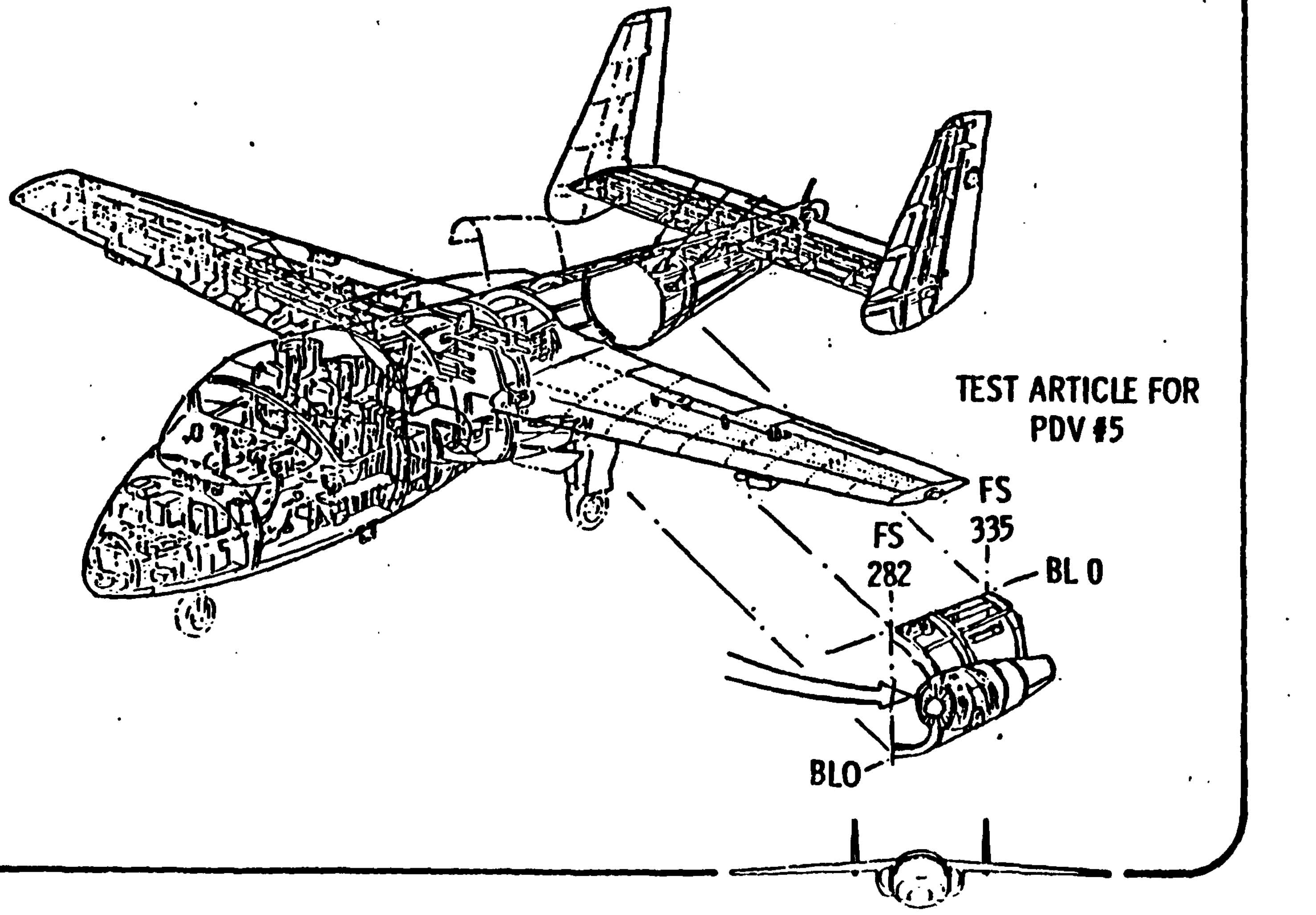
T-46A

PREPRODUCTION DESIGN VERIFICATION (PDV) TEST
PDV #3 — EMPENNAGE/FUSELAGE ATTACHMENT



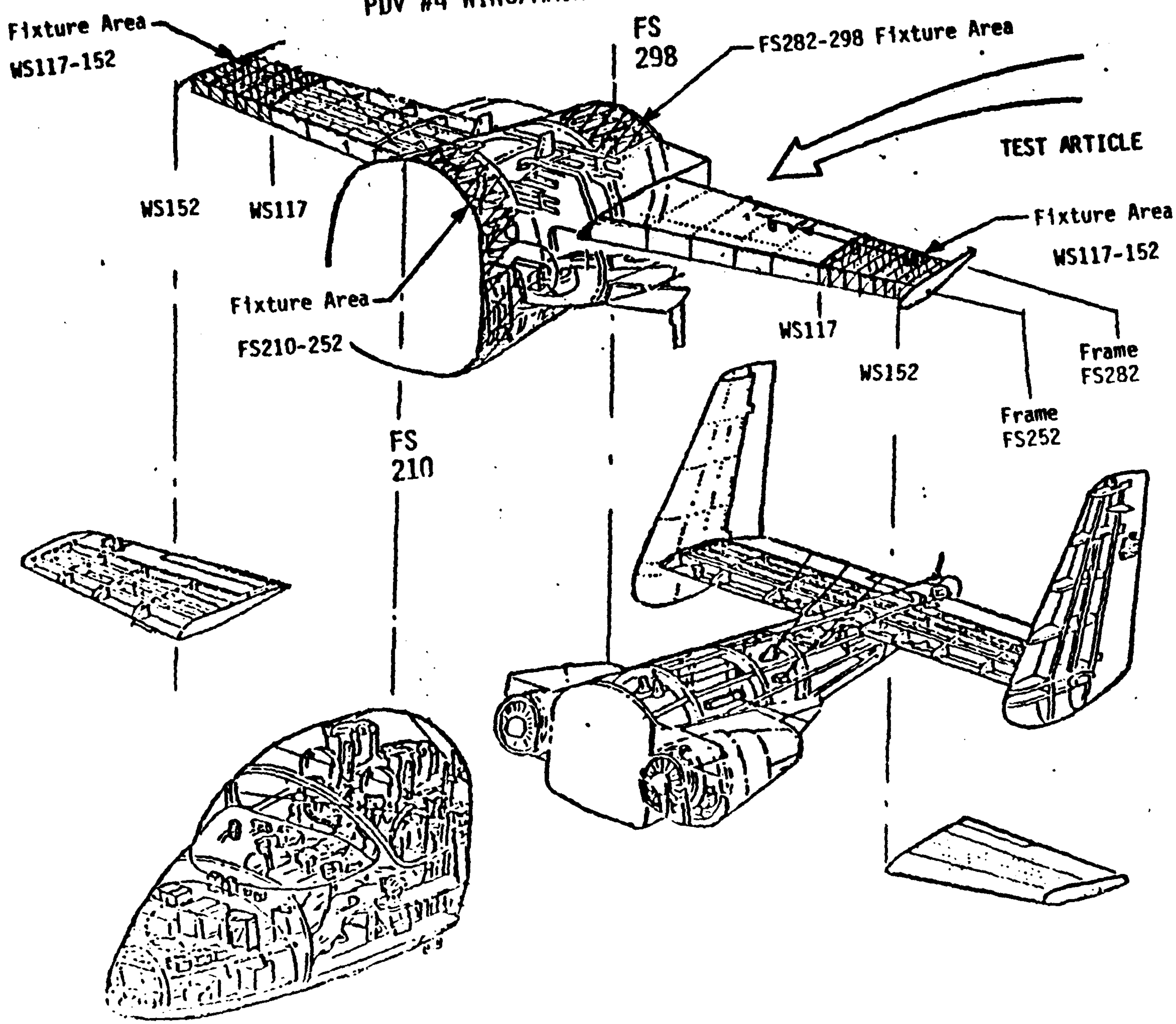
T-46A

PREPRODUCTION DESIGN VERIFICATION TEST
PDV #5 ENGINE THRUST FITTING/FUSELAGE ATTACHMENT



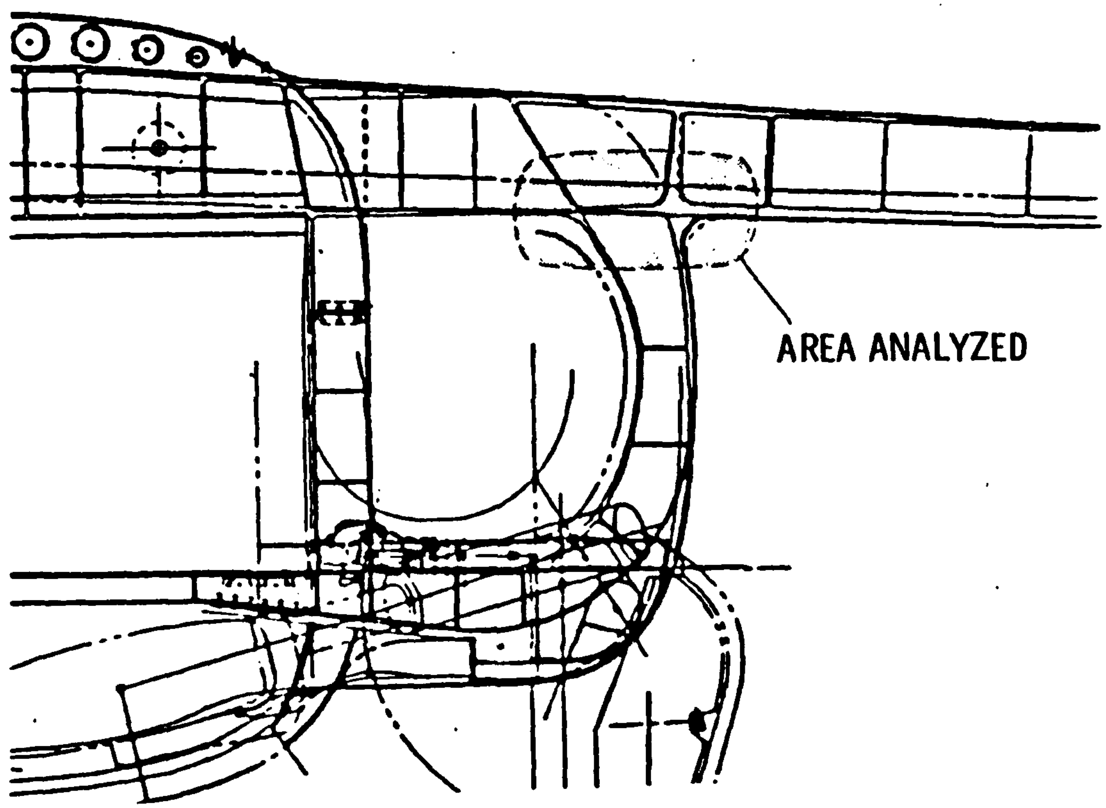
12

PREPRODUCTION DESIGN VERIFICATION (PDV) TEST
PDV #4 WING/MAIN LANDING GEAR/FUSELAGE ATTACHMENTS



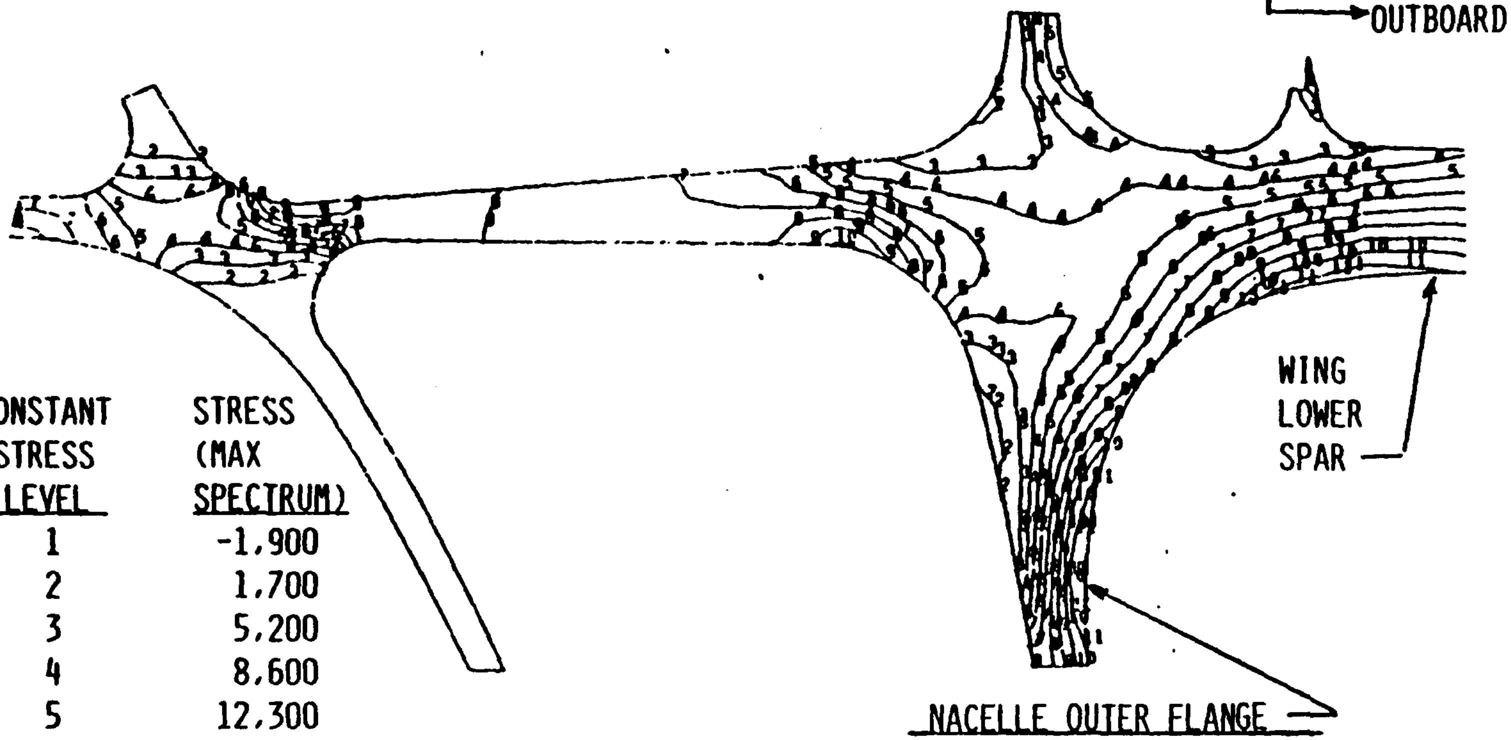
T-46A

WING/FUSELAGE FORGING @ F.S. 252



T-46A

FRAME 252
FINE GRID FEM

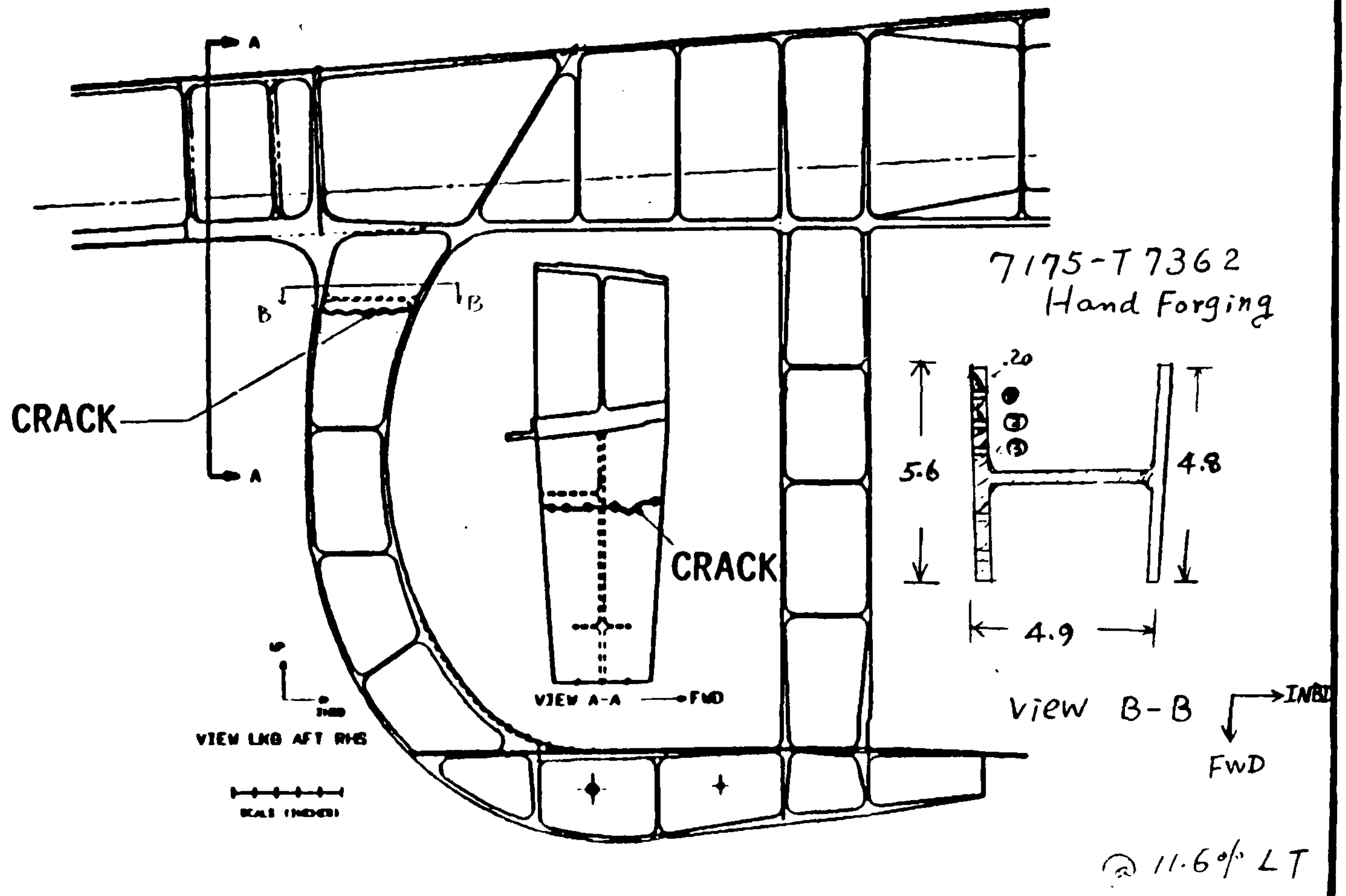


CONSTANT STRESS LEVEL	STRESS (MAX SPECTRUM)
1	-1,900
2	1,700
3	5,200
4	8,600
5	12,300
6	16,000
7	19,000
8	22,700
9	26,400
10	30,100
11	33,100

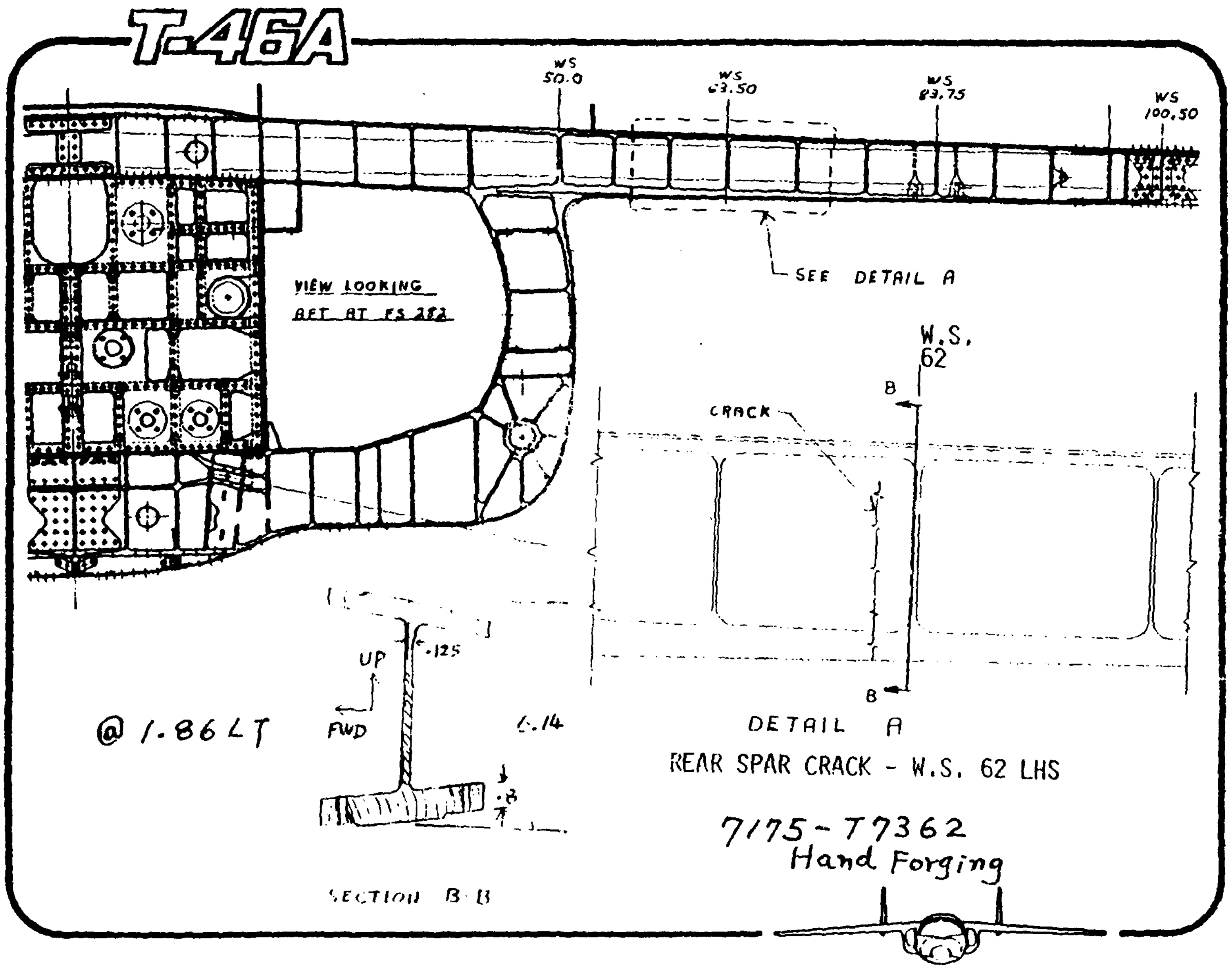
14

T-46A

FRAME 252 SHOWING LOCATION OF CRACK.



15


T-46A
WS 50.0
WS 63.50
WS 83.75
WS 100.50
SEE DETAIL A
VIEW LOOKING AFT AT FS 282
W.S. 62
B
CRACK
B
UP
FWD
.125
6.14
.8
@ 1.86 LT
DETAIL A
REAR SPAR CRACK - W.S. 62 LHS
7175-T7362
Hand Forging
SECTION B-B

STRUCTURAL MODIFICATIONS IDENTIFIED BY THE DEVELOPMENT TEST PROGRAM

DECRIPTION	DURABILITY ARTICLE
* FRAME 252 R.H. CRACK AT 11.6 % LT	* REDESIGNED DIE FORGED FRAME
* FASTENERS CRACKED 22 % LT	* REPLACED WITH LARGER SIZE FASTENERS
* CRACK IN WING UPPER SKIN AT WS78 L/H AT 78%LT	* WOULD INSTALL NEW SKIN AT 2ND LT TESTING
* FRONT SPAR CRACK WS 96 R/H AT 103 % LT	* EXTERNAL STEEL STRAPS
* REAR SPAR FAILURE L/H AT 186% LT	* COLD WORK AT 2ND LT TESTING

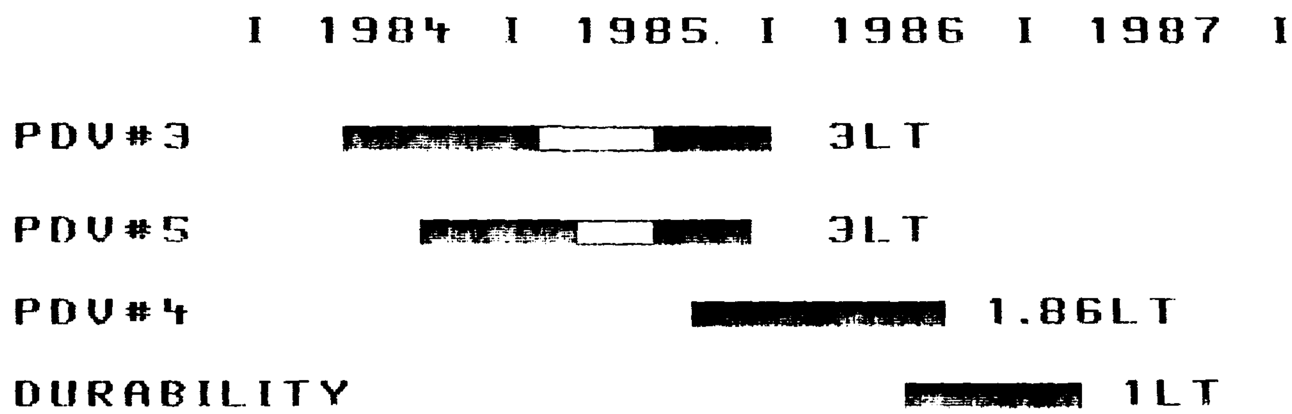
FULL SCALE DURABILITY TEST

* TEST STARTED	JULY 1986
* FIRST LIFETIME (TEST STOPPED)	JAN 1987
* INSPECTIONS AND REPAIRS	JAN - MARCH 1987
* PROGRAM CONCLUDED	MARCH 1987

LESSONS LEARNED

* DESIGN SPECTRUM
* MATERIAL SELECTION/GEOMETRICAL CONFIGURATION
* CRACK GROWTH DATA
* FINITE ELEMENT MODEL
* CRACK GROWTH METHODOLOGY VALIDATION
* TEST SCHEDULE

DEVELOPMENT TEST SCHEDULE



CONCLUSION

INCORPORATION OF THE LESSONS LEARNED FOR FUTURE PROGRAMS WILL SIGNIFICANTLY REDUCE DURABILITY AND DAMAGE TOLERANCE TECHNICAL RISK

SESSION III: MECSIP/ENSIP

87-(MCS)#132/10EN-114-1

ASIP
CONF
BRIEFING

EXPANSION OF THE INTEGRITY PROCESS TO MECHANICAL SUBSYSTEMS AND EQUIPMENT

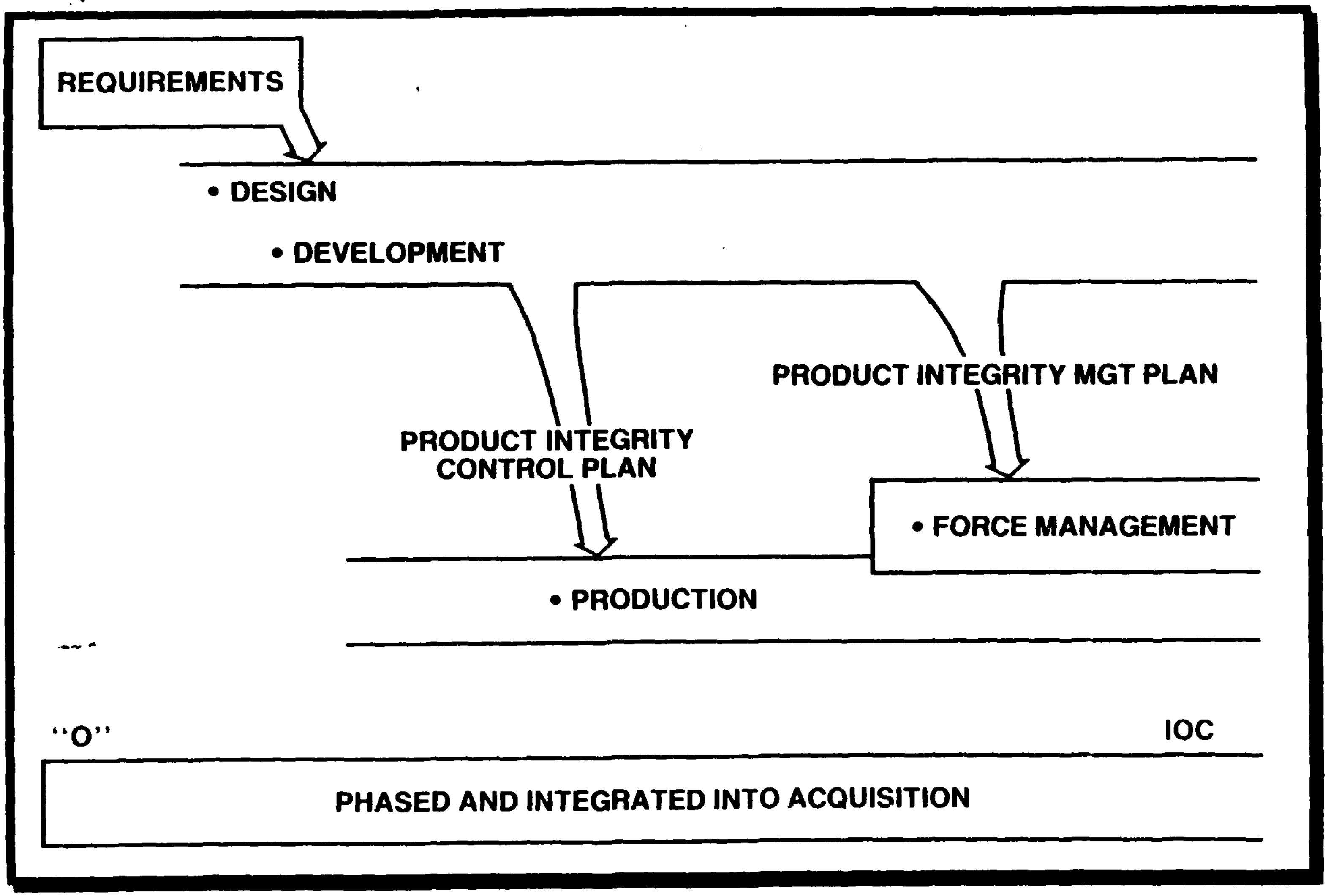
HOWARD A. WOOD
TECHNICAL ADVISOR
FLIGHT SYSTEMS ENGINEERING
AERONAUTICAL SYSTEMS DIVISION

THE "INTEGRITY" PROCESS

- AN ORGANIZED, DISCIPLINED AND TIME PHASED APPROACH TO DESIGN, DEVELOPMENT, QUALIFICATION, MANUFACTURE AND IN SERVICE MANAGEMENT OF A PRODUCT TO ACHIEVE
  - MISSION PERFORMANCE AND EFFECTIVENESS
  - SAFETY
  - DURABILITY
  - RELIABILITY
  - SUPPORTABILITY

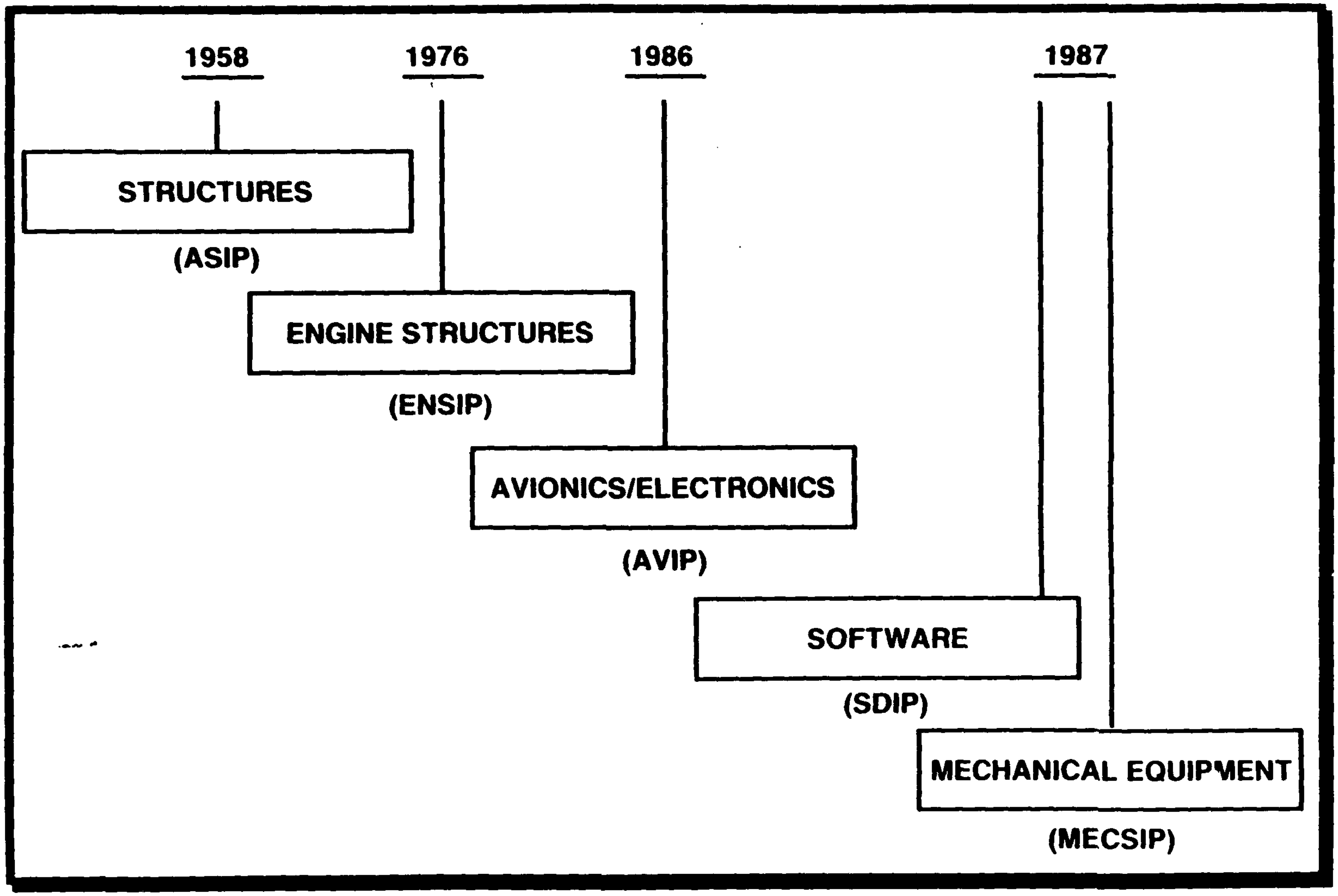
INTEG
DISK AA

INTEGRITY PROCESS FLOW



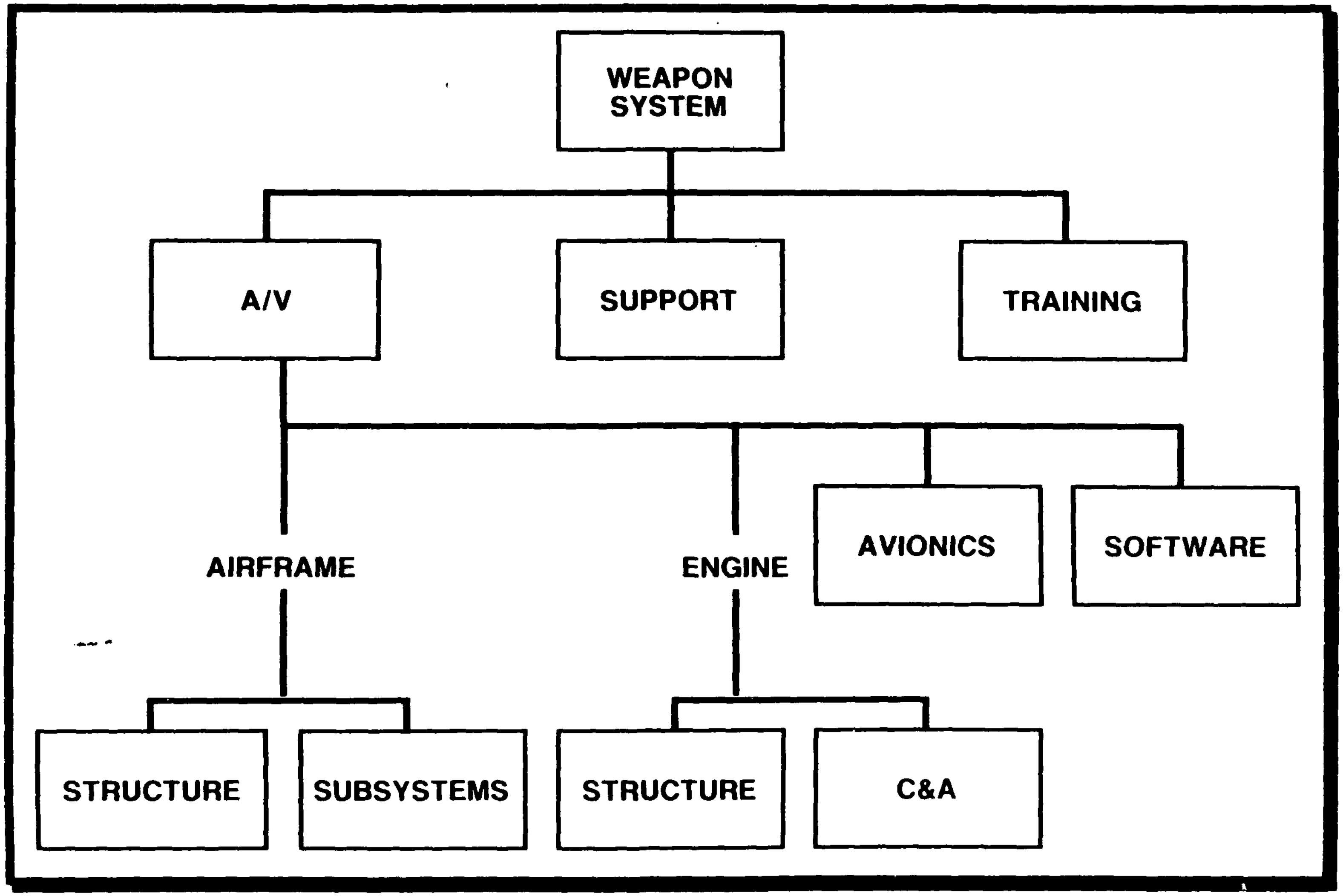
DISK /

APPLICATION OF THE PROCESS HAS EXPANDED



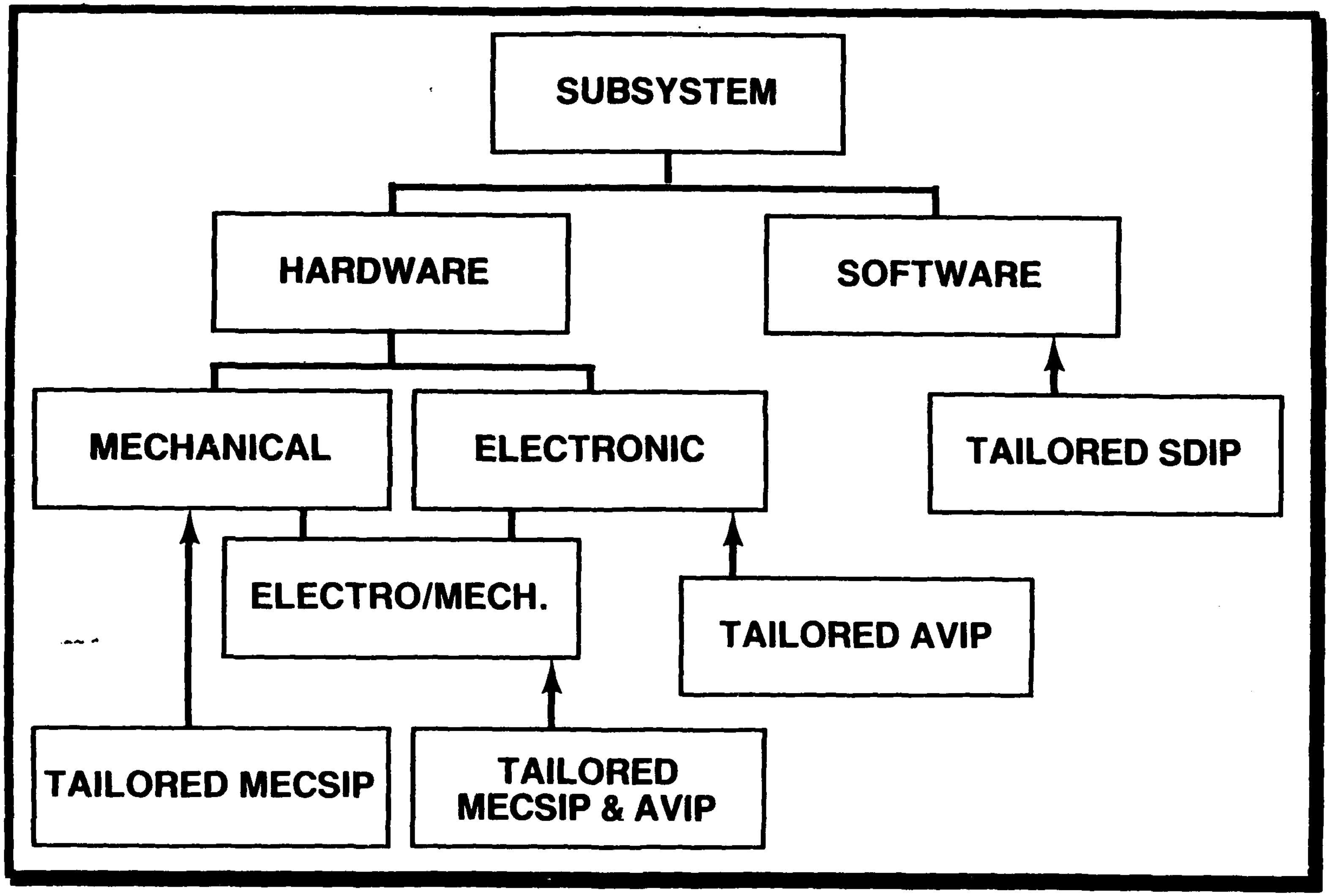
integ
DISK 88I

INTEGRITY PROCESS - THRUST AT ASD



SUB DISK C(

INTEGRITY PROCESS FOR SUBSYSTEMS

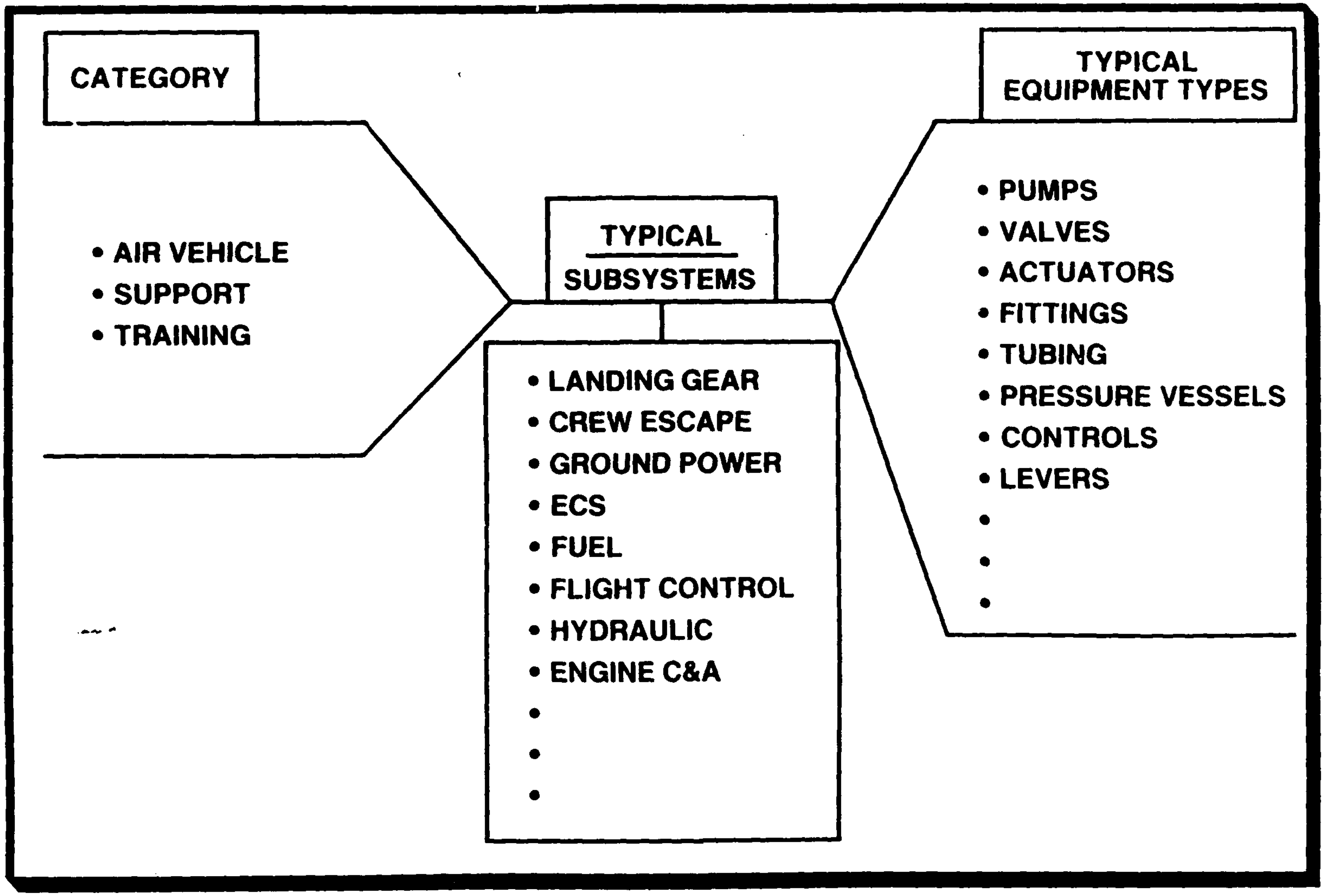


87-(MCS)#132/10EN-114-8

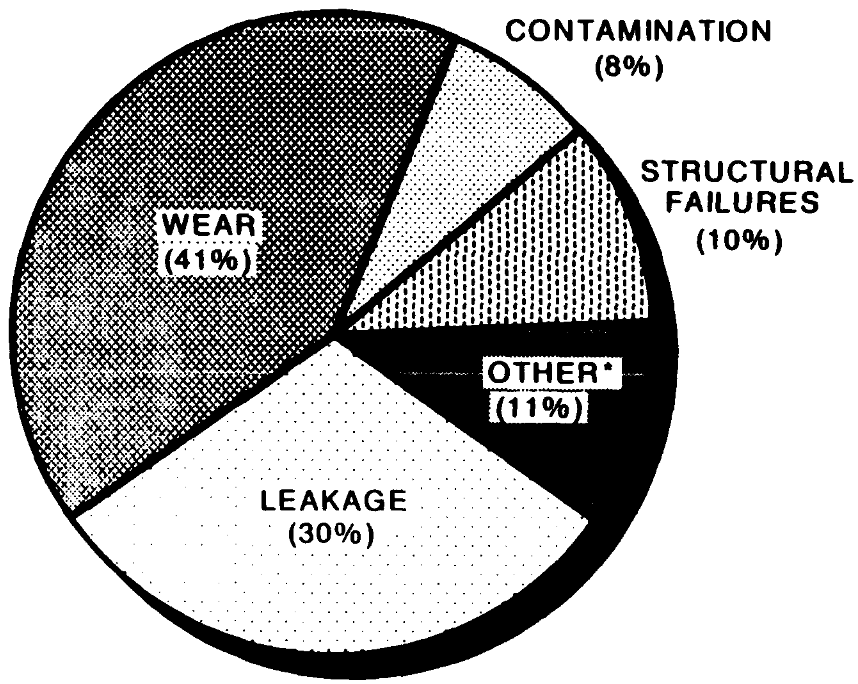
MECSIP - THE INTEGRITY PROCESS DEVELOPED SPECIFICALLY TO ADDRESS MECHANICAL EQUIPMENT AND MECHANICAL SUBSYSTEMS

mecsip1
DISK AAA

MECSIP



MECSIP
EXAMPLE-TYPICAL FAILURE CAUSES



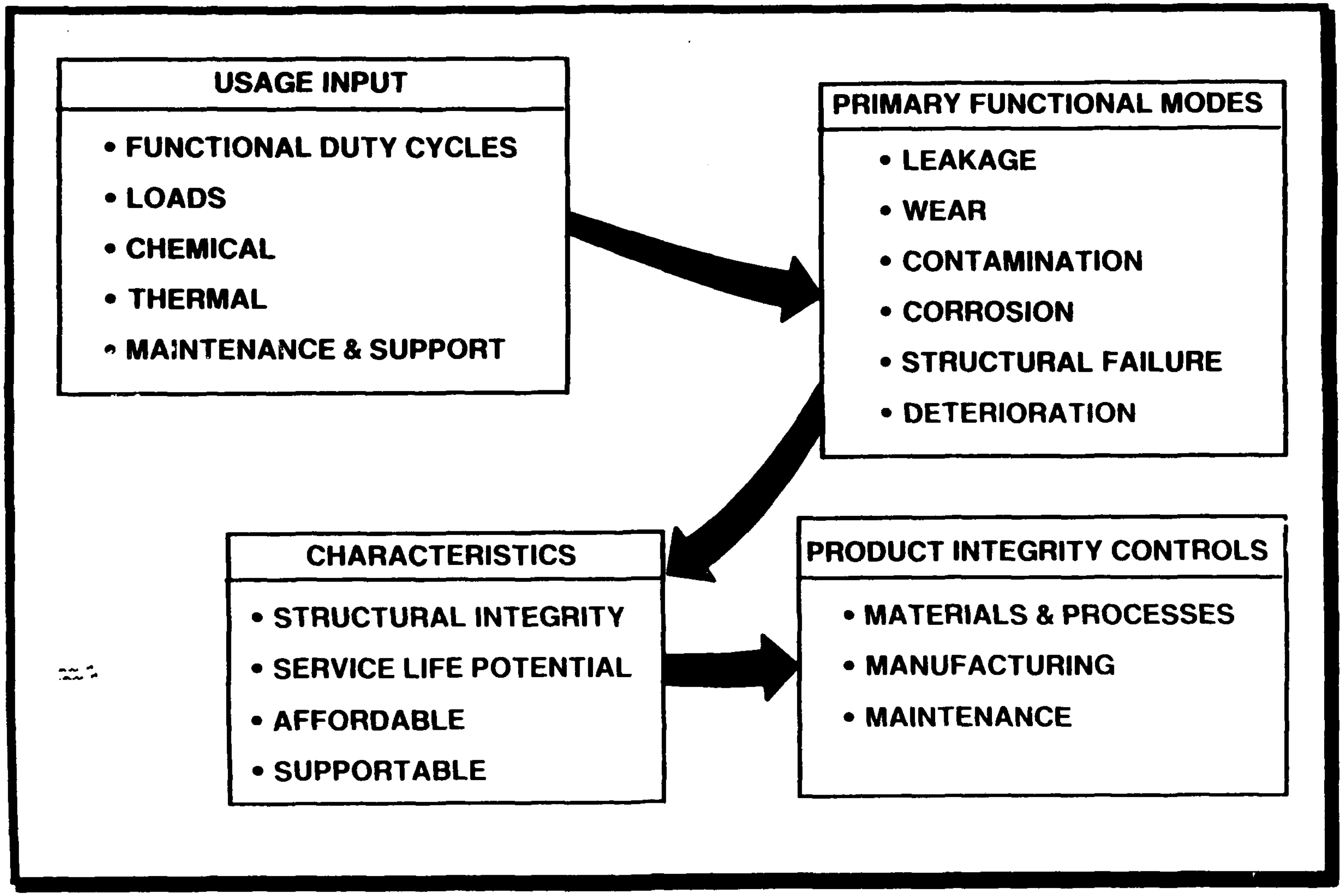
* – MAINTENANCE
– MANUFACTURING
RELATED

REF: F100 – MAIN
FUEL PUMP – (F – 15)

8710EN-0114-47 DISK 9

DISK

MECSIP SCOPE



MECSIP
GENERAL APPROACH

- DISCIPLINED SYSTEMS ENGINEERING APPROACH - DESIGN/ DEVELOPMENT - EMPHASIS ON DETERMINING AND UNDERSTANDING FAILURE PROCESSES AND CONSEQUENCES ON OPERATIONAL PERFORMANCE
- UNDERSTAND OPERATIONAL ENVIRONMENT, SUPPORT NEEDS - DEVELOP REQUIREMENT AND SUBSYSTEM / EQUIPMENT CHARACTERISTICS DIRECTED TOWARD THESE NEEDS
- EARLY TRADE STUDIES - REQUIRED OPERATIONAL SERVICE PERIOD. MAINTENANCE FREE OPERATION.
- UNDERSTAND MATERIALS, MANUFACTURING PROCESSING, PRODUCIBILITY ISSUES - INCORPORATE INTO DESIGN PROCESS
- DESIGN FOR FINITE "LIFE" BASED ON USAGE

MECSIP
GENERAL APPROACH

- CONDUCT COMPREHENSIVE DEVELOPMENT TEST PROGRAM - "BUILDING BLOCK CONCEPT"
- SCHEDULE TESTS TO ASSURE THAT FINDINGS ARE INCORPORATED INTO DESIGN IN ADVANCE OF SIGNIFICANT PRODUCTION COMMITMENTS OR EXPENDITURES
- CONTROL MANUFACTURING PROCESS
- DEVELOP FORCE MANAGEMENT REQUIREMENTS FROM DEVELOPMENT PROGRAM FINDINGS
- ESTABLISH TRACKING NEEDS SERVICE USAGE
- DEVELOP AIR FORCE PROGRAM TO ACCOMPLISH FORCE MANAGEMENT

87-(MCS)#132/10EN-114-12

MECSIP
APPROACH - REQUIREMENTS

- TASK - ESTABLISH DESIGN REQUIREMENTS FOR SUBSYSTEMS AND EQUIPMENT

OBJECTIVE

- CONVEY SYSTEM OPERATIONAL AND SUPPORT NEEDS TO REQUIREMENTS DOCUMENTS

INCLUDE:

- FUNCTIONAL DUTY CYCLE & USAGE
- OPERATIONAL ENVIRONMENTS
- REQUIRED OPERATIONAL SERVICE PERIOD
- MAINTENANCE & SUPPORT
- STRENGTH, DURABILITY, DAMAGE TOLERANCE, ETC.

MECSIP PROCESS

MIL-PRIME SPECIFICATION - "REQUIREMENTS FOR THE INTEGRITY OF MECHANICAL EQUIPMENT AND SUBSYSTEMS"

OUTLINE

- SERVICE LIFE, USAGE AND ENVIRONMENTS
  - REQUIRED OPERATIONAL SERVICE LIFE
  - FLIGHT HOURS
  - GROUND RUN HOURS
  - OPERATIONAL EVENTS/CYCLES
  - MAINTENANCE/REPAIR/TEST CYCLES
  - AIR VEHICLE ENVELOPE
  - GROUND VEHICLE ENVELOPE
  - OTHER ENVELOPE
  - EXTERNAL AND INSTALLATION ENVIRONMENT
  - INTERNAL ENVIRONMENT
  - LOGISTICS, MAINTENANCE/STORAGE/TRANSPORTATION ENVIRONMENT

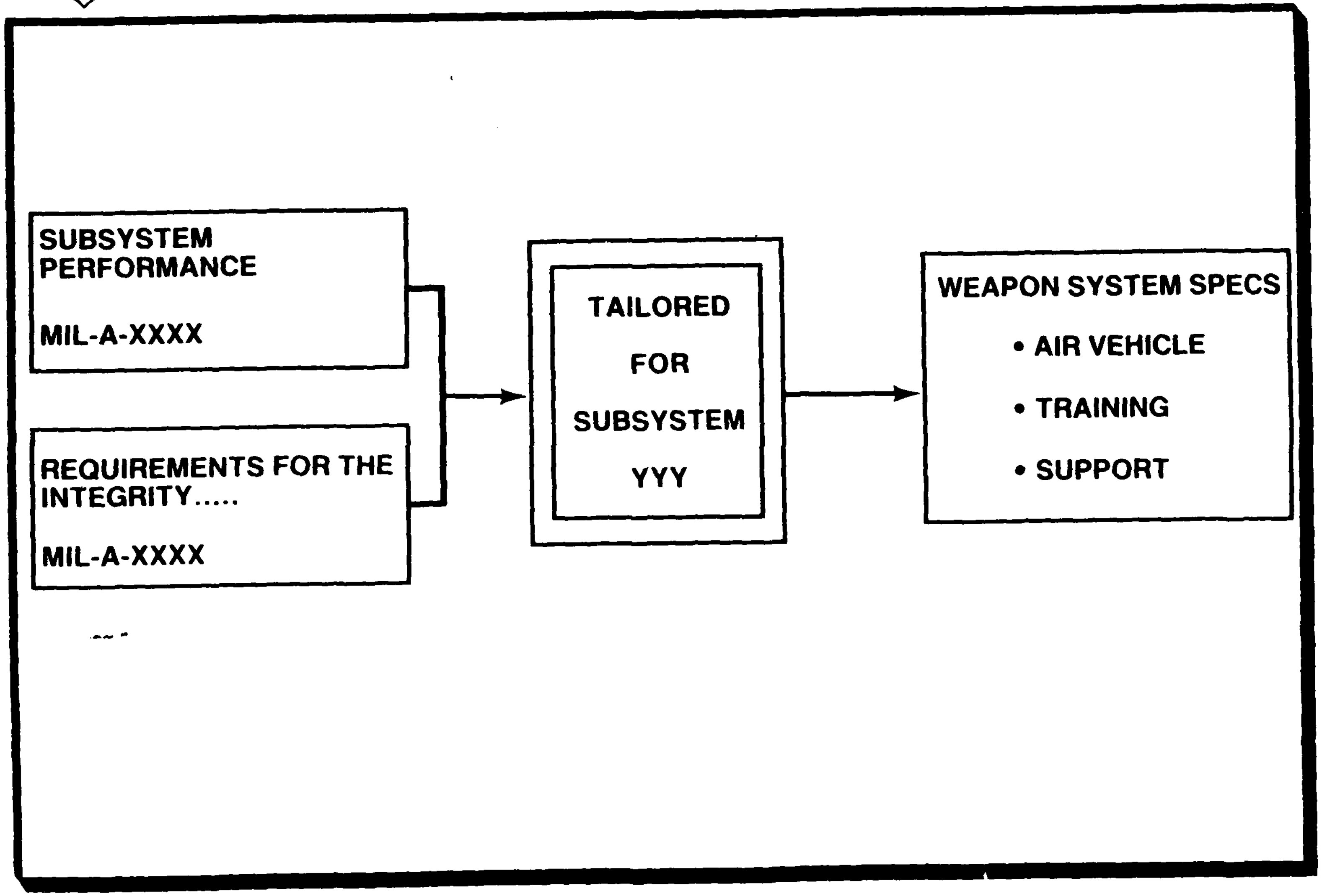
MECSIP PROCESS

OUTLINE (CONT'D)

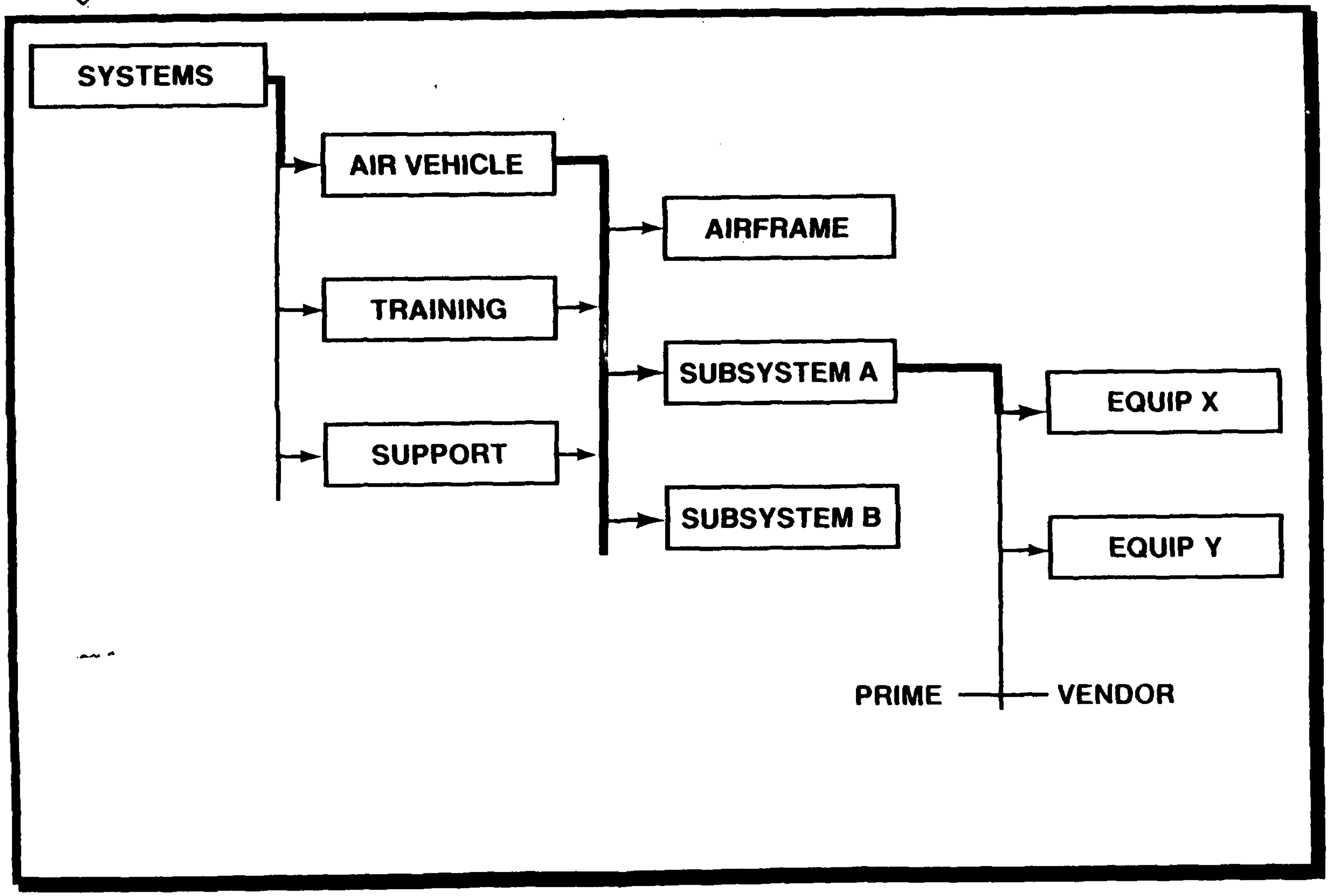
- MATERIALS AND PROCESSES
- DURABILITY, ECONOMIC LIFE
  - CORROSION/MATERIAL DEGRADATION RESISTANCE
  - WEAR/DETERIORATION
  - FATIGUE CRACKING/DAMAGE DEVELOPMENT
  - OTHER
- DAMAGE TOLERANCE
- STRENGTH
- VIBRATION/DYNAMIC RESPONSE
- PROVISIONS FOR INTEGRITY MANAGEMENT
  - MAINTAINABILITY
  - INSPECTABILITY
  - SUBSYSTEM INSTALLATION
- MANPOWER, PERSONNEL, TRAINING
- VERIFICATION
- HANDBOOK

DISK AAA

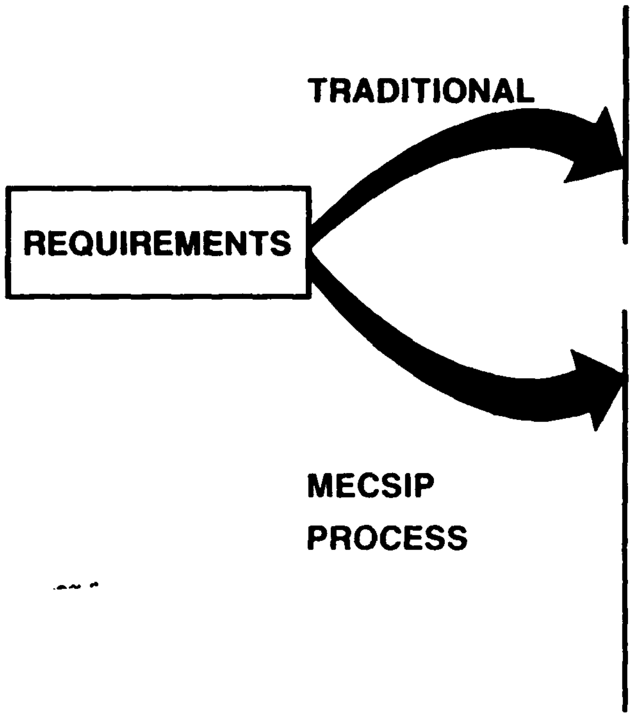
SPECIFICATION DEVELOPMENT



SPECIFICATION DEVELOPMENT



MECSIP



- "COOKBOOK" QUALIFICATION
- NO UNIFORMITY

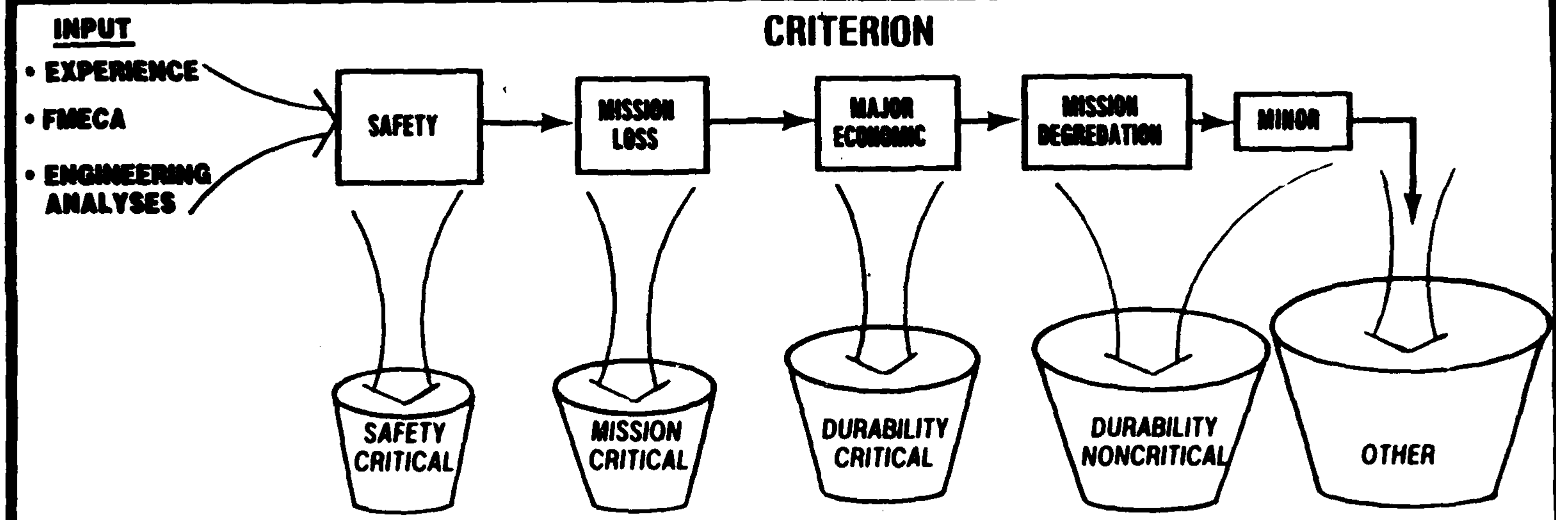
- REALISTIC BASIS
- EXPECTED SERVICE USAGE
- SUPPORTABILITY/MAINTAINABILITY
- USER/OPERATIONAL, REQMTS - TRANSFORMED

MECSIP

APPROACH - DESIGN - DEVELOPMENT - PRODUCTION

- DISCIPLINED PROCESS APPLIED
  - DESIGN
  - ANALYSES
  - TESTS
- CONTROLS IMPOSED ON MATERIALS, PROCESSES, MANUFACTURING
- FOCUS AND DIRECTION - PARTS CLASSIFICATION

COMPONENT CLASSIFICATION DECISION LOGIC



CLASSIFICATION CATEGORIES	CANDIDATE SPECIAL PROVISIONS*						
	EXPANDED DESIGN CRITERIA	TIGHTENED PROCESS CONTROL	ENHANCED QUALITY CONTROL	ACCEPTANCE TESTING	RESTRICTED REPROCUREMENT	PARTS TRACKING	PREVENTATIVE MAINTENANCE
SAFETY CRITICAL	●	●	●	●	●	●	●
MISSION CRITICAL	●	●	●	●	●	●	●
DURABILITY CRITICAL		●	●	●	●	●	●
DURABILITY NONCRITICAL				●			●
OTHER							

* THE EXTENT TO WHICH THE SPECIAL PROVISIONS ARE APPLIED IS COMPONENT SPECIFIC

MECHANICAL EQUIPMENT AND SUBSYSTEMS (MECSIP) TASKS

TASK I	TASK II	TASK III	TASK IV
PRELIMINARY PLANNING AND EVALUATION	DESIGN INFORMATION	DESIGN ANALYSES AND DEVELOPMENT TESTS	COMPONENT DEV AND SYS FUNCTIONAL TESTS
• PROGRAM STRATEGY	• MECSIP MASTER PLAN	• LOAD ANALYSES	• FUNCTIONAL QUAL TEST
• TRADE STUDIES	• DESIGN SERVICE LIFE/ DESIGN USAGE	• DESIGN STRESS/ ENVIRONMENTAL SPECTRA DEV	• STRENGTH TESTING
• DEVELOPMENT & REFINEMENT OF REQUIREMENTS	• CRITICAL PARTS ANALYSES & CLASS	• PERFORMANCE & FUNCTIONAL SIZING AND ANALYSIS	• DURABILITY TESTING
• PRELIMINARY INTEGRITY ANALYSIS	• DESIGN CRITERIA	• THERMAL/ENVIRONMENT ANALYSIS	• VIBRATION DYNAMICS/ ACOUSTIC TESTING
	• M&P SELECTION/ CHARACTERIZATION	• STRESS/STRENGTH ANALYSIS	• DAMAGE TOLERANCE TESTS
	• PRODUCT INTEGRITY CONTROL PLAN	• DURABILITY ANALYSIS	• THERMAL & ENVIRONMENT SURVEY
	• CORROSION PREVENTION & CONTROL PLAN	• DAMAGE TOLERANCE ANALYSIS	• MAINTAINABILITY/ REPAIRABILITY DEMO
		• VIBRATION/DYNAMICS/ ACOUSTICS ANALYSIS	• EVAL AND INTERPRET OF TEST RESULTS
		• MATERIAL CHARACTERIZATION TEST	• INTEGRATED TEST PLAN
		• DESIGN DEVELOPMENT TEST	

MECHANICAL EQUIPMENT AND SUBSYSTEMS (MECSIP) TASKS (CONT'D)

TASK V

INTEGRITY MANAGEMENT DATA PACKAGE

- UPDATED ANALYSES
- MAINTENANCE PLANNING AND TASK DEVELOPMENT
- INDIVIDUAL SYSTEMS TRACKING

TASK VI

INTEGRITY MANAGEMENT

- OPERATIONAL USAGE
- MAINTENANCE RECORDS SERVICE REPORTING
- INDIVIDUAL SUBSYSTEM MAINTENANCE TIME

87-(MCS)#132/10EN-114-16

MECSIP
APPROACH - DURABILITY

. . . ABILITY OF THE SUBSYSTEM / COMPONENT TO RESIST DETERIORATION, WEAR, CRACKING, CORROSION, THERMAL DEGRADATION, ETC.. . . FOR A SPECIFIED SERVICE USAGE PERIOD. . .

REQUIREMENT . . .SHALL BE DURABLE AND ECONOMICALLY MAINTAINABLE THROUGHOUT OPERATIONAL SERVICE . . .

MECSIP
APPROACH - DURABILITY

- <u>ANALYSIS</u> - PREDICT EXPECTED OPERATIONAL SERVICE PERIOD AVAILABLE WITH AND WITHOUT SCHEDULED MAINTENANCE
  - MAJOR FAILURE MODES
  - FUNCTIONAL LIMITS
  - MATERIAL VARIABILITY
  - MANUFACTURING QUALITY
  - DURABILITY MARGIN BASED ON PARTS CLASSIFICATION
- <u>DESIGN</u>
  - DISCIPLINED DETAIL DESIGN PRACTICES - EXPERIENCE DATA BASE UTILIZED
  - MATERIAL SELECTION
  - CONTROL OPERATING STRESSES

MECSIP
APPROACH - DURABILITY

- TESTING - BUILDING BLOCK APPROACH
  - MATERIALS CHARACTERIZATION TESTS
  - DESIGN DEVELOPMENT
    - JOINTS
    - FITTING
    - SEALING CONCEPTS
    - CONTROLS
    - MECHANISMS
    - ETC.
  - SUBSYSTEM / COMPONENTS
- PURPOSE
  - IDENTIFY "HOT SPOTS" IN DESIGN
  - VERIFY REQUIREMENTS
  - ESTABLISH MAINTENANCE REQ'MTS

MECSIP

APPROACH - DURABILITY

TESTING - SUBSYSTEM / COMPONENTS - GOALS

- <u>TESTING</u>
  - TEST ARTICLES - REPRESENT PRODUCTION CONFIGURATION
  - TEST LOADS / ENVIRONMENTS - REPRESENT DESIGN USAGE
  - TEST DURATION - EXCESS OF REQUIRED OPERATION SERVICE LIFE - MARGIN
  - SCHEDULE - ONE LIFE TIME PRIOR TO MAJOR PRODUCTION COMMITMENTS (E.G., LONG LEAD TIME ON EXPENSIVE COMPONENTS)

MECSIP
APPROACH - DURABILITY

- **<u>TESTING</u>**
  - THOROUGH POST TEST TEARDOWN
  - DISPOSITION OF EVERY FINDING
  - EVALUATION BASED ON POTENTIAL IMPACT DURING OPERATION - ECONOMIC, PERFORMANCE, DOWNTIME, MAINTENANCE DEMAND, ETC.
    - REDESIGN
    - ACCEPT FINDING, ESTABLISH MAINTENANCE ACTION

MECSIP
APPROACH - DAMAGE TOLERANCE (FAILURE TOLERANCE)

. . . ABILITY OF THE SUBSYSTEM / COMPONENT TO RESIST FAILURE OR LOSS OF FUNCTION . . . OPERATE SAFELY IN PRESENCE OF DEFECTS / DAMAGE . . . FOR A SPECIFIED SERVICE USAGE PERIOD.

REQUIREMENT: PROVIDE MARGIN AND TOLERANCE FOR POTENTIAL OCCURANCE OF

- MANUFACTURING DEFECTS
- MAINTENANCE DAMAGE
- SERVICE INDUCED DAMAGE AND FAILURES

MECSIP
APPROACH - DAMAGE TOLERANCE

DESIGN BASIS

- FLAW TOLERANCE - CONSISTENT WITH STRUCTURES
- SYSTEM REDUNDANCY
- LEAK BEFORE BREAK

ANALYSIS AND TEST

- PREDICT RESIDUAL STRENGTH / FUNCTIONAL CAPACITY THROUGHOUT SAFE OPERATIONAL PERIOD
- PREDICT REQUIRED INSPECTION INTERVAL
- TESTING TO SUPPORT ANALYSIS - VERIFY REQUIREMENTS

MECSIP
APPROACH - INTEGRITY CONTROL

- **PRODUCT INTEGRITY CONTROL PLAN**

  . . . DEFINES SPECIAL CONTROLS TO ENSURE THAT INTEGRITY IS MAINTAINED THROUGHOUT PRODUCTION . . .

  - CRITICAL PARTS LIST
  - MATERIAL, QUALITY, MANUFACTURING SPECIFICATIONS
  - VENDOR & SUPPLIER CONTROLS
  - MATERIAL / PART TRACEABILITY
  - SPECIAL INSPECTIONS REQUIRED

MECSIP PROCESS

PROCESS
- DESIGN
- DEVELOPMENT
- PRODUCTION

TRADITIONAL

- INFORMAL APPROACH, "COOKBOOK" QUALIFICATION
- LIMITED SCOPE TEST VERIFICATION

MECSIP PROCESS

- FORMALIZATION OF PROCESS
- MASTER PLAN
- PHASING/SCHEDULING OF TASKS/SUCCESS CRITERIA
- EMPHASIS ON EVALUATION OF DEVELOPMENT DATA
- FOCUS ON CRITICAL PARTS (SAFETY, MISSION, DURABILITY)
- PRODUCTION/MANUFACTURING - PRODUCT INTEGRITY CONTROLS
- DEVELOPMENT OF FORCE MANAGEMENT DATA PACKAGE

MECSIP

APPROACH - IN SERVICE INTEGRITY MANAGEMENT DATA

- REQUIREMENT

  . . . SUBSYSTEM / EQUIPMENT SHALL BE MAINTAINABLE AND INSPECTABLE OVER THE REQUIRED OPERATIONAL LIFE . . .

- INTEGRITY MANAGEMENT DATA PACKAGE
  - MAINTENANCE PLANNING
  - REPAIRS DEVELOPED AND DEMONSTRATED
  - INSPECTION / TECHNIQUES & TIMES DEVELOPED
  - CRITICAL COMPONENT TRACKING REQUIREMENTS
- DATA BASE - DEVELOPMENT PROGRAM FINDINGS

MECSIP

APPROACH - IN SERVICE INTEGRITY MANAGEMENT DATA

TRACKING OF INDIVIDUAL COMPONENTS

- REQUIRED FOR CRITICAL COMPONENTS SUPPORTED BY PREVENTIVE MAINTENANCE
- SPECIFIC APPROACH NOT SPECIFIED
- SERIALIZATION NEEDS ESTABLISHED

MECSIP
APPROACH - IN SERVICE INTEGRITY MANAGEMENT

- AIR FORCE RESPONSIBILITY
- PROGRAM TO OBTAIN USAGE DATA AS REQUIRED
- CONDUCT INDIVIDUAL COMPONENT TRACKING AS REQUIRED

MECSIP PROCESS

PROCESS

- FORCE MANAGEMENT

TRADITIONAL

- NO FORMAL PROCESS

MECSIP PROCESS

- RECOGNITION OF NEED
- USER INVOLVEMENT
- DEVELOPMENT OF FORCE MGT. DATA PACKAGE
- CONCEPT BASED ON USER NEEDS AND EARLY TRADES
- USAGE AND TRACKING REQUIREMENTS DEFINED
- IMPLEMENTATION PLAN - USER RESPONSIBILITIES AND SCHEDULED TASKS
- CONTROL OF MAINTENANCE

SUMMARY

- THE INTEGRITY APPROACH HAS BEEN FORMULATED FOR MECHANICAL SYSTEMS AND EQUIPMENT
- WE KNOW THE PROCESS WORKS AND IS EFFECTIVE
- DOCUMENTATION COMPLETED AND UNDER FINAL REVIEW
- INDUSTRY INPUT (AIA) FORTHCOMING
- PUBLICATION – LATE 1987

RELIABILITY/INTEGRITY

- RELIABILITY – CONSEQUENCE OF THE "DESIGN"
- "DESIGN" – CONSEQUENCE OF THE INTEGRITY PROCESS
- AT ASD – INTEGRITY PROCESS RECOGNIZED AS THE VEHICLE TO ACHIEVE RELIABILITY FOR SYSTEMS, SUBSYSTEMS AND EQUIPMENT

CONCLUSION

- AT ASD WE HAVE ADOPTED THE INTEGRITY PROCESS AS A KEY VEHICLE TO ACHIEVE PERFORMANCE, RELIABILITY AND SUPPORTABILITY FOR OUR SYSTEMS, SUBSYSTEMS, AND EQUIPMENT.

- THE INTEGRITY PROCESS IS OUR PRIMARY INSTRUMENT FOR ACHIEVING R&M 2000 OBJECTIVES.

A REVIEW OF THE QUALITY OF SCREW THREADED PRODUCTS

C. L. PETRIN, JR.
ASD/ENFS

NARRATIVE

OUTLINE

THIS IS THE ORDER OF PRESENTATION. I WILL GIVE YOU SOME OF THE BACKGROUND WHICH LED TO THE FORMATION OF A WORKING GROUP, WHAT THE WORKING GROUP FINDINGS WERE, THE RECOMMENDED REVISED POLICY FOR ENSURING THE QUALITY OF SCREW THREADED PRODUCTS, AND SOME OF THE IMPLEMENTING ACTIONS IDENTIFIED.

NARRATIVE

A REVIEW OF THE QUALITY OF SCREW THREADED PRODUCTS WAS CONDUCTED DURING THE PERIOD MAR - OCT 1987. RECOMMENDATIONS TO IMPROVE THE QUALITY OF THESE PRODUCTS WERE DEVELOPED AND HAVE BEEN PRESENTED TO HQ AFLC, HQ AFSC, AND HQ USAF/LE AND SAF/AQ. THE RESULTING POLICY IS PRESENTED.

OBJECTIVE

- PRESENT EVENTS LEADING TO USAF POLICY ON SCREW THREADED POLICY
- PRESENT SALIENT REQUIREMENTS OF THE POLICY

OUTLINE

- BACKGROUND
- WORKING GROUP
- WORKING GROUP FINDINGS
- REVISED POLICY
- DEFENSE LOGISTICS AGENCY TEST RESULTS
- IMPLICATIONS
- SOME IMPLEMENTATION ACTIONS
- SUMMARY

NARRATIVE

BACKGROUND

THIS REVIEW WAS INITIATED ABOUT 15 MONTHS AGO WHEN HQ USAF/LE-RD CIRCULATED A DRAFT POLICY PROPOSAL ON THREAD GEOMETRY. THIS POLICY AFFECTED ALL CLASS 3 FIT THREADS, BOTH INTERNAL AND EXTERNAL. THERE ARE TWO SPECIFICATIONS DEFINING CLASS 3 THREAD GEOMETRY, MIL-S-7742 AND MIL-S-8879. THE DRAFT POLICY WOULD HAVE UPGRADED THE INSPECTION REQUIREMENTS FOR INTERNAL THREADS AND RAISED THE INSPECTION SAMPLING RATES FOR BOTH INTERNAL AND EXTERNAL THREADS. OUR REVIEW RAISED TWO CONCERNS. FIRST, THAT ALTHOUGH THERE WERE PERIODIC MATERIAL AND PROCESS PROBLEMS IN FASTENERS, THAT WE HAD NO AWARENESS FROM OUR DEVELOPMENT AND IN-SERVICE EXPERIENCE THAT SCREW THREAD GEOMETRY WAS A PROBLEM. SECOND, THAT THE INSPECTION OF SAMPLING RATES WERE SET SO HIGH THAT THEY WERE TANTAMOUNT TO A 100% INSPECTION OF ALL FASTENERS AND OTHER SCREW THREADED PRODUCTS. THESE CONCERNS WERE DISCUSSED WITH USAF/LE-RD, HQ AFLC, HQ AFSC, AND AEROSPACE INDUSTRIES ASSOCIATION DURING OCT 86 THROUGH FEB 87 INCLUDING BRIEFINGS TO HQ USAF/LE-RD ON 21 NOV AND 2 FEB 87. THESE BRIEFINGS STRESSED THE IMPORTANCE OF CONSIDERING ALL THE ESSENTIAL CHARACTERISTICS OF A COMPONENT AND RECOMMENDED ESTABLISHMENT OF A TECHNICAL COMMITTEE TO REVIEW SCREW THREADED COMPONENTS TO IDENTIFY PROBLEMS AND MAKE APPROPRIATE RECOMMENDATIONS. ON 26 FEB 87, HQ USAF/LE-RD ISSUED A POLICY LETTER REQUIRING TIGHTER INSPECTION REQUIREMENTS FOR INTERNAL THREADS. THE TECHNICAL COMMITTEE WAS GIVEN A GO-AHEAD WITH A REQUIREMENT TO REPORT IN SEP 87. A JOINT TEAM CO-CHAIRED BY HQ AFLC AND ASD WITH PARTICIPATION BY DEFENSE LOGISTICS AGENCY AND SAN ANTONIO AND OKLAHOMA AIR LOGISTICS CENTERS WAS FORMED. THE TEAM MADE SEVERAL SITE VISITS DURING THE PERIOD MAR THROUGH JUL. FINDINGS AND RECOMMENDATIONS WERE PUT INTO A BRIEFING WHICH HAS BEEN PRESENTED TO THE COMMANDERS OF AFLC, AFSC, THE DEPUTY DIRECTOR DEFENSE LOGISTICS AGENCY, HQ USAF/LE-RD, HQ USAF/LE, AND SAF/AQ.

BACKGROUND

- USAF / LE - RD (22 SEP 86)

  PRELIMINARY POLICY PROPOSAL FOR SCREW THREADED COMPONENTS

  "THREAD DIMENSIONS"

- HQ USAF, AFSC / AFLC & DEFENSE LOGISTICS AGENCY DISCUSSIONS (21 NOV 86)
  - MULTIPLE ESSENTIAL CHARACTERISTICS FOR SCREW THREADED COMPONENTS
  - TECHNICAL COMMITTEE: ADEQUACY OF SPECs / PROCUREMENT PRACTICES

- USAF / LE - RD POLICY LETTER (26 FEB 87)
  - TIGHTEN CONFORMANCE REQUIREMENTS TO DIMENSIONAL CHARACTERISTICS IN MIL-S-8879 AND MIL-S-7742 EFFECTIVE FY 88 PROCUREMENTS
  - ESTABLISH AFSC / AFLC TECHNICAL COMMITTEE TO ASSESS SPEC REQUIREMENTS AND CURRENT PROCUREMENT PRACTICES TO ASSURE BALANCED EMPHASIS ON ALL ESSENTIAL CHARACTERISTICS
  - TEAM REPORT - SEP 87

- STATUS REPORT - MAY 87

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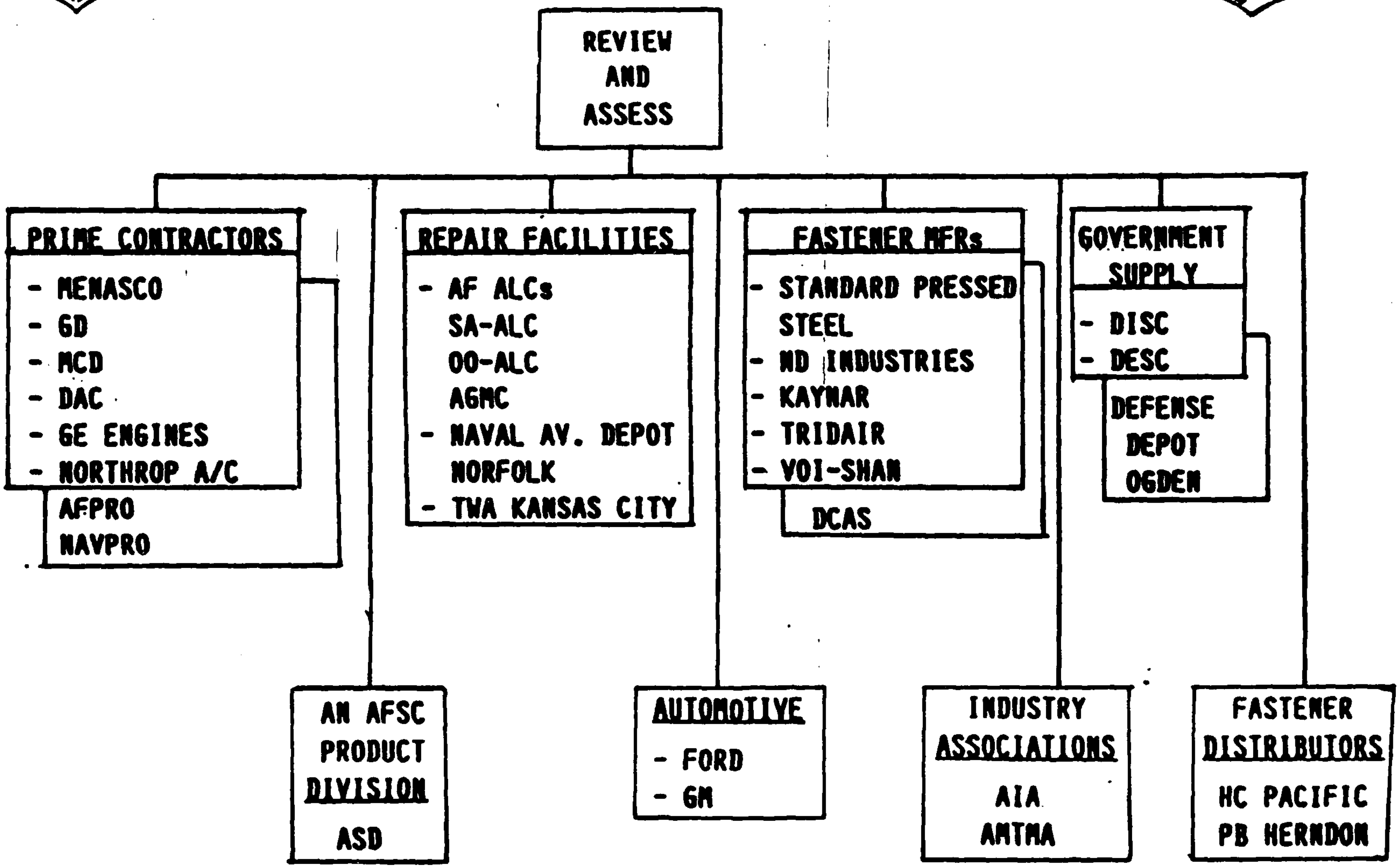
NARRATIVE

MANY SOURCES OF INFO/INPUT WERE USED

THIS CHART WILL GIVE YOU SOME INSIGHT INTO THE SCOPE OF THE REVIEW. WE CONTACTED SEVERAL PRIVATE SECTOR COMPANIES IN AND OUT OF AEROSPACE TO DETERMINE THEIR PROBLEMS AND IF THEIR PROCUREMENT AND QUALITY ASSURANCE PRACTICES OFFERED A BETTER WAY FOR THE GOVERNMENT TO OPERATE.

MANY SOURCES OF INFO / INPUTS WERE USED!



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NARRATIVE

TECHNICAL TEAM FINDINGS

THIS SECTION WILL COVER THE TEAM FINDINGS.

TECHNICAL TEAM FINDINGS

NARRATIVE

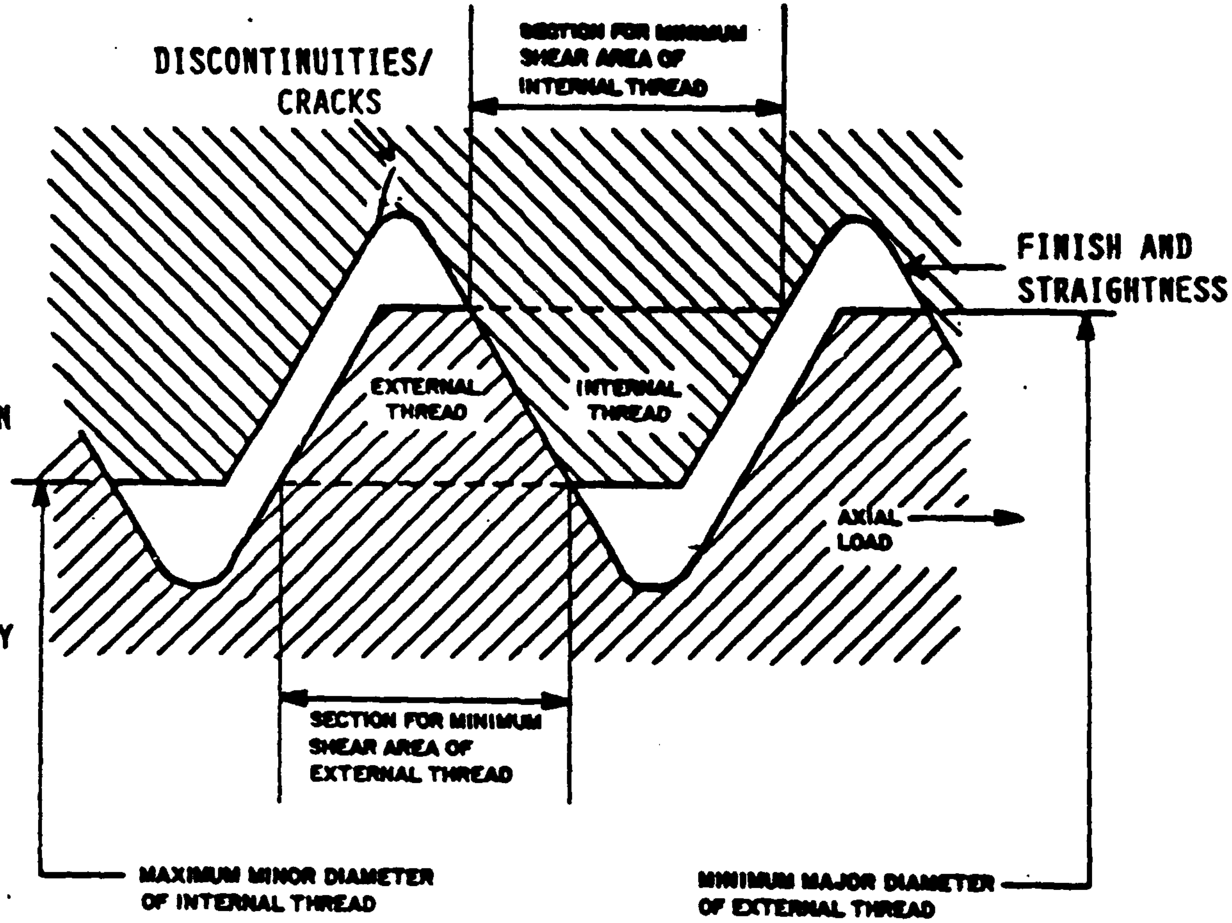
THERE ARE MANY CHARACTERISTICS TO MAKE THREADS WORK

ALTHOUGH THE PROPOSED POLICY HAD ADDRESSED THREAD GEOMETRY, THE FIRST CONCLUSION REACHED WAS THAT MANY OTHER CHARACTERISTICS WERE ESSENTIAL TO GET THE THREAD JOINT TO WORK PROPERLY. IT WOULD BE OF LITTLE USE TO HAVE A PERFECT SHAPE IF THE MATERIAL STRENGTH WAS LOW OR ROOT CRACKS INDUCED FATIGUE FAILURES AND SO FORTH. ANY POLICY ON QUALITY OF THREADED PRODUCTS SHOULD BE A BALANCED POLICY ON QUALITY ADDRESSING ALL ESSENTIAL CHARACTERISTICS.

THERE ARE MANY CHARACTERISTICS

TO MAKE THREADS WORK

HARDNESS
TENSILE STRENGTH
SHEAR STRENGTH
FATIGUE
CARBURIZATION/
  NITROGENIZATION
WORK EFFECT
STRUCTURE/
  GRAIN FLOW
BURNS
STRESS DURABILITY
CORROSION



THREAD DIMENSIONS

NARRATIVE

REPRESENTATIVE BOLT SPECIFICATION

THESE ARE THE REQUIREMENTS FOR A TYPICAL BOLT. WHAT WE NOTED HERE WAS THAT THREAD GEOMETRY WAS ONLY ONE OF THESE CHARACTERISTICS. WE NOTED ALSO THAT WHILE SOME OF THESE CHARACTERISTICS WERE UNIVERSALLY APPLIED TO ALL BOLTS, OTHERS WERE DESCRIBED ONLY IN THE INDIVIDUAL BOLT SPECS. THIS MEANT THAT IF CHANGES IN POLICY AFFECTED ALL ESSENTIAL CHARACTERISTICS, IMPLEMENTATION OF THE POLICY MIGHT REQUIRE REVISION OF EACH SPEC. THERE ARE ON THE ORDER OF 170,000 INDIVIDUAL CLASS 3 FASTENER SPECS.

REPRESENTATIVE BOLT SPECIFICATION

SEVERAL ESSENTIAL / CRITICAL CHARACTERISTICS

MIL-B-8907A* BOLT, ALLOY STEEL, SHEAR, AND TENSILE (156 KSI F AND 260 KSI F, 450^{o}F), EXTERNAL WRENCHING, FLANGED HEAD

ATTRIBUTES	INSPECTION/TEST PROCEDURES**
THREADS	MIL-S-8879/7742B
HARDNESS	MIL-S-1312, TEST 6
FINISH/STRAIGHTNESS***	MIL-C-8837 & MIL-B-8907A
DISCONTINUITIES	MIL-I-6868
TENSILE STRENGTH	MIL-STD-1312, TEST 8
SHEAR STRENGTH	MIL-STD-1312, TEST 13
FATIGUE	MIL-STD-1312, TEST 11
CARBURIZATION/NITROGENIZATION	MIL-B-8907A
WORK EFFECT	MIL-B-8907A
STRUCTURE/GRAIN FLOW	MIL-B-8907A
BURNS	MIL-B-8907A
STRESS DURABILITY***	MIL-B-8907A

*NAVY SPEC. **SPECIFIES QUALITY REQUIREMENTS/LEVELS ***CORROSION CONTROL

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NARRATIVE

ESSENTIAL/CRITICAL PART CHARACTERISTICS

AS I HAVE MENTIONED, THE REVIEW BROUGHT OUT THAT THESE SPECS WERE ESSENTIALLY WRITTEN AROUND THE PERSPECTIVE OF THE THREAD MAKER. IT DID NOT MATTER IF THE THREADED JOINT WAS TO HOLD A WING TO THE FUSELAGE OR A TOILET SEAT TO THE BOWL, THE REQUIREMENTS WERE THE SAME. IT WAS THE TEAM'S CONCLUSION THAT THE ESSENTIAL CHARACTERISTICS AND THE QUALITY REQUIREMENTS SHOULD BE A FUNCTION OF THE INTENDED APPLICATION. THIS WOULD REQUIRE SOME CONSCIOUS CHOICES ON THE PART OF DESIGNERS AND COULD MEAN A GIVEN STANDARD FASTENER COULD HAVE MORE THAN ONE INSPECTION REQUIREMENT BECAUSE OF VARYING DEGREES OF CRITICALITY IN ITS APPLICATIONS.

ESSENTIAL / CRITICAL PART CHARACTERISTICS

- MIL SPECs FOR "QUALIFIED PARTS" REPRESENT CONTROL FROM THE ORIGINAL EQUIPMENT (PART) MANUFACTURER'S (OEM'S) PERSPECTIVE
- APPLICATION OF THE PART ESTABLISHES THE:
  - ESSENTIAL CHARACTERISTICS
  - LEVEL OF INSPECTION
- ESSENTIAL CHARACTERISTICS CAN CHANGE WITH APPLICATION

  (STRENGTH, HARDNESS, CORROSION, CLOSENESS OF FIT)

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NARRATIVE

MIL-SPEC QUALIFIED PARTS PROGRAM

ANOTHER ASPECT OF THE REVIEW WAS THE STANDARD PARTS PROGRAM. AS SHOWN, THERE ARE SEVERAL DOCUMENTS WHICH ADDRESS THE QUALITY OF STANDARD PARTS. THE SUM RESULT OF ALL THESE REQUIREMENTS IS THAT THE BUYER OF A STANDARD PART IS ASSURED ONLY THAT AT ONE TIME THE MANUFACTURER MET ALL THE PERFORMANCE AND QUALITY REQUIREMENTS. THE PROGRAM DOES NOT ENSURE A STANDARD PART WILL MEET SPEC PERFORMANCE REQUIREMENTS. THE BUYER OF A STANDARD PART MUST CONDUCT HIS OWN QUALITY PROGRAM. IF HE ACCEPTS A DEFECTIVE STANDARD PART, HE IS LIABLE FOR THE CONSEQUENCES OF ITS FAILURE.

MIL SPEC QUALIFIED PARTS PROGRAM

DoD 4120.3M CHP 4
QUALIFICATION PROCEDURES

MIL-STD 965
PARTS CONTROL PROGRAM

MIL-Q-9858A
QUALITY PROGRAM

FAR SUBPART 9.2
QUALIFICATION REQUIREMENTS

QUALIFIED

PARTS

LIST/PROGRAM

IDENTIFIES

RESPONSIBILITIES

- ORIGINAL COMPONENT
  MANUFACTURER
  (QUAL, VERF.)

- BUYER
  DISC, DESC, ALC's
  PRIME CONTRACTORS
  (PROTECT YOURSELF)*
  - RECEIVING INSPECTION
  - SOURCE INSPECTION

* MAY BE REQUIRED BY POLICY

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NARRATIVE

WHAT WE EXPECT OF OUR SYSTEMS CONTRACTORS

PART OF THE REVIEW WAS AN EXAMINATION OF HOW WE INTRODUCE THREADED PARTS INTO THE AIR FORCE IN NEW SYSTEMS. AS SHOWN HERE, THERE ARE SEVERAL POINTS WHERE VERIFICATION OF QUALITY MUST BE DEMONSTRATED. THIS EVEN INCLUDES ACCEPTANCE TEST RESULTS BEFORE THE DD 250 IS SIGNED AND SERVICE REPORTS FROM THE USER AFTER DELIVERY. IT SHOULD BE NOTED THAT WE REQUIRE OUR PRIME CONTRACTORS OBTAIN AND MAINTAIN IN HIS ARCHIVES, DATA ON VENDOR SUPPLIED PARTS PERMITTING TRACEABILITY BACK TO THE ORIGIN. LAST, ALL COSTS INCURRED BY THE CONTRACTOR IN DEMONSTRATING THAT HARDWARE IS OF ACCEPTABLE QUALITY ARE CHARGEABLE TO THE CONTRACT.

WHAT WE EXPECT OF OUR SYSTEMS CONTRACTORS

- PROCUREMENT PRACTICES
  - GENERAL POLICY (SPECIFICATIONS)
    - -- SECTION 3: ESSENTIAL CHARACTERISTICS
    - -- SECTION 4: VERIFICATION OF ESSENTIAL CHARACTERISTICS
  - AFSC PRODUCT DIVISIONS
    - -- AIRCRAFT / ENGINE / POD / RADAR / ETC.
    - -- VERIFICATION:
      CONTRACTOR TESTING / USAF ACCEPTANCE TESTS
  - PRIME CONTRACTORS (PIECE PARTS, FASTENERS, ETC.) (PRELIMINARY)
    - -- VENDOR TEST DATA
    - -- INCOMING INSPECTION (SAMPLING BASIS)
      MIL-STD-105 AND MIL-STD-1535
    - -- TRACEABILITY TO ORIGIN
    - -- ACCEPTABLE VENDOR LISTS

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NARRATIVE

HISTORICAL CHARACTERISTICS

THE FACTORS PRESENTED ON THIS CHART HAVE THE CUMULATIVE EFFECT THAT GOVERNMENT REPROVISIONING IS DIFFERENT THAN THE PROCESS USED BY OUR PRIME CONTRACTORS AND THAT USED IN PRIVATE INDUSTRY. ONE OF THE BIGGEST DIFFERENCES IS THAT WE HAVE NOT BEEN CONDUCTING OUR OWN RECEIVING INSPECTIONS.

HISTORICAL CHARACTERISTICS

- GOVERNMENT VERSUS PRIVATE INDUSTRY
  - DIFFICULTY IMPLEMENTING PAST PERFORMANCE AS VENDOR SELECTION CRITERIA
  - DIFFICULTY IMPLEMENTING CUSTOMER (FIELD) FEEDBACK TO VENDORS/PURCHASING AGENTS
  - LIMITED RESOURCES TO OVERSEE VENDORS/ DISTRIBUTORS
  - PREFERENCE FOR:
    - -- PRICE COMPETITION
    - -- MULTI-VENDOR BASE VERSUS CONCENTRATING BUYS

CONSEQUENCE: GOVERNMENT REPROVISIONING DIFFERENT THAN
- PRIME CONTRACTORS
- PRIVATE INDUSTRY

NARRATIVE

ARE THERE NONCONFORMING CHARACTERISTICS IN THE INVENTORY?

THE ANSWER TO THE QUESTION POSED IN THE TITLE TO THIS CHART IS YES, THERE ARE NONCONFORMANCES IN THE FIELD. AND, THEY ARE NOT LIMITED TO DIMENSIONS. THERE HAVE BEEN OCCURRENCES OF JUST ABOUT ALL CHARACTERISTICS BEING FOUND TO BE IN NONCONFORMANCE. AS THE EXCERPT SHOWS, DLA-Q HAS NOTIFIED GOVERNMENT QUALITY ASSURANCE SPECIALISTS OF THE PROBLEMS AND CALLS FOR RECEIVING INSPECTIONS NOT ONLY OF PHYSICAL APPEARANCES BUT OF BASIC PROPERTIES.

ARE THERE NONCONFORMING CHARACTERISTICS IN THE INVENTORY?

TYPICAL CHARACTERISTICS	NONCONFORMANCES
STRENGTH	YES (SOME)
HARDNESS	YES (SOME)
PLATING	YES (SOME)
METALLURGY	YES (SOME)
THREAD GEOMETRIES	YES (SOME)

DEFENSE LOGISTICS AGENCY
HEADQUARTERS
CAMERON STATION
ALEXANDRIA, VIRGINIA 22304-6100

REPLY REFER TO DLA-Q

Q-TIP 87-9

31 AUG 1987

Nonconforming fasteners are not only a waste of money but may, in some cases, contribute to personal injury or death. The infiltration of nonconforming fasteners is due mainly to profit incentives, and ineffective or nonexistent contractor quality control procedures coupled with perceptions of "freedom" from detection and prosecution. Industrial fasteners are a prime candidate for willful misrepresentation and wrongdoing since detection of a nonconforming item (which is often mismarked) is difficult and requires technical testing of the basic metals content.

NARRATIVE

FIELD EXPERIENCE

THE IMPROVING SAFETY RECORD SINCE THE 1950s INDICATES THAT SAFETY EXPECTATIONS ARE BEING MET. ONE REASON IS THAT THE ACCEPTED DESIGN PRACTICE IS TO AVOID MAKING THE SCREW THREAD A CRITICAL ELEMENT IN THE LOAD PATH IF AT ALL POSSIBLE.

HOWEVER, THE AMOUNT OF UNSCHEDULED MAINTENANCE DUE TO NONCONFORMING PRODUCTS IS NOT KNOWN.

FIELD EXPERIENCE

WE BELIEVE:

- CURRENT SYSTEMS ARE MEETING SAFETY EXPECTATIONS
  - GOOD DESIGN PRACTICES

"MAKE THE FASTENER (SCREW THREAD) NON-CRITICAL IF AT ALL POSSIBLE"

- UNKNOWN ECONOMIC AND COMBAT CAPABILITY LIMITATIONS DUE TO NONCONFORMANCES IN THE SYSTEM

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NARRATIVE

A QUESTION

THE GOVERNMENT DIRECTS THE USE OF CLASS 3 THREADS

ONE CONTRACTOR WAS ASKED WHY HE USED CLASS 3 THREADS. THE ANSWER WAS THAT THE GOVERNMENT DICTATED THIS IN THE PRIME ITEM DEVELOPMENT SPECIFICATIONS PLACED ON THE CONTRACT. WHEN WE LOOKED INTO SPECS AND HOW WE GENERATE REQUIREMENTS FOR THREADED PRODUCTS, WE FOUND THAT WE HAD "MET THE ENEMY AND HE IS US." AS IS SEEN ON THE NEXT CHART, TWO KEY SPECS, MIL-STD-1515 AND MIL-STD-454 BOTH STATE THAT CLASS 3 THREADS PER MIL-S-8879 OR MIL-S-7742 MUST BE USED. THE NET RESULT IS THE USE OF HIGH PERFORMANCE FASTENERS IN RELATIVELY MUNDANE EQUIPMENT. NEEDLESS TO SAY, WE WILL REMOVE THIS DIRECTIVE LANGUAGE FROM THESE STANDARDS.

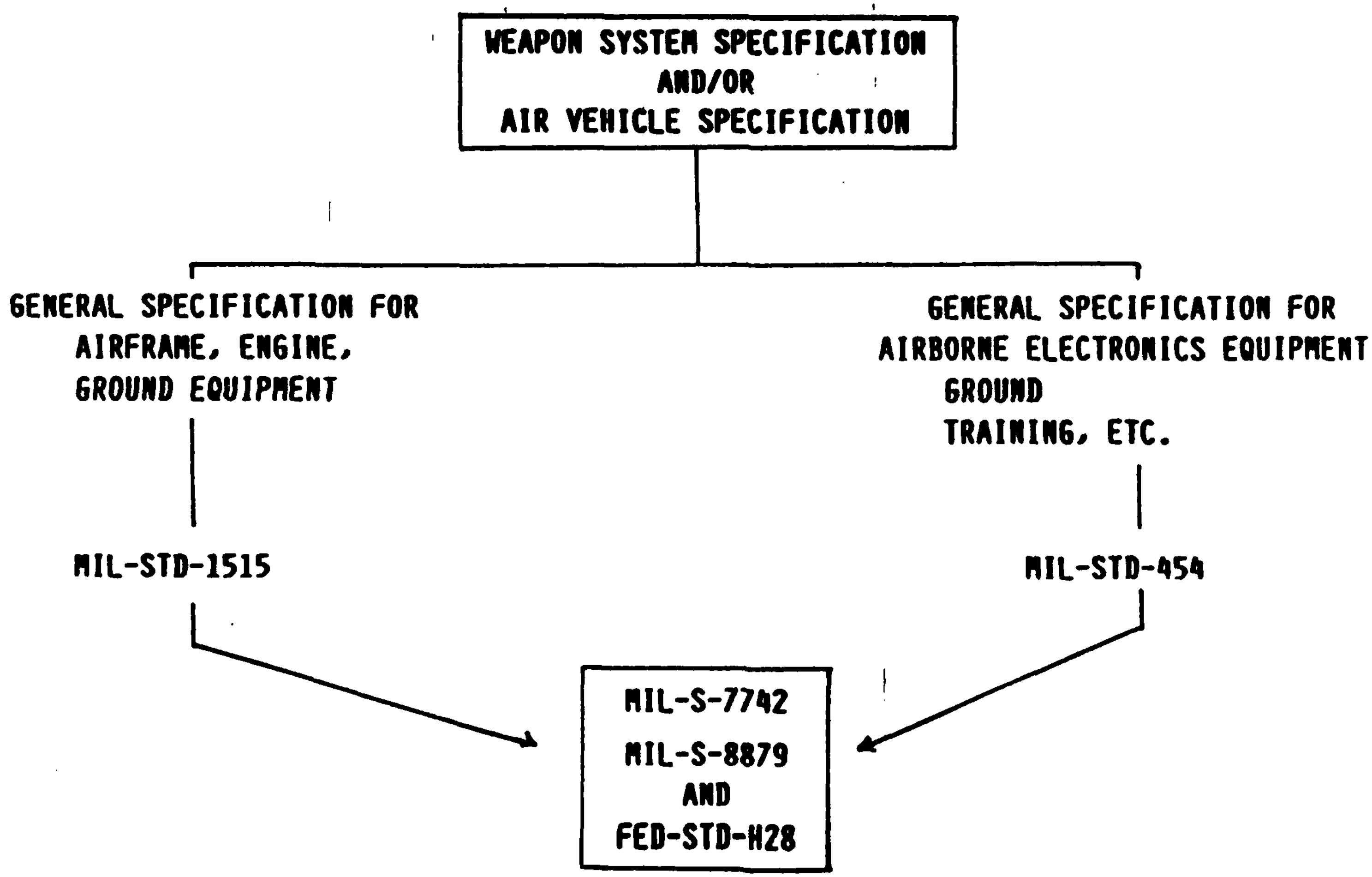
A QUESTION

- HOW / WHY DO WE PUT / HAVE REQUIREMENTS FOR CLASS 3 THREADS ON CONTRACT?

  (SEE NEXT CHART)

- SHOULD WE CHANGE?
  - YES!
  - CURRENT SYSTEM TELLS CONTRACTOR;
    "HOW TO"
    NOT
    "WHAT WE WANT"!

THE GOVERNMENT DIRECTS THE USE OF CLASS 3 THREADS



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NARRATIVE

THE POLICY

THE TEAM FINDINGS WERE SYNTHESIZED INTO A RECOMMENDED POLICY TO BE IMPLEMENTED IN THE AIR FORCE AND HOPEFULLY ACROSS DOD. ONE OF THE KEY ELEMENTS FOR THE POLICY WAS THE RECOGNITION THAT REGULATIONS ALREADY IN EXISTENCE WERE NOT BEING ENFORCED. THE MAIN EFFECT OF THE POLICY WOULD BE TO REBASELINE GOVERNMENT ACCEPTANCE PRACTICES TO INDEPENDENTLY VERIFY THE QUALITY OF PRODUCTS BEFORE ACCEPTANCE.

THE POLICY

NARRATIVE

FUNDAMENTAL PRINCIPLES FOR THE POLICY

AS A RESULT OF THE TEAM'S FINDINGS, A POLICY WAS RECOMMENDED WHICH WOULD ADDRESS THE DEFICIENCIES. WE WOULD HAVE TO CHANGE OUR SPECIFICATION OF REQUIREMENTS TO A PERFORMANCE BASIS AND NOT USE "HOW TO" LANGUAGE.

- THREADED PRODUCTS SHOULD BE ASSESSED IN THE DESIGN AND PROCUREMENT PROCESS FROM THE PERSPECTIVE OF THE CONSEQUENCES OF A FAILURE NOT FROM THE TRADITIONAL PERSPECTIVE OF INITIAL PIECE PART COST.
- THE VERIFICATION OF QUALITY SHOULD BE RELATED TO THE APPLICATION AND ALL ESSENTIAL CHARACTERISTICS SHOULD BE VERIFIED.
- THE RESPONSIBILITIES OF PRIME CONTRACTORS AND VENDORS AND DISTRIBUTORS HAS TO BE BROUGHT TO A CONSISTENT LEVEL.
- THE GOVERNMENT HAS TO CONDUCT ITS OWN QUALITY INSPECTIONS JUST AS IT REQUIRES ITS PRIME CONTRACTORS.

THIS INCLUDES RECOGNITION THAT THE BOLT THAT HOLDS TWO FRACTURE CRITICAL COMPONENTS TOGETHER IS JUST AS CRITICAL AND MUST BE MANAGED WITH THE SAME PROCEDURES AS THE MEMBERS IT JOINS.

FUNDAMENTAL PRINCIPLES FOR THE POLICY

- EMPHASIZE:
  - "WHAT WE WANT" / MIL PRIME

"MAKE THE FASTENER (SCREW THREAD) NON-CRITICAL IF AT ALL POSSIBLE"

  - POTENTIAL CONSEQUENCE OF IN-SERVICE FAILURE
    VERSUS
    TRADITIONAL INITIAL COST
  - TAILOR VERIFICATION TASKS
  - BALANCED ATTENTION (ALL ESSENTIAL CHARACTERISTICS)
  - PRIME CONTRACTOR / VENDOR / DISTRIBUTOR RESPONSIBILITIES
  - GOVERNMENTAL RESPONSIBILITY TO IMPLEMENT THE SAME DISCIPLINE IT DEMANDS OF OTHERS
  - SAME RESPONSIBILITY TO MANAGE CRITICAL (ETC.) PIECE PARTS AS MAJOR STRUCTURAL COMPONENTS (I.E., SAFETY CRITICAL TURBINE BLADE, WING ATTACHMENT, ETC.)

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NARRATIVE

FASTENER/SCREW THREAD POLICY (1)

IT IS AIR FORCE POLICY TO USE SCREW THREADED PRODUCTS THAT CONFORM TO THE DESIGN SPECIFICATIONS AND STANDARDS THAT PERMIT AEROSPACE DESIGNERS THE FLEXIBILITY TO ACHIEVE AN OPTIMUM BALANCE OF PERFORMANCE, SAFETY, RELIABILITY, AND INTERCHANGEABILITY WITH MINIMUM COSTS, LOGISTICS INVENTORY, AND MAINTENANCE. CONTRACTS WILL NOT BE AWARDED TO THOSE MANUFACTURERS, SUPPLIERS, OR DISTRIBUTORS WHO HAVE A HISTORY OF PROVIDING PRODUCTS OF AN UNSATISFACTORY QUALITY.

EFFECTIVE IMMEDIATELY, ORGANIZATIONS RESPONSIBLE FOR TECHNICAL REQUIREMENTS SHALL CLASSIFY SCREW THREADED PRODUCTS ACCORDING TO THE CONSEQUENCE OF THEIR FAILURE IN THE WEAPON SYSTEM AND/OR SUPPORT EQUIPMENT. THE FOLLOWING CATEGORIES - CATASTROPHIC, CRITICAL, MARGINAL, AND MINOR - ARE TYPICAL FOR A FAILURE MODE, EFFECTS AND CRITICALITY ANALYSIS (FMECA) OF MIL-STD-1629A. THE RESULTS OF THE FMECA WILL BE USED TO ESTABLISH TWO APPLICATION CATEGORIES FOR THREADED PRODUCTS. THE TWO APPLICATION CATEGORIES ARE DEFINED AS FOLLOWS:

A. SAFETY CRITICAL: A FAILURE WHICH MAY CAUSE DEATH, SEVERE INJURY, OR WEAPON SYSTEM LOSS.

B. ALL OTHER FAILURE CONSEQUENCES.

NARRATIVE

FASTENER/SCREW THREAD POLICY (1) (CONTINUED)

CONSEQUENCE OF FAILURE DETERMINES THE APPLICATION CATEGORY WITH SAFETY CRITICAL RECEIVING THE MOST STRINGENT CONTROLS.

A. FOR SAFETY CRITICAL PRODUCTS, THE SELLER IS REQUIRED TO INSPECT EACH PART FOR IDENTIFIED NONDESTRUCTIVE CHARACTERISTICS AND PERFORM APPROPRIATE SAMPLING FOR DESTRUCTIVE TESTING WITHIN EACH LOT.

B. FOR OTHER THREADED PRODUCTS, THE SELLER SHALL PERFORM SAMPLE INSPECTION PER THE ITEM SPECIFICATIONS.

SELLERS THAT HAVE DEMONSTRATED AN UNSATISFACTORY QUALITY HISTORY REPRESENT AN ADDITIONAL COST BURDEN TO THE BUYING ORGANIZATION. THAT ADDITIONAL COST BURDEN OF THE ACCEPTANCE TESTING SHOULD BE CONSIDERED AS A FACTOR IN THE AWARD OF THE CONTRACT.

FASTENER/SCREW THREAD POLICY

- TAILORED INSPECTION REQUIREMENTS BY APPLICATION CATEGORY

APPLICATION	CRITERIA	INSPECTION (SELLER)
SAFETY CRITICAL	WEAPON SYSTEM LOSS DEATH SEVERE INJURY	- EACH PART FOR NONDESTRUCTIVE INSPECTION - APPROPRIATE SAMPLING WITHIN EACH LOT FOR DESTRUCTIVE TESTS
OTHER	ALL OTHER FAILURE CONSEQUENCES	- APPROPRIATE SAMPLING PER ITEM SPECIFICATIONS

NARRATIVE

FASTENER/SCREW THREAD POLICY (2)

THE BUYER (PRIME CONTRACTOR, GOVERNMENT, ETC.) IS RESPONSIBLE FOR ASSURING THAT THE SELLER HAS MET THESE REQUIREMENTS. THE BUYER'S RESPONSIBILITY CAN BE SATISFIED THROUGH EITHER SOURCE WITNESSING OF THE APPROPRIATE TESTING OR RECEIVING INSPECTION TO THE SAME LEVEL AS DIRECTED UPON THE SELLER UNTIL A QUALITY HISTORY HAS BEEN ESTABLISHED. THIS RESPONSIBILITY MAY BE MAINTAINED WITH REDUCED INSPECTION AFTER ESTABLISHMENT OF THE QUALITY HISTORY AS WARRANTED BY THE INSPECTION RESULTS.

THE BUYER IS RESPONSIBLE FOR ASSURING THAT THE SELLER HAS ACCOMPLISHED THIS. THE BUYER SHALL IMPLEMENT APPROPRIATE ACCEPTANCE SAMPLING INSPECTIONS.

FASTENER/SCREW THREAD POLICY

- FREQUENCY OF INSPECTION BY BUYER (SOURCE OR RECEIVING)

APPLICATION	FREQUENCY
SAFETY CRITICAL	TO SAME LEVEL AS DIRECTED ON SELLER UNTIL QUALITY HISTORY ESTABLISHED. REDUCED INSPECTION WHEN QUALITY HISTORY WARRANTS.
OTHER	APPROPRIATE SAMPLING INSPECTIONS TO ENSURE SELLER IS COMPLYING WITH ITEM SPECIFICATIONS.

NARRATIVE

FASTENER/SCREW THREAD POLICY (3)

CLASSIFICATION OF THREADED PRODUCTS WILL INVOLVE THE FOLLOWING ACTIONS:

A. FOR APPLICATION TO SYSTEMS UNDER DEVELOPMENT OR PRODUCTION THIS TASK CAN BE ACCOMPLISHED THROUGH A FAILURE MODES, EFFECTS AND CRITICALITY ANALYSIS (FMECA) IN ACCORDANCE WITH MIL-STD-1629A. OTHER APPROACHES EQUIVALENT TO AN FMECA ARE ACCEPTABLE.

B. FOR SYSTEMS NO LONGER IN PRODUCTION, APPLICATION CAN BE ESTABLISHED BASED ON EXPERIENCE. AN APPROPRIATELY TAILORED FMECA MAY BE USED IF JUDGED NECESSARY.

C. THE APPROPRIATE APPLICATION DRAWINGS MUST BE MODIFIED AND THE REPROCUREMENT DATA MUST BE UPDATED TO REFLECT THE ASSIGNED APPLICATION.

EFFECTIVE IMMEDIATELY, GAGING METHOD C OF MIL-S-8879A AND MIL-S-7742B OR FED-STD-H28/20 SYSTEM 23 REFERENCING ANSI/ASME B1.3M-1986 SHALL BE USED FOR ACCEPTANCE OF SAFETY CRITICAL APPLICATION CLASS 3 THREADS. GAGING METHOD B OF MIL-S-8879A AND MIL-S-7742B OR FED-STD-H28/20 SYSTEM 22 REFERENCING ANSI/ASME B1.3M-1986 ISSUED 15 MARCH 1986 SHALL BE USED FOR ACCEPTANCE OF OTHER CLASS 3 EXTERNAL AND INTERNAL THREADED PRODUCTS. AMERICAN NATIONAL STANDARD INSTITUTE (ANSI) NATION STANDARD ANSI/ASME B1.3M-1986 DEFINES GAGING SYSTEMS 21, 22, AND 23 FOR DIMENSTIONAL ACCEPTABILITY OF THREADS. THESE GAGING SYSTEMS ARE EQUIVALENT TO MILITARY GAGING SYSTEMS A, B, AND C, RESPECTIVELY. NEW CLASS 3 SCREW THREADED PRODUCTS PURCHASED FOR OR MANUFACTURED BY AIR FORCE AGENCIES SHALL MEET THE REQUIREMENTS OF THIS POLICY. THIS DIRECTION WILL BE INCORPORATED INTO REVISIONS TO MIL-S-8879 AND MIL-S-7742.

NARRATIVE

FASTENER/SCREW THREAD POLICY (3) (CONTINUED)

MORE PRECISE STATE OF THE ART METHODS ARE PERMITTED. THIS GUIDANCE DOES NOT PRECLUDE ADOPTION OF METHODS FOR ASSURING SPECIFICATION CONFORMANCE USING ADVANCED TECHNOLOGIES AS THEY BECOME AVAILABLE. THE CHOICE OF ALTERNATE METHODS DOES NOT INFER PERMISSION TO DEPART FROM SPECIFIED DIMENSIONS AND TOLERANCES. FOR EFFICIENT PRODUCTION, THREAD MANUFACTURERS SHOULD EMPLOY PROCESS CONTROL METHODS TO THE LEVEL OF CRITICAL APPLICATION SPECIFICATIONS.

FASTENER/SCREW THREAD POLICY

- OTHER REQUIREMENTS
  - DETERMINATION OF APPLICATION CATEGORY
    - SYSTEMS IN DEVELOPMENT OR PRODUCTION SHALL USE A FAILURE MODE, EFFECTS AND CRITICALITY ANALYSIS (FMECA) (MIL-STD-1629A)
    - FOR SYSTEMS NO LONGER IN PRODUCTION, USE SERVICE EXPERIENCE AND/OR TAILORED FMECA
  - DRAWINGS AND REPROCUREMENT DATA MUST BE UPDATED TO REFLECT APPLICATION CATEGORY
  - GAGING METHODS OF MIL-S-8879/7742 OR ANSI/ASME B1.3M-1986 (15 MAR 86) SHALL BE USED AS FOLLOWS FOR EXTERNAL AND INTERNAL THREADS:

SAFETY CRITICAL	METHOD C OR SYSTEM 23
OTHER	METHOD B OR SYSTEM 22

  - MORE ADVANCE INSPECTION METHODS MAY BE USED AS TECHNOLOGY BECOMES AVAILABLE
  - THREAD MANUFACTURERS SHOULD EMPLOY PROCESS CONTROL MEASURES AS NECESSARY TO ENSURE ITEM SPEC IS SATISFIED

NARRATIVE

IMPLEMENTATION ACTIONS

ALTHOUGH THE POLICY RECOMMENDED WAS NOT TOO DIFFICULT TO PREPARE, THE IMPLEMENTATION WILL REQUIRE MANY SPECS AND STANDARDS, INCLUDING THE ASIP AND ENSIP STANDARDS TO BE REVISED. THE NEXT SERIES OF CHARTS WILL SHOW THE ACTIONS ASSIGNED TO AFSC AND AFLC.

IMPLEMENTATION ACTIONS

NARRATIVE

AFSC ACTIONS

COMMAND ACTIONS TO IMPLEMENT THIS POLICY FOR NEW AND EMERGING SYSTEMS ARE AS SHOWN. ONE OF THE LONGEST TASKS TO COMPLETE WILL BE THE UPDATING OF THE MIL SPECS AND STANDARDS. THE ADDED TASKS AT THE BOTTOM OF THE CHART WILL REQUIRE CONTRACTORS TO REVIEW THEIR DRAWINGS TO ACCOMMODATE CHANGES IN THE MANNER IN WHICH FASTENERS AND SCREW THREADS ARE DESIGNATED AND ESSENTIAL CHARACTERISTICS IDENTIFIED.

AFSC ACTIONS

- REVISE USAF DEVELOPMENT STDS, SPECS, AND PRODUCT SPECIFICATIONS TO REFLECT POLICY
- INCORPORATE REVISED POLICY IN NEW PROCUREMENT ACTIONS
  - AD HOC CONTRACT VERBIAGE (SHORT TERM)
  - REVISED MIL STDS, SPECS, ETC. (AS AVAILABLE)
- MODIFY EXISTING CONTRACTS FOR EMERGING WEAPON SYSTEMS, ETC.
  - NEXT (I.E., BLOCK CHANGE, LRIP, ETC.) CONTRACT CHANGE
  - ADDED TASKS
    - ESTABLISH APPLICATION CATEGORY (PIECE PARTS)
    - MODIFY DRAWING (AS APPROPRIATE)
    - UPDATE REPROCUREMENT DATA

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NARRATIVE

AFSC ACTIONS (CONTINUED)

THESE ARE THE GENERAL SPECIFICATIONS WHICH HAVE TO BE MODIFIED. IN ADDITION, THERE ARE SEVERAL THOUSAND SPECS FOR STANDARD PARTS THAT WILL REQUIRE MODIFICATION TO INCORPORATE INSPECTION BY APPLICATION AND DEFINITION OF ESSENTIAL CHARACTERISTICS FOR EACH APPLICATION. IT IS ENVISIONED THAT THIS WOULD BE ACCOMPLISHED WHEN A RESUPPLY ORDER FOR THESE PIECE PARTS IS PLACED. BECAUSE MANY OF THESE ARE ASSIGNED TO OTHER AGENCIES AS PREPARING ACTIVITY, A JOINT SERVICE COORDINATION IS REQUIRED AND THE EFFORTS OF HQ USAF LE/RD TO ADVOCATE THESE CHANGES WILL BE REQUIRED.

AFSC ACTIONS
(CONTINUED)

- AERONAUTICAL SYSTEMS DIVISION (ASD)

  - MODIFY THE FOLLOWING DEVELOPMENT STDS, SPECS, ETC.

DOCUMENT	OPR	TITLE
MIL-STD-1515	ASD	FASTENER SYSTEMS FOR AEROSPACE APPLICATIONS
MIL-STD-454	AFSC/PL	STANDARD GENERAL REQUIREMENTS FOR ELECTRONIC EQUIPMENT
MIL-S-8879	ASD	SCREW THREADS, CONTROLLED RADIUS, ETC.
MIL-S-7742	ASD	SCREW THREADS, STANDARD, ETC.
MIL-STD-1530	ASD	STRUCTURAL INTEGRITY PROGRAM (ASIP)
MIL-A-87221	ASD	GENERAL SPECIFICATION FOR AIRCRAFT STRUCTURES
MIL-STD-1793	ASD	ENGINE STRUCTURAL INTEGRITY PROGRAM (ENSIP)
MIL-A-87244	ASD	AVIONICS / ELECTRONICS INTEGRITY PROGRAM (AVIP)
MIL-STD-1792	ASD	FASTENER SYSTEMS, MECHANICAL, AIR VEHICLE

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AFSC ACTIONS
(CONTINUED)

- AERONAUTICAL SYSTEMS DIVISION (ASD)
  - RECOMMEND MODIFICATION TO CUSTODIANS OF OTHER SPECS

DOCUMENT	OPR	TITLE
FED-STD-H-28	DISC	SCREW THREAD STANDARDS FOR FEDERAL SERVICE
CLASS 3 ITEM SPECS	OTHERS	X X X X

  - PREPARE INTERIM STATEMENT OF WORK
  - INCORPORATE IN NEW / EMERGING CONTRACTS

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OTHER ACTIONS

- ITEM SPECS

  ISSUE: APPROXIMATELY 170,000 CLASS 3 ITEM SPECs, ETC.
  WILL HAVE TO BE SCREENED AND REVISED, AS APPROPRIATE

  - MOST ITEM SPECs BELONG TO OTHER SERVICES / ORGANIZATIONS
  - JOINT SERVICE COORDINATION REQUIRED
    AIR FORCE / ARMY / NAVY / DLA
  - WORKLOAD / SCHEDULE UNCERTAIN
  - WILL SUPPORT USAF / LE-RD LEAD

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NARRATIVE

AFLC ACTIONS

HQ AFLC REVIEWED THE TEAM RECOMMENDATIONS AND DECIDED THAT SOME GOVERNMENT RECEIVING INSPECTIONS WERE REQUIRED. SYSTEM AND ITEM MANAGERS ARE BEING TASKED TO REVIEW THEIR THREADED PRODUCTS FROM THE APPLICATIONS VIEWPOINT. THE CATEGORY FOR THIS HARDWARE IS TO BE DETERMINED FROM FMECA DATA IF IT EXISTS OR DEVELOPED THROUGH AN EQUIVALENT ANALYSIS USING THE ACCUMULATED SERVICE EXPERIENCE AS A DATA BASE. EXISTING INVENTORIES WILL BE CONSUMED. IF A PART HAS CATASTROPHIC OR CRITICAL APPLICATIONS, THE CRITICAL ATTRIBUTES WILL BE INSPECTED BEFORE THE PART IS ISSUED FOR INSTALLATION. ITEMS REPROCURED WILL CONTAIN REQUIREMENTS REFLECTING THE NEW POLICY BOTH IN THE TECHNICAL AND QUALITY SECTIONS OF THE PROCUREMENT PACKAGE. THE LAST BULLET ON THIS CHART IS A REQUEST FOR THE LOGISTICS MANAGERS TO ACCOUNT FOR PAST PERFORMANCE IN AWARDING NEW CONTRACTS. THERE ARE SEVERAL SUCH INITIATIVES AMONG THE PURCHASING CENTERS OF DLA WHICH COULD BE ADOPTED BY THE AIR LOGISTICS CENTERS.

AFLC ACTIONS

- ACCEPT THE WORKLOAD ASSOCIATED WITH QUALITY VERIFICATION OF CRITICAL ITEMS
- SYSTEM PROGRAM MANAGER / ITEM MANAGER TASKING
  - ACCOMPLISH FMECA OR EQUIVALENT ANALYSIS BASED UPON EXPERIENCE
  - ENGINEERING REVIEW OF CRITICAL ITEMS
  - CONSUME EXISTING INVENTORY WITH COMBINATION OF INSPECTION / TEST / FMECA / EXPERIENCE
- INCORPORATE REVISED POLICY IN REPROCUREMENT ACTIONS
- EMPHASIZE PROCUREMENT PRACTICES WHICH ENCOURAGE VENDORS TO MANAGE THEIR QUALITY RESPONSIBILITIES

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DEFENSE INDUSTRIAL SUPPLY CENTER

SURVEY OF FASTENER QUALITY

NARRATIVE

DISC SURVEY: CLASS 3 EXTERNAL THREADED FASTENERS

WHILE THE TEAM WAS WRAPPING UP ITS REPORT, DEFENSE INDUSTRIAL SUPPLY CENTER UNDERTOOK A SAMPLING PROGRAM OF EXTERNALLY THREADED FASTENERS IT WAS PROCURING. ONE HUNDRED FIFTY-SEVEN CONTRACTS WERE CHECKED. TEN FASTENERS FROM EACH CONTRACTED QUANTITY WERE SELECTED AND SENT TO TESTING LABORATORIES OUTSIDE THE GOVERNMENT FOR PHYSICAL PROPERTY, DIMENSIONAL AND PLATING TESTS IN ACCORDANCE WITH SPECIFICATIONS FOR THE FASTENER. THE RESULTS OF THIS SURVEY ARE SHOWN ON THE BOTTOM HALF OF THIS CHART. THESE DATA PROVIDE THE CONFIRMATION OF THE NEED FOR RECEIVING INSPECTIONS. WHILE THESE CONTRACTS HAPPENED TO SAMPLE PRODUCTS FROM SMALL VENDORS AND DISTRIBUTORS, A FOLLOW-ON EFFORT IS GOING TO EVALUATE THE LARGE DOMESTIC FASTENER COMPANIES' PRODUCTS SHIPPED TO DISC.

DISC SURVEY: CLASS 3 EXTERNAL THREADED FASTENERS

- SAMPLED MATERIAL FROM 157 CONTRACTS (1987)
  - 10 UNITS PER CONTRACT
  - TYPICAL MEASUREMENTS:

    MATERIAL COMPOSITION — STRENGTH

    PLATING — HARDNESS

    VARIOUS THREAD DIMENSIONS

- RESULTS (OCT 87):
  - 27% OF CONTRACTS CONFORMED TO SPECs
  - 73% NONCONFORMANCE DUE TO:

VARIOUS MATERIAL CHARACTERISTICS	- 27%
VARIOUS THREAD CHARACTERISTICS	- 23%
BOTH MATERIAL AND THREADS	- 23%
	73%

- DLA/DISC SURVEY SUBSTANTIATES STUDY FINDINGS

IMPLICATIONS OF THE POLICY

NARRATIVE

IMPLICATIONS OF THE POLICY

AS WITH MOST CHANGES, THERE IS SOME GOOD NEWS AND SOME BAD NEWS. DESIGNERS ARE GOING TO HAVE TO START PAYING MORE ATTENTION TO THREAD SELECTION AND DEFINITION. WHILE THE CHANGES WILL GIVE THEM MORE FREEDOM OF CHOICE, IT ALSO WILL REQUIRE CONSIDERATION OF THE APPLICATION AND SPECIFICATION OF THOSE CHARACTERISTICS WHICH ARE ESSENTIAL FOR THE APPLICATION. THIS WILL REQUIRE MUCH MORE WORK THAN THE STANDARD "THREADS PER MIL-S-8879" NOTE WE FOUND ON MOST DRAWINGS. SIMILARLY, THE QUALITY ASSURANCE REQUIREMENTS WILL HAVE TO BE MUCH MORE SPECIFIC IN DEFINING HOW THE CHARACTERISTICS ARE TO BE VERIFIED AND THE SAMPLING REQUIREMENTS NECESSARY TO ENSURE RECEIVING INSPECTIONS ARE ADEQUATE FOR THE APPLICATION.

IMPLICATIONS OF THE POLICY

- FOR THE DESIGNER:
  - NO LONGER LIMITED TO CLASS 3 THREADS
  - DEFINITION OF THREADED PARTS WILL REQUIRE SPECIFICATION OF ESSENTIAL CHARACTERISTICS BASED ON APPLICATION
  - DESIGNATION WILL INCLUDE APPLICATION CATEGORY
- FOR THE QUALITY COMMUNITY:
  - DEFAULT REQUIREMENTS OF MIL-S-8879/7742 HAVE BEEN ELIMINATED
  - INSPECTION REQUIREMENTS WILL BE BASED ON ESSENTIAL CHARACTERISTICS
  - QUANTITIES INSPECTED WILL BE BASED ON APPLICATION CATEGORY
- FOR THE PARTS MANAGERS:
  - ITEM SPECS FOR STANDARD PARTS WILL HAVE TO BE UPDATED
  - MANAGING STOCK FOR DIFFERENT APPLICATIONS HAS TO BE DETERMINED

SUMMARY

NARRATIVE

SUMMARY

TO SUMMARIZE - DEFICIENCIES IN THE SPECIFICATIONS AND REPROCUREMENT OF THREADED PARTS HAVE BEEN IDENTIFIED. A SERIES OF CORRECTIVE ACTIONS HAVE BEEN IDENTIFIED AND AN AIR FORCE POLICY ISSUED TO IMPLEMENT THEM. TRI-SERVICE CONCURRENCE IN THE CHANGES TO SPECIFICATIONS AND PROCUREMENT PRACTICES IS BEING SOUGHT. TO FULLY IMPLEMENT THESE CORRECTIVE ACTIONS WILL AFFECT US ALL. THE BEST STRATEGY TO FULLY IMPLEMENT THE POLICY WILL REQUIRE FURTHER MEETINGS. ASD/EN-PA IS PLANNING A GOVERNMENT-INDUSTRY MEETING IN FEBRUARY 1988 FOR THIS PURPOSE.

SUMMARY

- A TECHNICAL REVIEW OF THREADED PARTS UNCOVERED/CONFIRMED DEFICIENCIES IN THE SPECIFICATION AND REPROCUREMENT OF SCREW THREADED PARTS
- SAMPLING TESTS BY DISC HAVE CONFIRMED EXISTENCE OF WIDESPREAD NONCONFORMANCES IN FASTENERS ENTERING DOD RESUPPLY SYSTEM
- AN AIR FORCE POLICY TO CORRECT THESE DEFICIENCIES HAS BEEN ISSUED AND TRI-SERVICE CONCURRENCE IS BEING PURSUED
- THE POLICY CHANGE WILL AFFECT US ALL
- FURTHER DISCUSSIONS ARE PLANNED FOR EARLY 1988

ENGINE DURABILITY & DAMAGE TOLERANCE ASSESSMENTS

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FLIGHT SYSTEMS ENGINEERING

87(MCS)#135/11EN-208-8

OVERVIEW

- INTRODUCTION
  - INTEGRITY PROCESS
  - ASSESSMENT OBJECTIVES/BENEFITS
- BACKGROUND
- TECHNICAL APPROACH & TASKS
- COST-BENEFIT CONSIDERATIONS
- SUMMARY

THE "INTEGRITY" PROCESS

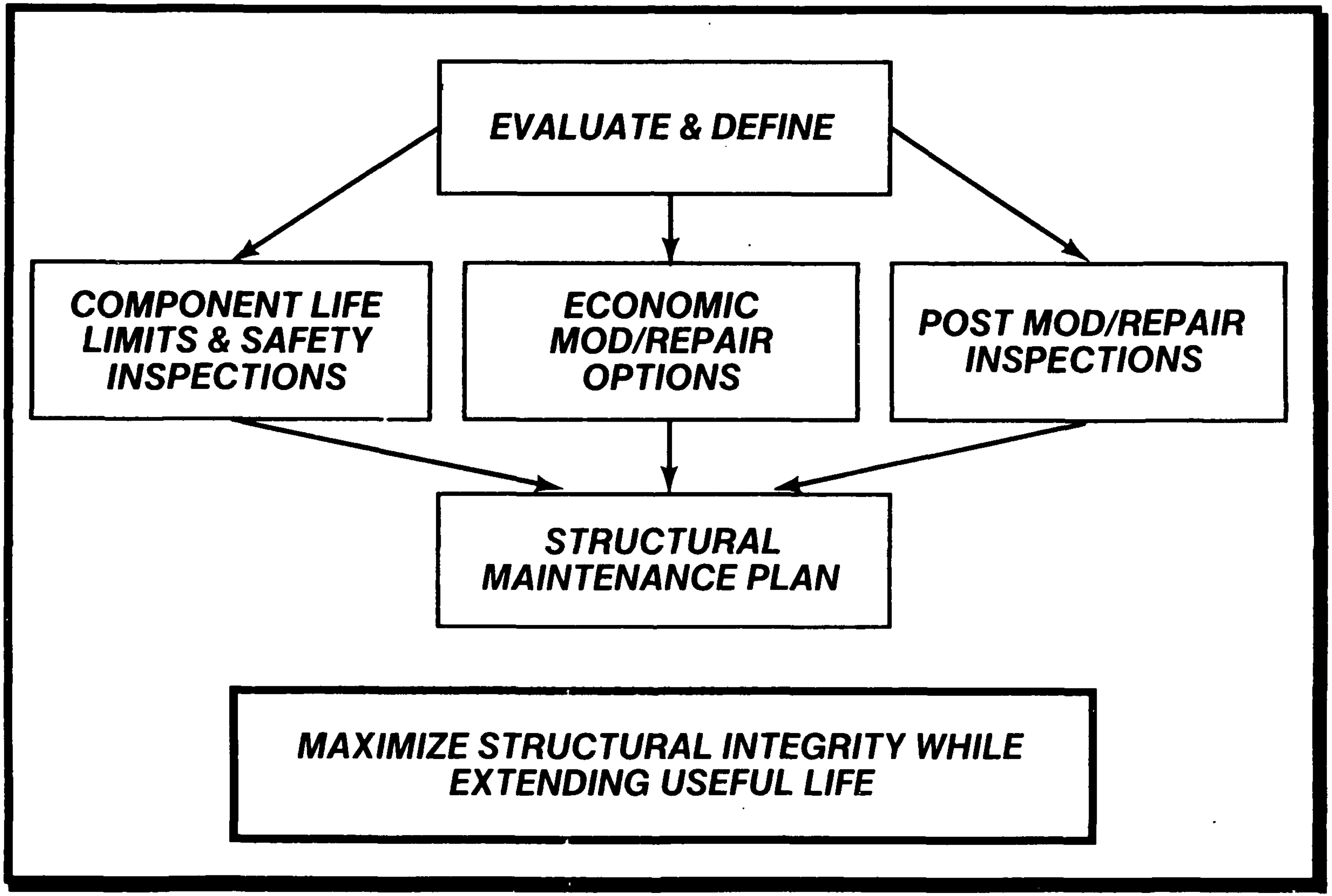
- AN ORGANIZED AND DISCIPLINED APPROACH TO THE DESIGN, DEVELOPMENT, QUALIFICATION, PRODUCTION, AND LIFE MANAGEMENT OF A PRODUCT WITH THE GOAL OF ENSURING:
  - SAFETY
  - MISSION CAPABILITY
  - RELIABILITY
  - MANTAINABILITY
  - REDUCED COST OF OWNERSHIP (LCC)

COMPARATIVE ANALYSIS OF AIR FORCE & COMMERCIAL PRACTICES

AIR FORCE (MIL-STD-1783)	FAA (FAR 33)
• LIFE DEFINITION ON ALL DURABILITY & FRACTURE CRITICAL HARDWARE (STATIC & ROTATING)	• DEFINITION OF LIFE ON ROTATING PARTS
• ESTABLISHMENT OF ECONOMIC & RESIDUAL LIFE FOR COMPONENTS	• ESTABLISHMENT OF ECONOMIC LIFE FOR MAJOR STRUCTURE
• EXTENSIVE ANALYSIS & TESTING (SPECIMEN, COMPONENT, ENGINE) TO CHARACTERIZE THE DESIGN	• TESTING AIMED AT CERTIFYING DESIGN – MODEL TESTS
• DAMAGE TOLERANCE USED IN FORMULATING SAFETY INSPECTION INTERVALS. ALSO, EMPLOYED IN EXTENDING USEFUL LIFE OF LIFE LIMITED COMPONENTS	• LCF LIFE SETS FIELD LIMITS (REPAIR/REPLACEMENT)
• INSTITUTION OF ENHANCED INSPECTIONS FOR SAFETY OR MISSION CRITICAL COMPONENTS	• NORMAL PRODUCTION PROCESSES/ INSPECTIONS
• LIFE MANAGEMENT PROGRAM	• CONTINUED AIRWORTHINESS
PREVENTIVE MAINTENANCE CONCEPT	CORRECTIVE MAINTENANCE CONCEPT

OBJECT
DISK TTT

OBJECTIVES



BENEFITS OF STRUCTURAL ASSESSMENT

DIRECT

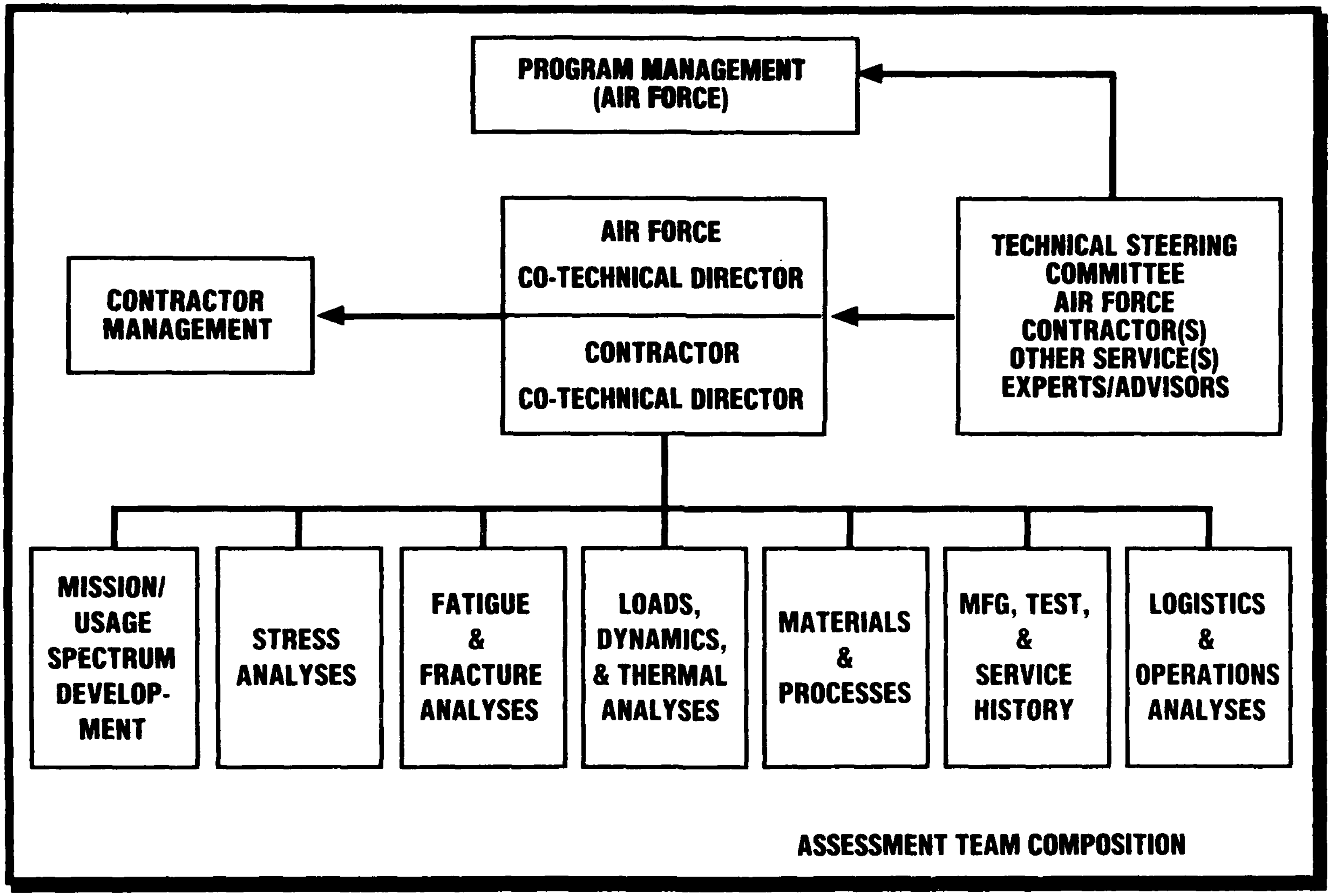
- IMPROVED SAFETY, RELIABILITY (DURABILITY & QUALITY), & ECONOMICS (SPARE PARTS)
- INSTITUTION OF INSPECTION/STRUCTURAL MAINTENANCE SYSTEM THAT ACCOMMODATES LIFE EXTENSION
- DEVELOPMENT OF PARAMETRIC TOOLS FOR ASSESSING IMPACT OF DESIGN/USAGE CHANGES
- SENSITIZATION OF DESIGN PROCESS FOR REDESIGNS UNDERTAKEN DURING DEVELOPMENT/COMPONENT IMPROVEMENT

INDIRECT

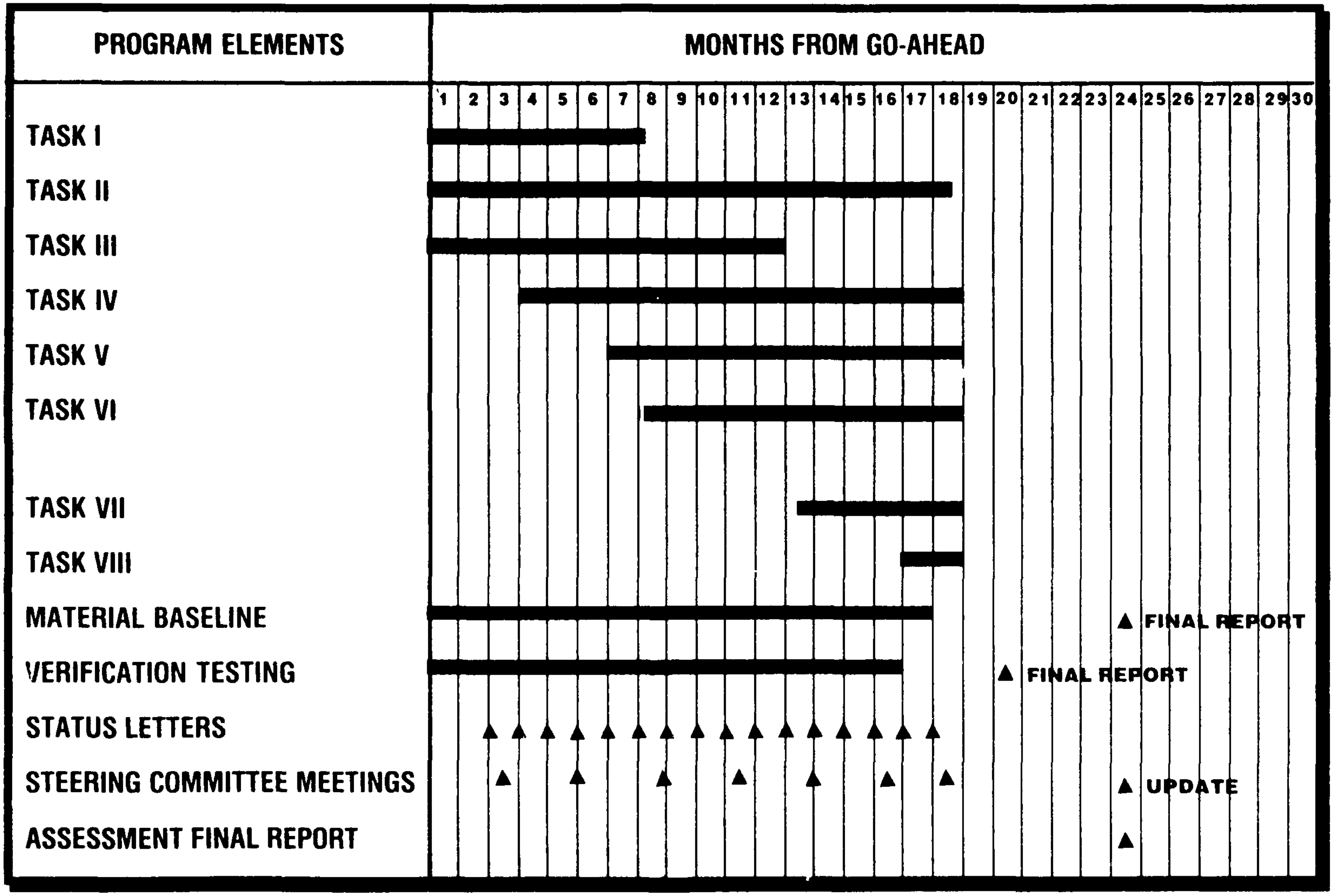
- EXPANDED KNOWLEDGE BASE VIA "SMART" TESTING/ANALYSES ON STRUCTURAL CAPABILITIES/LIMITATIONS OF DESIGN. (THROUGH CHARACTERIZATION OF DESIGN).
- IMPLEMENTATION OF IMPROVED MANUFACTURING PROCESS/CONTROLS
- SPIN-OFFS TO COMMERCIAL OR OTHER MILITARY PROGRAMS

EMPHASIS ON PREVENTATIVE APPROACH TO LIFE MANAGEMENT

DADTA ORGANIZATION

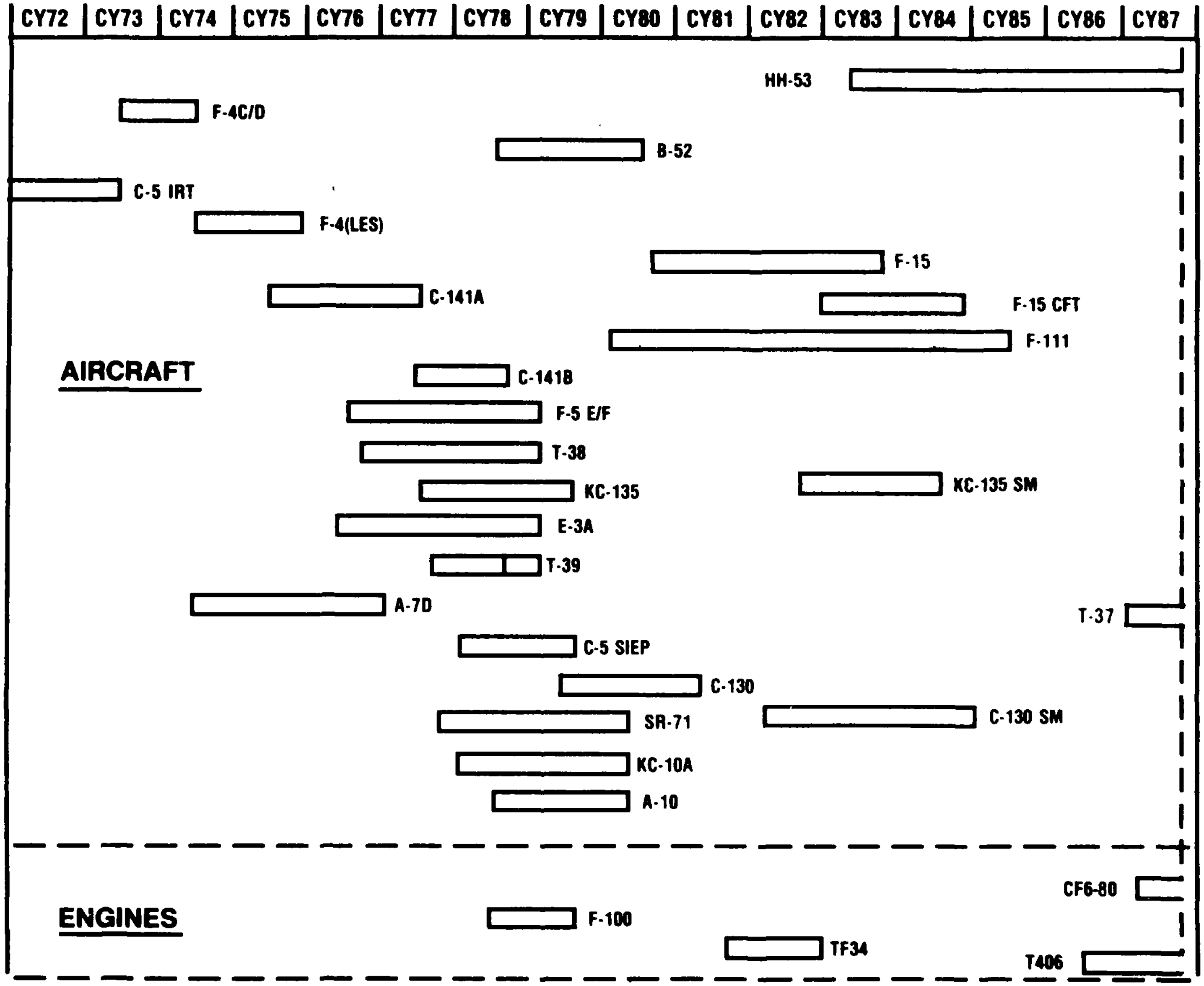


STRUCTURAL DURABILITY AND DAMAGE TOLERANCE ASSESSMENT SCHEDULE



EVOLUTION OF AS!P/ENSIP

DAMAGE TOLERANCE ASSESSMENTS



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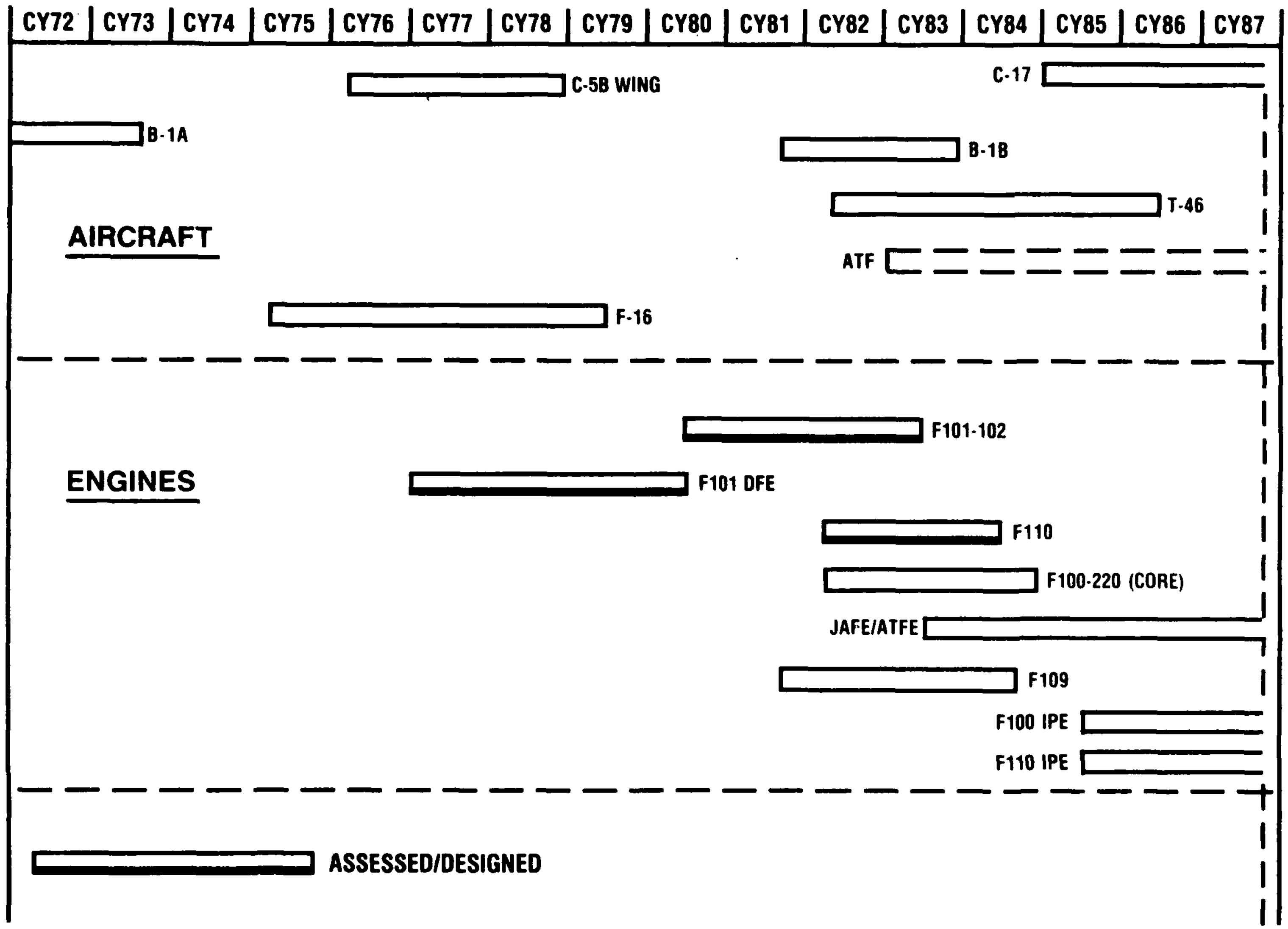
EVOLUTION OF ASIP/ENSIP
DAMAGE TOLERANCE DESIGNS



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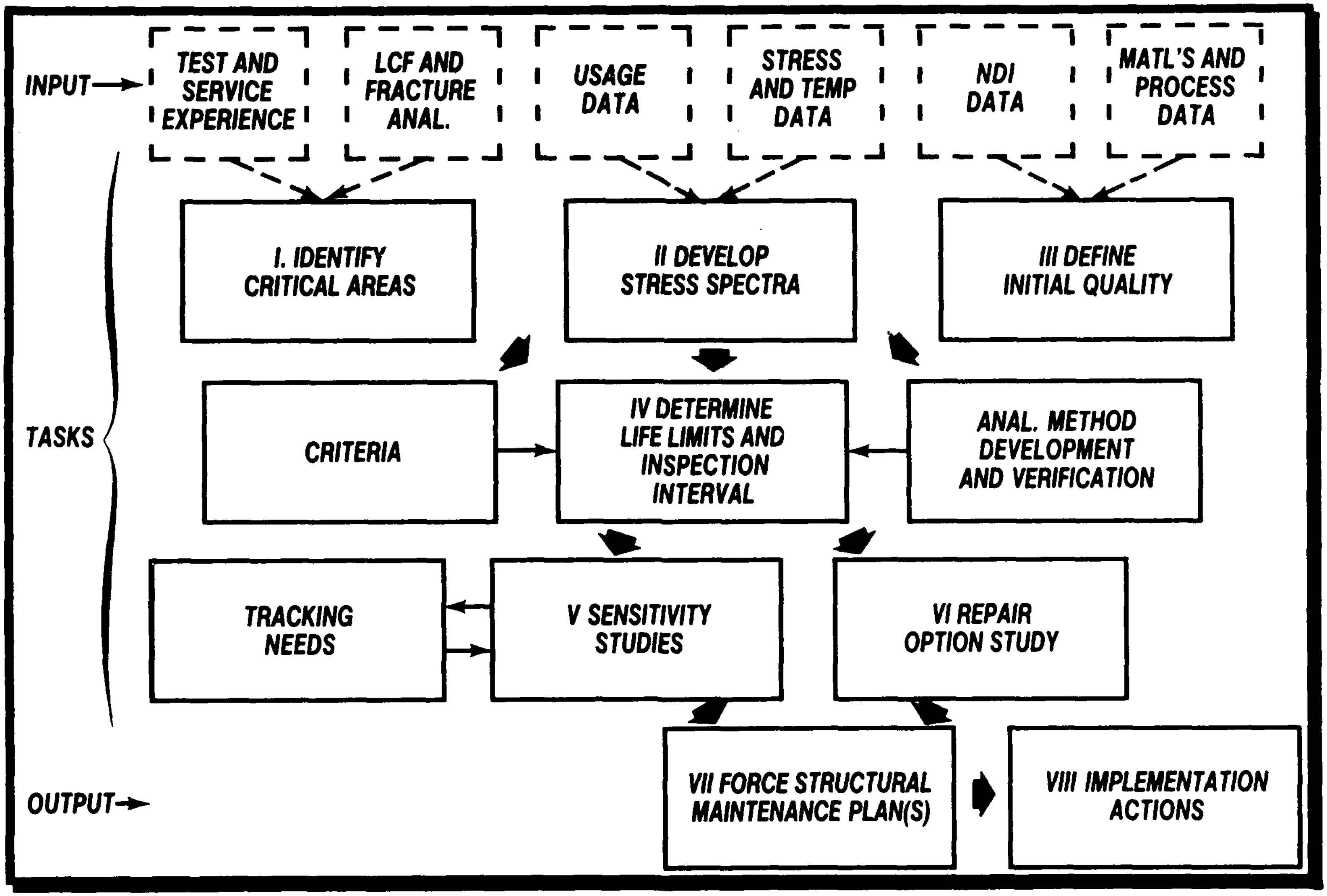
TECHNICAL APPROACH



TASK I – IDENTIFICATION OF CRITICAL AREAS

PURPOSE

- ESTABLISH COMPONENT CLASSIFICATION (FRACTURE/DURABILITY CRITICAL) BASED UPON CONSEQUENCE OF FAILURE
- DEFINE THE NATURE & EXTENT OF FOLLOW-ON EFFORT REQUIRED IN TASKS II-VI

APPROACH

- INDEPTH ANALYSIS REVIEW
  - HEAT TRANSFER/STRESS
  - LIFE
- EXPERIENCE REVIEW
  - FACTORY
  - FIELD
- FRACTURE SCREENING OF CRITICAL PARTS
  - FLAW SENSITIVITY
- PRIORITIZATION OF FOLLOW-ON ACTIONS

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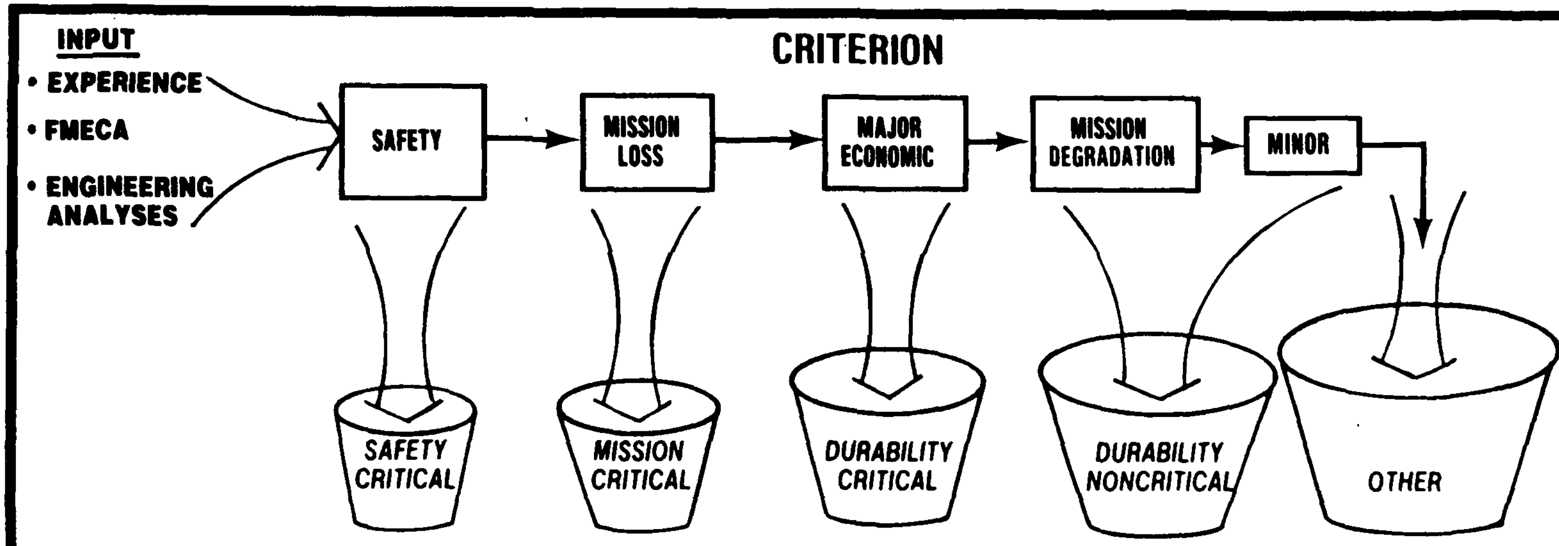
COMPONENT CLASSIFICATION DECISION LOGIC



CLASSIFICATION CATEGORIES	CANDIDATE SPECIAL PROVISIONS*						
	EXPANDED DESIGN CRITERIA	TIGHTENED PROCESS CONTROL	ENHANCED QUALITY CONTROL	ACCEPTANCE TESTING	RESTRICTED REPROCUREMENT	PARTS TRACKING	PREVENTATIVE MAINTENANCE
SAFETY CRITICAL	●	●	●	●	●	●	●
MISSION CRITICAL	●	●	●	●	●	●	●
DURABILITY CRITICAL		●	●	●	●	●	●
DURABILITY NONCRITICAL				●			●
OTHER							

* THE EXTENT TO WHICH THE SPECIAL PROVISIONS ARE APPLIED IS COMPONENT SPECIFIC

TASK II – STRESS SPECTRA DEVELOPMENT

PURPOSE

- PROVIDE THOROUGH UNDERSTANDING OF THE STRESS ENVIRONMENT FOR USE IN TASKS IV-VI

APPROACH

- UPDATE MISSION USAGE PROFILES & MIX AS APPROPRIATE
- CONDUCT THERMAL/MECHANICAL NOMINAL STRESS ANALYSES
- PERFORM COMPREHENSIVE ANALYSES OF STRESS CONCENTRATION AREAS
- DEVELOP DETAILED STRESS/ENVIRONMENT SPECTRA FOR USE IN LIFE ANALYSES & SENSITIVITY STUDIES

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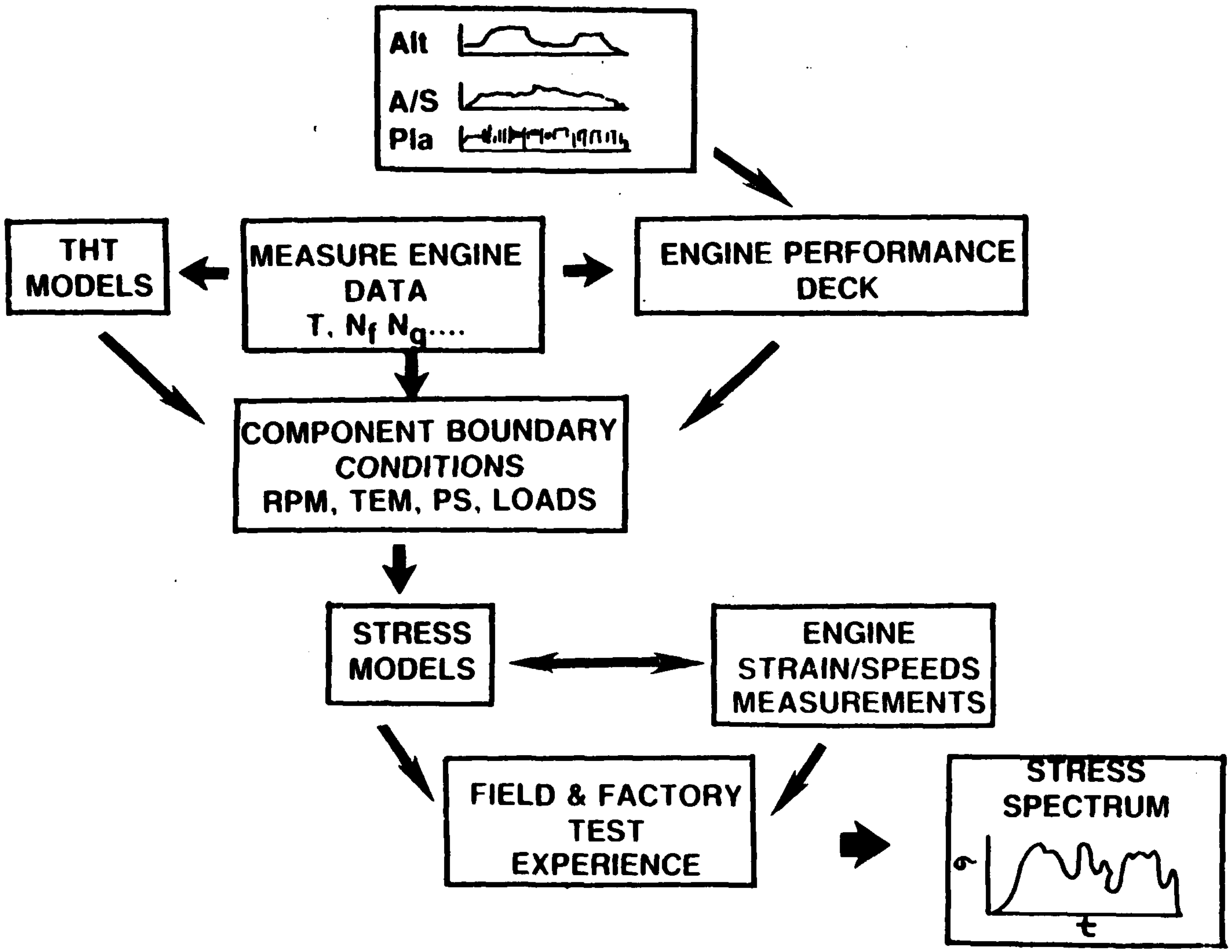
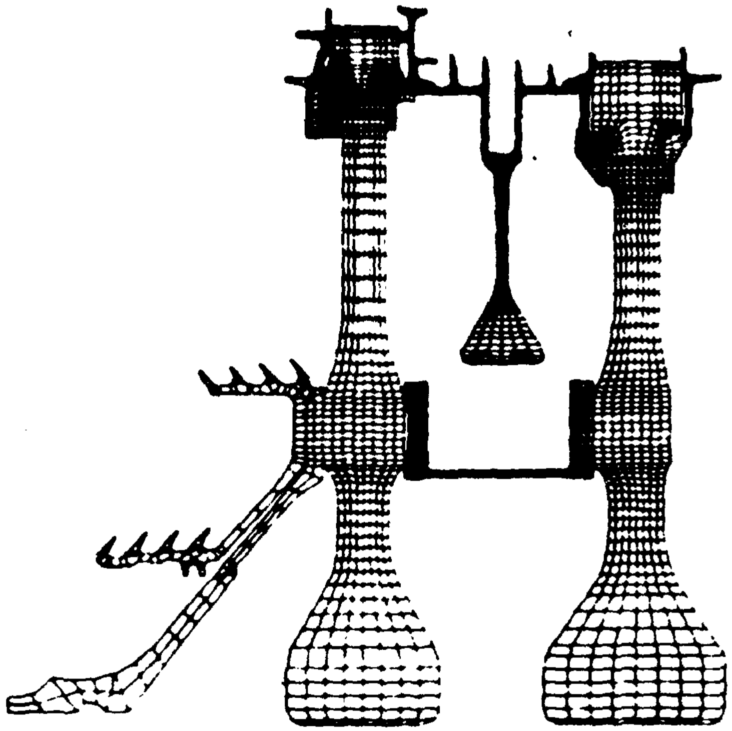
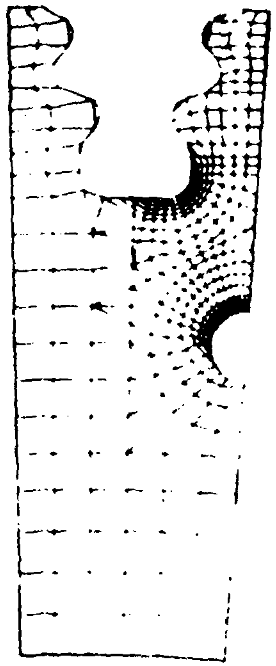
THERMAL AND MECHANICAL STRESS SPECTRUM DEVELOPMENT



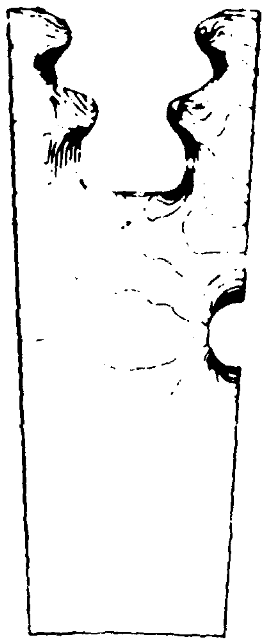
TF34 HIGH PRESSURE TURBINE ANALYSIS



2-D ANSYS MODEL



ISOPDQ ELEMENT MODEL



PRINCIPAL STRESS CONTOURS

STAGE 2 DISK OUTER BOLTHOLE - DOVETAIL

TASK III - HARDWARE QUALITY ASSESSMENT

PURPOSE

- ESTABLISH FLAW SIZES FOR USE IN TASK IV ANALYSES
  - INITIAL QUALITY (PRODUCTION)
  - RECURRING QUALITY (DEPOT)

APPROACH

- REVIEW NDI RELIABILITY/CAPABILITY DATA FOR PRODUCTION & DEPOT
  - ULTRASONIC
  - FPI/MPI
  - EDDY CURRENT
- EVALUATE MANUFACTURING PROCESS & CONTROLS
  - DRAWINGS
  - METHODS/PRACTICES
  - SCRAP/REWORK RECORDS
  - NDI RESULTS
- REVIEW EXPERIENCE

TASK IV – LIFE LIMITS AND INSPECTION INTERVALS

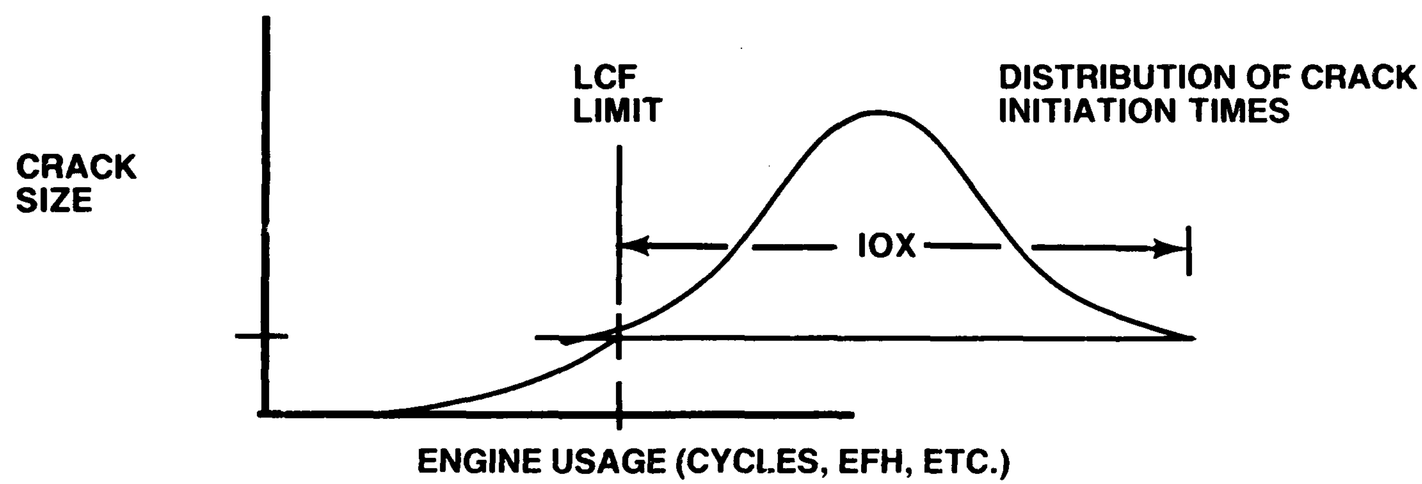
PURPOSE

- TO ESTABLISH THE LIFE LIMITS AND INSPECTION INTERVALS FOR CRITICAL COMPONENTS

APPROACH

- PERFORM LIFE ANALYSES (LCF, CREEP, STRESS RUPTURE, ETC) USING EXISTING/REFINED LIFE CODES/PROGRAMS
- CONDUCT CRACK GROWTH ANALYSES FOR FRACTURE CRITICAL COMPONENTS
- ESTABLISH USEFUL SERVICE LIFE & INSPECTION INTERVALS FROM RESULTS OF LIFE ANALYSES & QUALITY ASSESSMENTS
- IDENTIFY CANDIDATE COMPONENTS FOR LIFE EXTENSION THROUGH AN RFC APPROACH
- VERIFY ANALYTICAL PREDICTIONS THROUGH SPECIMEN, COMPONENT, RIG & ENGINE TESTING

LOW CYCLE FATIGUE (DURABILITY LIMIT) ANALYSIS



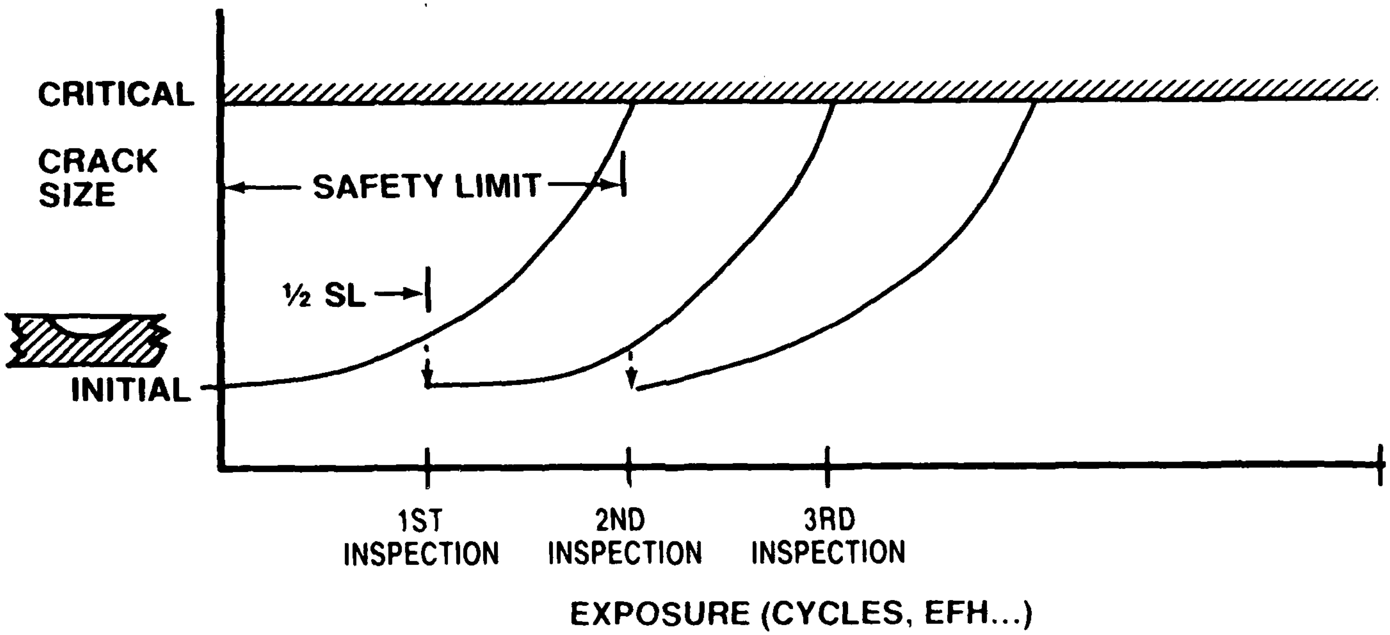
CRITERIA: LOW CYCLE FATIGUE (LCF) LIMIT BASED ON LOWER BOUND (-3σ OR 1/1000) DISTRIBUTION OF CRACK INITIATION TIME

ACTION: 100% PART REPLACEMENT AT LCF LIMIT

LIMITATIONS: NO RECOGNITION OF THE IMPACT THAT INITIAL DEFECTS CAN HAVE ON TOTAL PART LIFE (PART FAILURE CAN OCCUR PRIOR TO LCF LIMIT).

DISK TTT

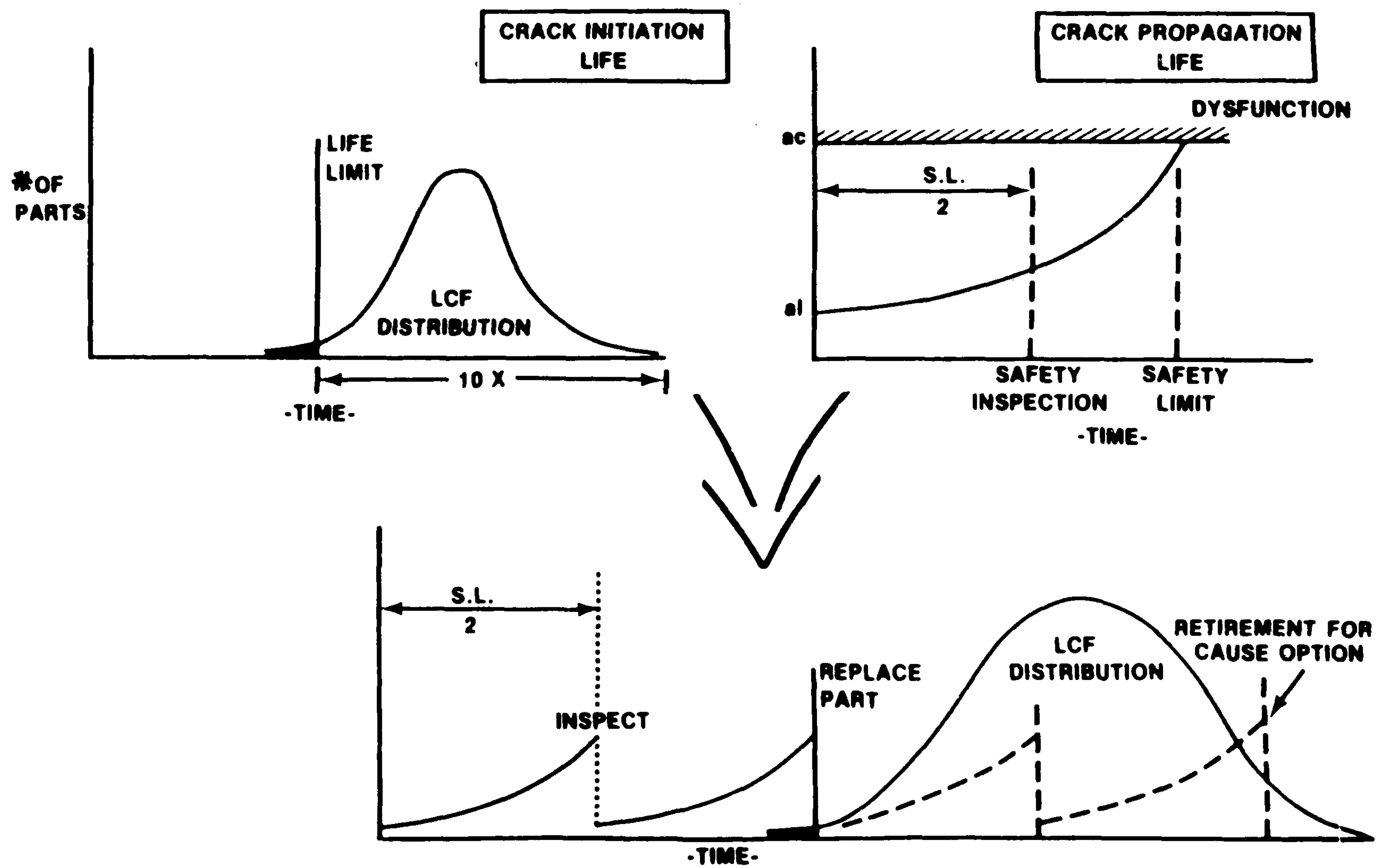
DAMAGE TOLERANCE (SAFETY LIMIT) ANALYSIS



CRITERIA: TIME FOR MAX PROBABLE FLAW TO GROW TO CRITICAL SIZE

ACTION: INSPECT OR PROOF TEST

LIFE MANAGEMENT PHILOSOPHY



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ANALYTICAL METHOD DEVELOPMENT & VERIFICATION

PURPOSE

- VERIFY METHODS THAT ARE EMPLOYED IN ESTABLISHING LIFE LIMITS & INSPECTION INTERVALS

APPROACH

- CONDUCT BASELINE MATERIAL CHARACTERIZATION PROGRAM FOR CRITICAL COMPONENT MATERIALS
  - TEMPERATURE & STRESS RATIOS
  - CRITICAL HARDWARE ALLOYS
  - HOLD TIME EVALUATION
- PERFORM VERIFICATION OF ANALYTICAL CODE/METHODS USING BASELINE DATA
  - CONFIGURED SPECIMENS
  - COMPLEX WAVE FORMS
  - APPLICABLE MATERIALS, TEMPERATURES, & STRESS/STRAINS

TASK V - SENSITIVITY ANALYSES

<u>PURPOSE</u>

- DETERMINE EFFECT OF PARAMETER VARIATIONS ON BASELINE LIFE LIMITS & INSPECTION INTERVALS

<u>APPROACH</u>

- SELECT KEY PARAMETERS FOR STUDIES
  - USAGE
  - FLAW SHAPE/SIZE
  - VIBRATION
  - TEMPERATURE
  - MANUFACTURING TOLERANCES
- CONDUCT PARAMETRIC STUDIES TO EVALUATE COMPONENT SENSITIVITY TO POSSIBLE VARIATIONS IN KEY LIFING FACTORS
- INCORPORATE RESULTS INTO LIFE MANAGEMENT PLAN

INFLUENCE OF INERT ATMOSPHERE (WITH AND WITHOUT RETARDATION) ON EMBEDDED FLAW CRACK GROWTH

LOCATION	BASELINE (MSSN. HRS.*)	BASELINE + RETARDATION	BASELINE + INERT	BASELINE + INERT + RETARDATION
FAN				
DISK BORE	1.0	-	1.15	1.23
COMPRESSOR				
STG. 10 BORE (NEW)	1.0	1.9	-	-
STG. 14 BORE (OLD)	1.0	1.8	-	-
STG. 14 BORE (NEW)	1.0	1.4	3.2	4.1
HPT				
STG. 1 BORE	1.0	1.5		
STG. 2 BORE	1.0	1.3		
OUTER T/C BORE	1.0	1.1		
LPT				
STG. 3 BORE	1.0	1.6		
STG. 4 BORE	1.0	1.7		
STG. 5 BORE	1.0	1.6	3.7	4.6
STG. 6 BORE	1.0	1.6	4.4	5.3

*RESIDUAL LIFE FROM A 1700 SQ MIL FLAW (.046" DIA)

TASK VI – MODIFICATION/REPAIR OPTION STUDY

PURPOSE

- IDENTIFY PREFERRED STRUCTURAL IMPROVEMENT OPTIONS FOR COMPONENTS WITH LOW LIFE LIMITS OR INSPECTION INTERVALS

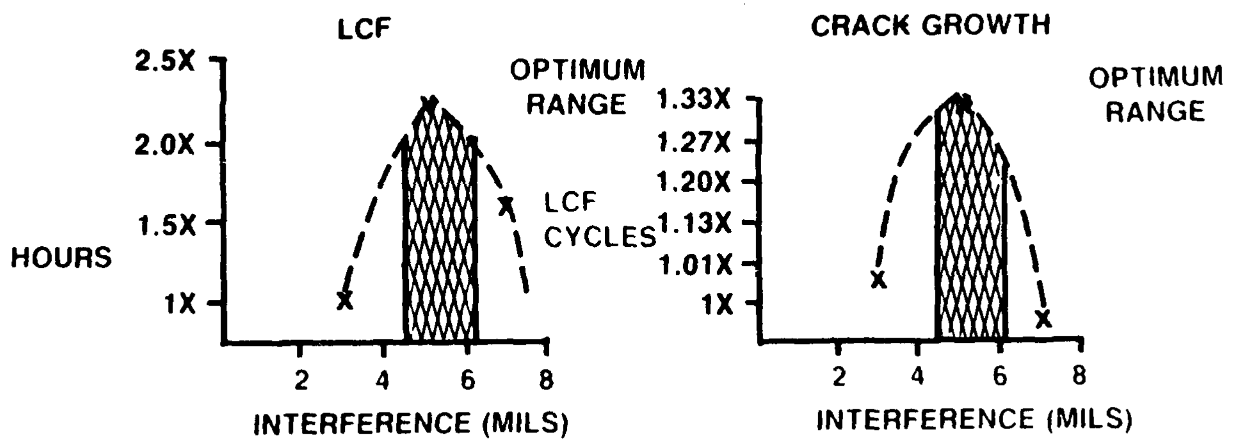
APPROACH

- DEFINE STRUCTURAL IMPROVEMENT OPTIONS
  - REWORK/MODS
  - SPECIFIC MATERIAL CHARACTERIZATION
  - RFC
- ESTABLISH POST MODIFICATION LIFE LIMITS & INSPECTIONS
- CONDUCT COST-BENEFIT ANALYSIS FOR USE IN SELECTING PREFERRED OPTION FOR INCLUSION INTO LIFE MANAGEMENT PLAN

B47228

TASK VI -- MODIFICATION/REWORK OPTION STUDY

- EXTEND COMPONENT ULITIZATION TO 8000 HOURS
- CRITERIA
  - LCC
  - DIFFICULTY OF INTRODUCTION
  - ENHANCEMENT AND RISK FACTOR



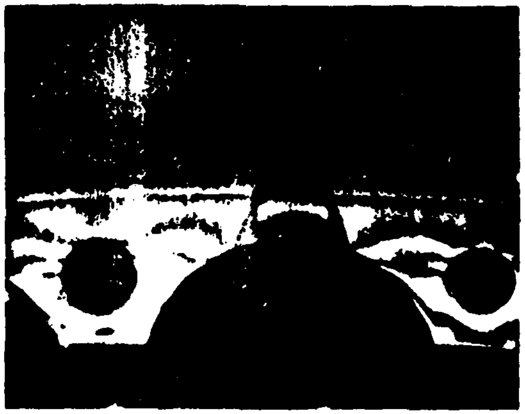
EXAMPLE - EFFECT OF INTERFERENCE FIT ON FAN DISK PIN HOLE LIFE

LPT Stage 4 Flange 3D Photoelastic Study

Typical Frozen Stress Isochromatic Fringe Patterns - As Is

$K_t = 2.2$

Airslot Rework



$K_t = 2.0$

Scallop Rework

$K_t = 1.9$

Full Rework

$K_t = 1.7$

TASK VII - STRUCTURAL MAINTENANCE PLAN

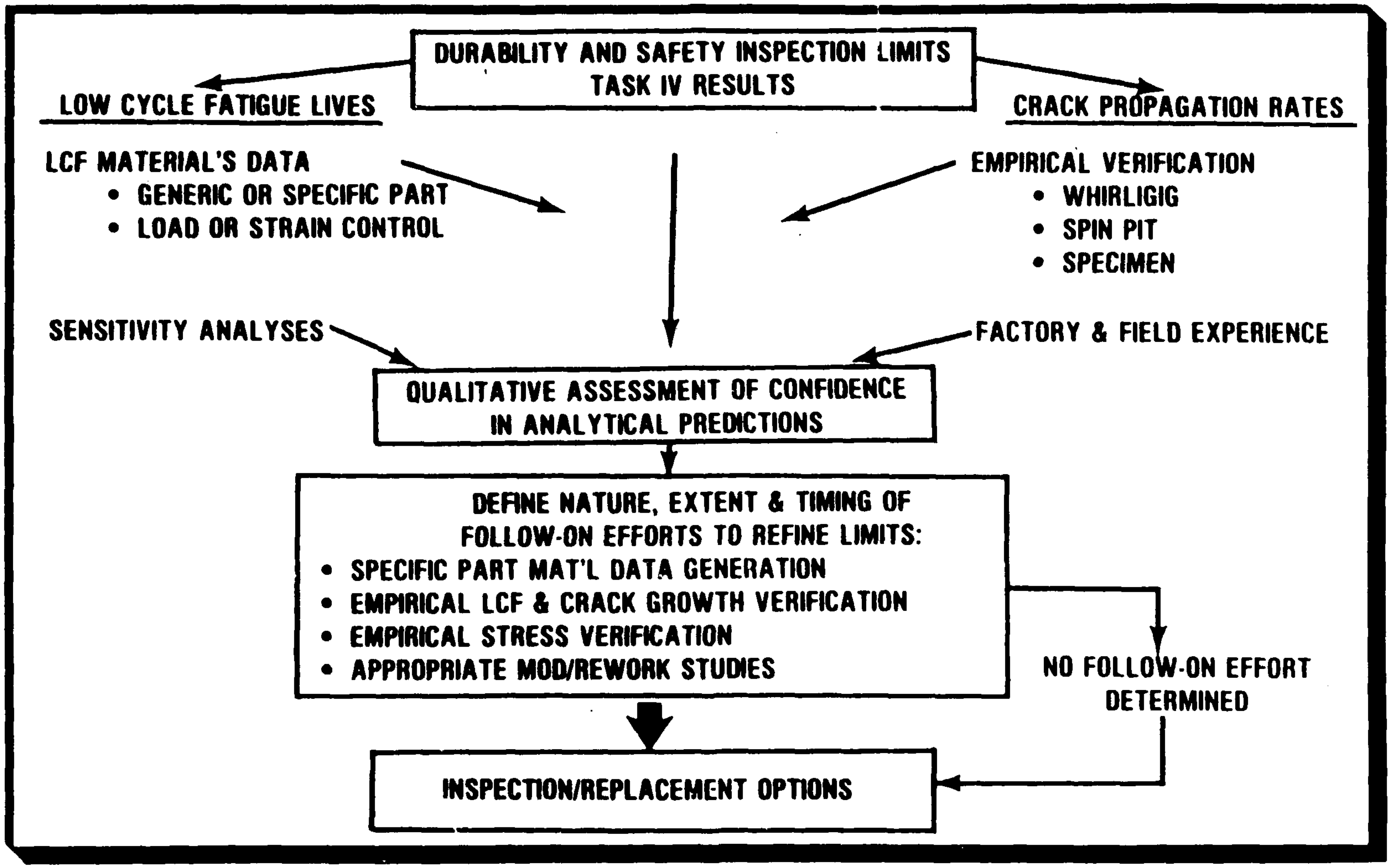
PURPOSE

- IDENTIFY ACTIONS FOR LIFE MANAGEMENT OF CRITICAL PARTS AND INTEGRATE RESULTS INTO AN OVERALL STRUCTURAL MAINTENANCE PLAN

APPROACH

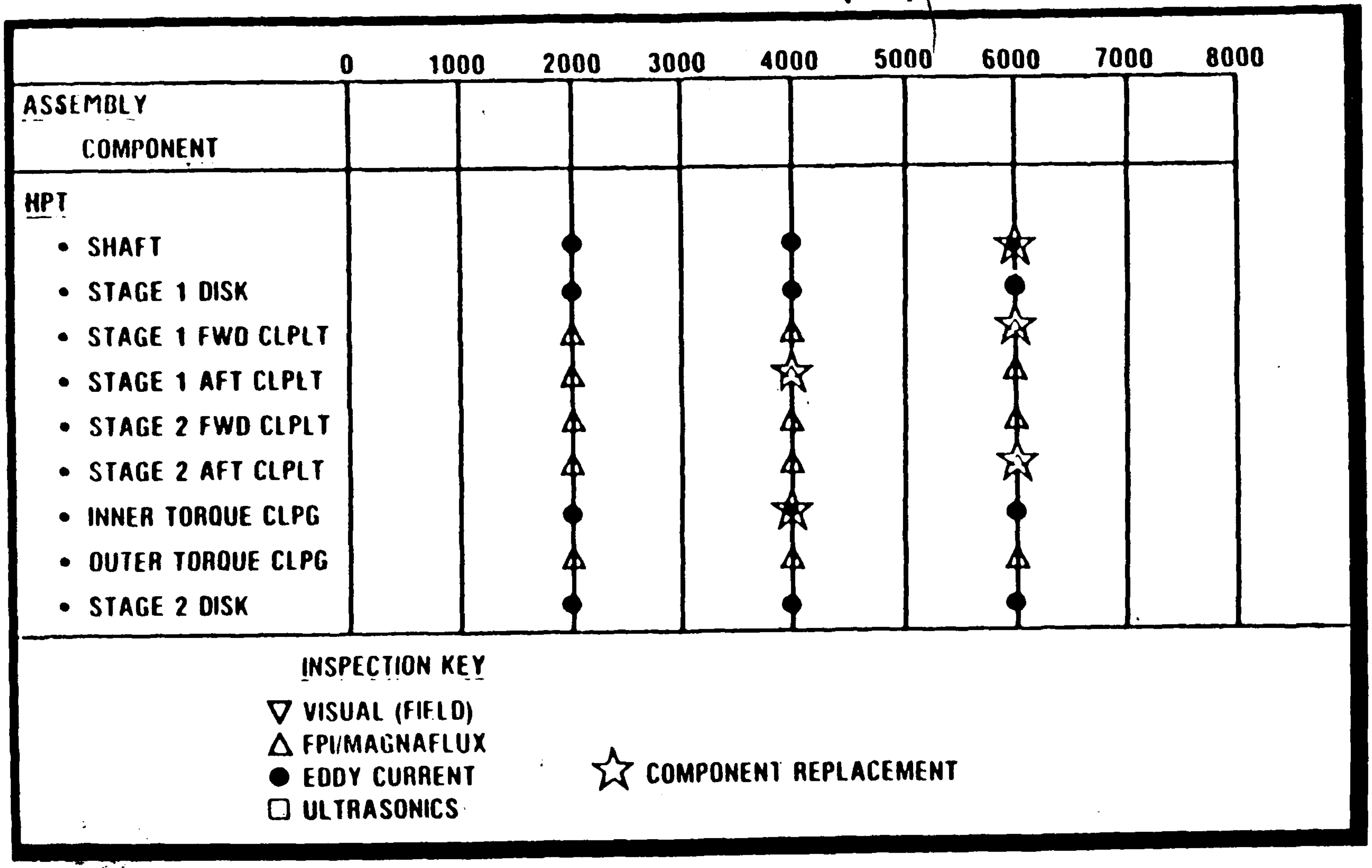
- DEFINITIZE REPLACEMENT TIMES FOR LIFE LIMITED COMPONENTS
- ESTABLISH INSPECTION REQUIREMENTS (METHODS, LOCATIONS, & FREQUENCY) FOR CRITICAL PARTS
- DEFINE PREFERRED MODIFICATIONS FOR LIMITED FEATURES/PARTS
- INTEGRATE ADDITIONAL ACTIONS INTO OVERALL MAINTENANCE PLAN
- DEVELOP COST ESTIMATES & PAYOFFS FOR RECOMMENDED ACTIONS

DADTA LIFE MANAGEMENT METHODOLOGY



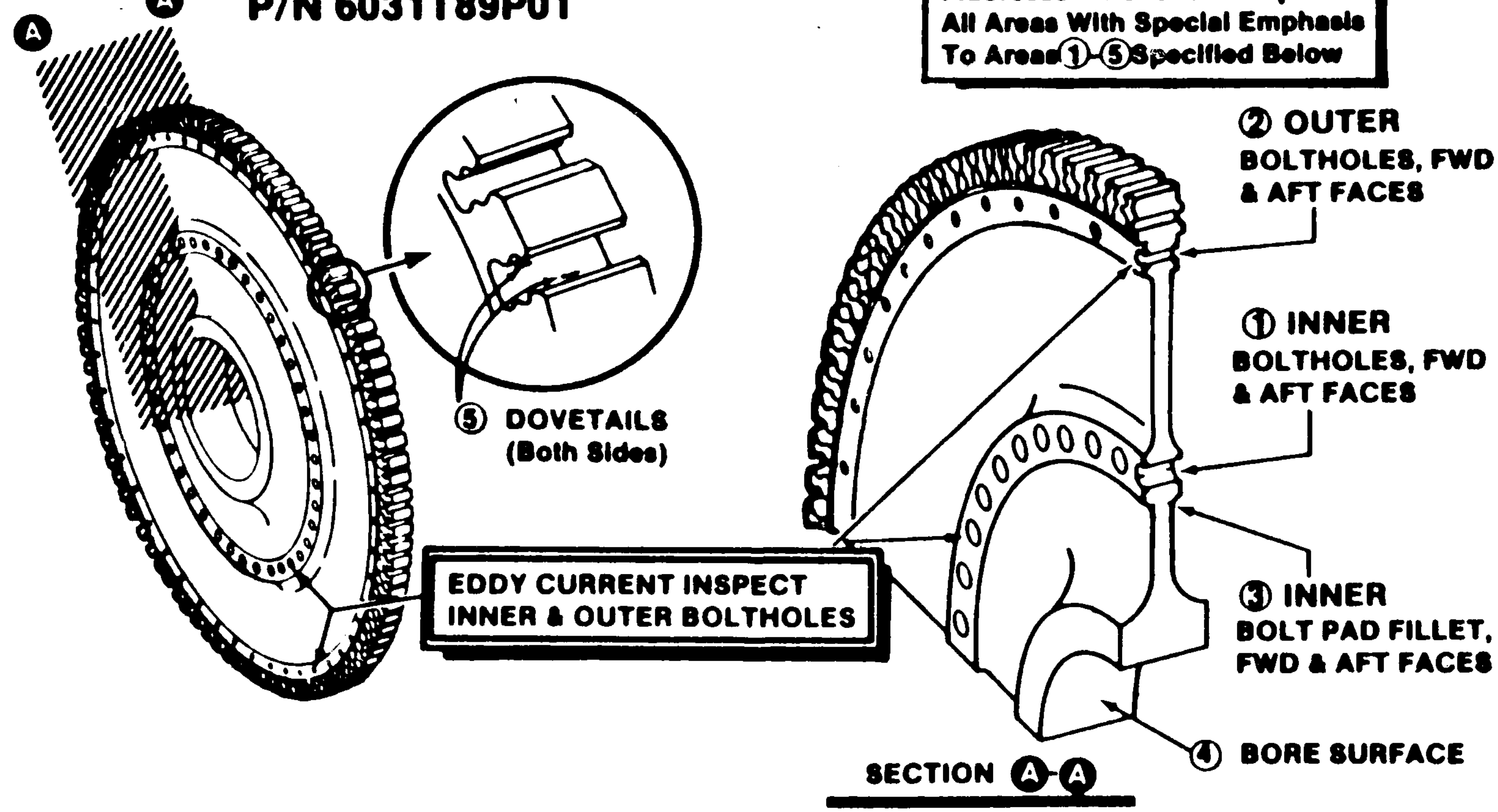
RECOMMENDED COMPONENT INSPECTION/REPLACEMENT PLAN

ENGINE OPERATING TIME (EOT)



TF34-100 INSPECTION SHEET

HPT STAGE 1 DISK:
P/N 6031T89P01



FOCUSED INSPECTIONS: LOCATIONS 1 - 5; FPI Group VI B (ZL30A*/ZR30A*/approved developer) *or equivalent
LOCATIONS 1 & 2; EDDY CURRENT

TASK VIII – IMPLEMENTATION ACTIONS

PURPOSE

- TO IDENTIFY THE ACTIONS & OPRs FOR IMPLEMENTING THE STRUCTURAL MAINTENANCE PLAN

APPROACH

- IDENTIFY AIR FORCE PROGRAM OFFICE, DEPOT, USER, & CONTRACTOR RESPONSIBILITIES & TASKS FOR IMPLEMENTING SMP
- ESTABLISH FUNDING REQUIREMENTS & SCHEDULE
- ORGANIZE TASK TEAM FOR WORKING IMPLEMENTATION

COST-BENEFIT ANALYSIS OF CONDUCTING STRUCTURAL ASSESSMENTS

FACTORS	F100	TF34	T406	CF6-80C2
• PROGRAM COST	(-)	(-)	(- -)	(- -)
• IMPLEMENTATION OF ENHANCED INSPECTION FOR CRITICAL COMPONENTS	(- -)	(-)	(TBD)	(TBD)
• IMPROVED SAFETY & MISSION CAPABILITY THROUGH INSTITUTION OF SAFETY INSPECTIONS	(+++)	(+++)	(TBD)	(TBD)
• ECONOMIC SAVINGS VIA				
LIFE LIMIT INCREASE	(+++)	(+++)	(TBD)	(TBD)
MODIFICATION / REWORKS	(+++)	(+)	(TBD)	(TBD)
RETIREMENT FOR CAUSE	(+++)	(++)	(TBD)	(TBD)

LEGEND

(+/–)	0 ⟶ ±$5M
(++/- -)	±$5M ⟶ ±$15M
(+++/- - -)	$\gtrless$ ±$15M

SUMMARY

- AIR FORCE DADTA PROGRAM IS AIMED AT EVALUATING & DOCUMENTING EXISTING DESIGNS TO AIR FORCE INTEGRITY PHILOSOPHY / CRITERION
- ALTHOUGH DIFFICULT TO QUANTIFY A PRIORI, ENGINE STRUCTURAL ASSESSMENTS PROVIDE A POSITIVE ROI OVER THE SYSTEM'S LIFE CYCLE
  - ENHANCED SAFETY
  - IMPROVED PRODUCTION QUALITY
  - REDUCED LCC VIA LIFE EXTENSION (REPAIRS/MODS/RFC)
- INSTITUTIONALIZATION OF THE INTEGRITY PROCESS FOR ALL AIR FORCE AIRCRAFT SYSTEMS, SUBSYSTEMS, & EQUIPMENT IS UNDERWAY. CONTROLLING REGULATION (AFR 80-13) SPECIFICALLY ADDRESSES APPLICATION OF INTEGRITY PROCESS TO "ENGINES USED, BUT NOT ORIGINALLY DEVELOPED BY THE AIR FORCE"

THE AMERICAN SOCIETY OF MECHANICAL ENGINEERS
345 E. 47 St., New York, N.Y. 10017

86-GT-38

Durability and Damage Tolerance Assessment of the TF34-100 Engine

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Deputy for Propulsion
Wright-Patterson AFB, Ohio
and
RONALD R. REINHOLD
General Electric Company
Lynn, Mass

ABSTRACT

The critical nature of the TF34-100 engine to the Air Force's A-10 Close Air Support weapon system made it important to obtain the best possible visibility of the engine's future structural maintenance needs and component life limits. Accordingly, an in-depth structural durability and damage tolerance assessment was performed on this engine by a joint Air Force/General Electric team.

Results of the assessment team's unprecedented analysis efforts culminated in a comprehensive Structural Maintenance Plan that identified both current and future maintenance actions necessary for insuring maximum flight safety. The plan entailed component inspection and replacement intervals, inspection systems, preferred modifications/reworks, and a life growth plan for extending the useful life of the TF34-100 upwards to 8000 A-10 mission hours.

This paper details the nature and extent of effort undertaken in conducting the 18 month structural assessment.

INTRODUCTION

The TF34-100, Figure 1, was among several engines/airframes recommended for Durability and Damage Tolerance Assessments (DADTA) by the Air Force Scientific Advisory Board in 1976. Air Force philosophy has moved towards conducting DADTA on aircraft engines as it did on airframe structures in the 1970-1978 time period. An Engine Structural Integrity Program (ENSIP) specification has been developed between USAF and industry to cover the structural requirements for future engine designs. Figure 2 summarizes the airframes and engines which have received similar assessments over the past decade. Also presented is a companion summary of aircraft and propulsion systems which have been designed employing a damage tolerance concept.

Figure 1 - TF34-100 Engine

The TF34-100 assessment was conducted over a period of 18 months by a joint Air Force/Contractor team. The primary objectives of the Durability and Damage Tolerance Assessment were to refine and update the part replacement times and define the inspection requirements necessary to insure structural integrity throughout the anticipated service life. This included identification of the specific components and locations for inspection, type of inspection (e.g., visual, fluorescent penetrant inspection (FPI), eddy current, etc.), estimated costs of inspections, and the logistics impacts. Also, economical modifications and/or repair options for components exhibiting low or marginal durability limits were established. This included investigating the technical feasibility of the options, estimating the probable costs and determining the post modification/repair life limits and inspection requirements.

The life limits, safety inspection requirements, modification/repair options, and post repair inspection requirements were integrated into an overall structural maintenance plan for the engine. Sensitivity studies were performed to determine the effects on component life limits and inspection intervals of different environmental/operational factors including partial cycle sequencing, flaw size variation, retardation, inert atmosphere, and analytical models/algorithms.

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Dusseldorf, West Germany—June 8-12, 1986

The structural assessment was comprised of eight major tasks. The technical approach followed and the inter-relationship of these eight tasks is shown in Figure 3. The following paragraphs provide a discussion of the efforts related to each of these tasks.

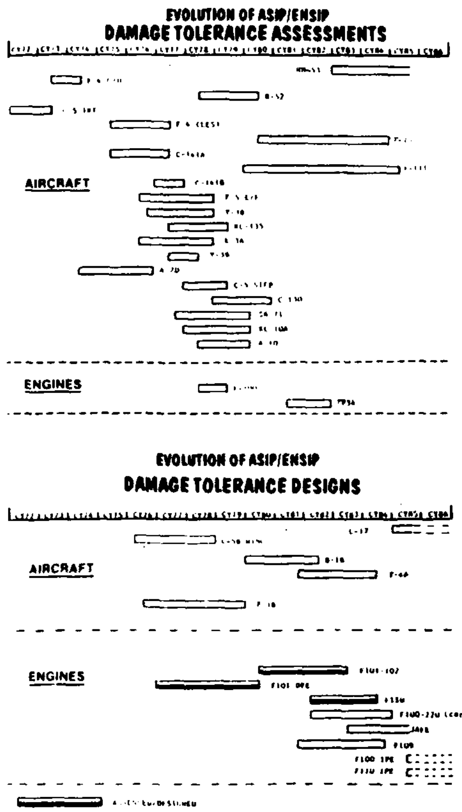

Figure 2 - Damage Tolerance Assessments/Designs

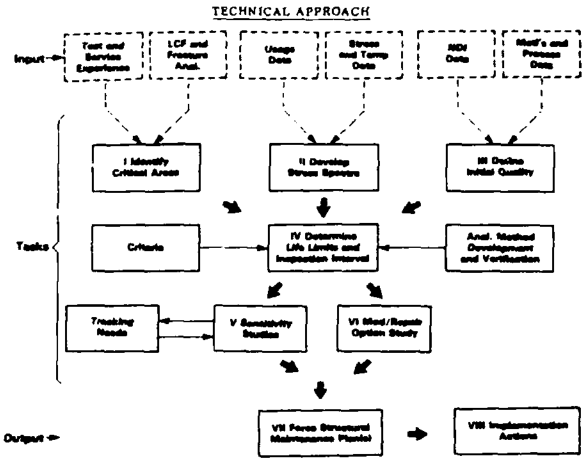

Figure 3 - DADTA Technical Approach

TASK I - IDENTIFICATION OF CRITICAL AREAS

Introduction

The purpose of this task was two-fold;

1) To review the various engine components and classify them as either FRACTURE CRITICAL or DURABILITY CRITICAL based upon the potential consequence of failure.
2) To define the nature and extent of additional work required in the subsequent tasks.

A detailed, disciplined approach was utilized in accomplishing this task and included the following key elements:

o Review of existing stress, heat transfer and life analyses (including materials data), as well as the life management plan for the individual components.
o Review of pertinent field and factory experience on each component.
o Establishment of FRACTURE or DURABILITY CRITICAL classification on a part-by-part basis.
o Evaluation of efforts underway or completed which address particular problem areas.
o Conducting initial fracture screening for critical areas of the FRACTURE CRITICAL components.
o Establishment of additional work (action) required by the DADTA team.

Criteria

A FRACTURE CRITICAL part is defined as a part which, if it failed, would likely jeopardize flight safety through single or progressive part failures.

A DURABILITY CRITICAL part is defined as a part whose failure could result in a significant maintenance burden but would not likely result in a flight safety problem. Engine experience was a primary consideration when judging part classification.

An in-depth review was undertaken of all major rotating components, pressure casings, frames, and mount load structures. The engine external controls and accessories were also evaluated for structural adequacy. All components containing internal pressure were scrutinized and preliminary fracture analyses performed. Leak before burst assessments and failure mode and effect analyses were conducted. Based on the culmination of engine experience and analysis the external accessories were classified as DURABILITY CRITICAL.

Crack experience on static structures and related Component Improvement Program (CIP) tasks were reviewed to ensure that potential problems were being addressed. This information, in conjunction with the aforementioned data summaries served to guide the team in establishing the nature and extent of effort required as part of the assessment.

In total, one hundred seventeen (117) components were reviewed or screened, forty-one (41) of which were classified as FRACTURE CRITICAL, and fifty-seven (57) were designated as requiring additional analyses. A summary by assembly/area follows:

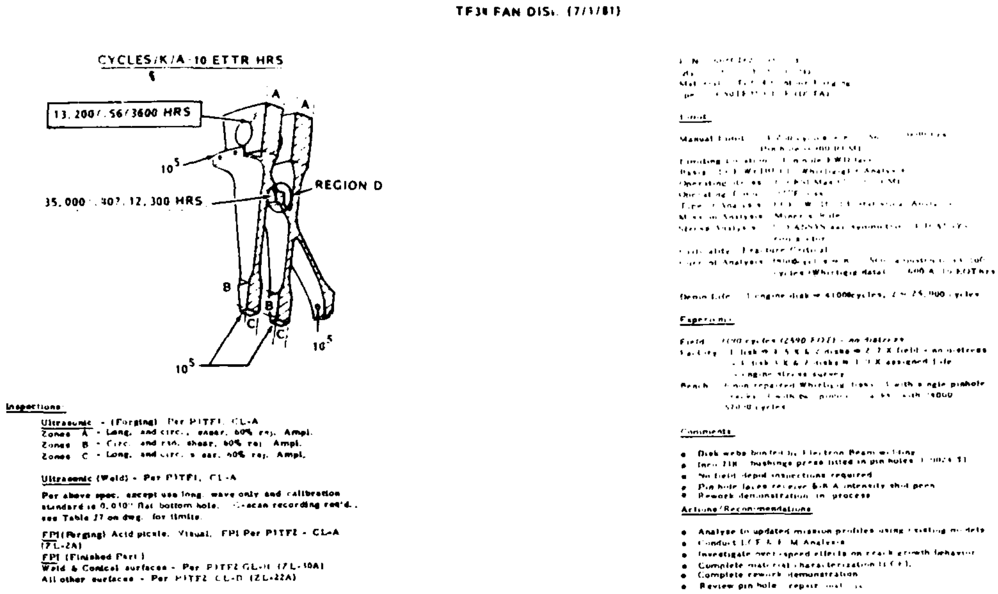

Figure 4 - Task I Summary Sheet for the Fan Disk

ASSEMBLY/MODULE GROUPING	PARTS REVIEWED	FRACTURE CRITICAL	ADDITIONAL ANALYSIS CONDUCTED
Fan Rotor	6	4	4
Compressor Rotors	41	8	17
High Pressure Turbine Rotors	13	7	10
Low Pressure Turbine Rotors	13	13	13
Static Structures	29	9	10
Controls and Accessories	15	--	3
TOTALS	117	41	57

Other rotating or static parts not on the FRACTURE CRITICAL list, as well as functional parts including seals, springs, etc., were classified as DURABILITY CRITICAL.

Figure 4 is a typical Summary Sheet compiled for each part. As shown, all relevant information was summarized for use in subsequent tasks.

Fracture screening summary sheets containing estimated critical crack sizes for high stress areas were also developed. This screening served to identify parts/locations especially sensitive to small defects thus guiding the selection of candidate parts for review in the hardware quality review undertaken in Task III.

The primary objectives of Task I were satisfied with the classification of components as either FRACTURE or DURABILITY CRITICAL and definition of analysis effort required for each component.

A complete reanalysis of the entire engine was dictated, with two separate and distinct analyses required for the compressor and LPT rotors due to the existence of two different field configurations. The reanalyses not only encompassed the traditional LCF but included the introduction of a fracture mechanics methodology. Updating existing heat transfer analyses, refinement of stress models, updating materials data, and developing representative A-10 mission profiles were undertaken as part of the assessment team's activities.

TASK II - STRESS ENVIRONMENT SPECTRA

Introduction

A prerequisite to establishing realistic Low Cycle Fatigue (LCF) lives and safety inspection intervals was the determination of an accurate stress spectrum. Essential ingredients to the overall stress spectrum development were the mission usage, engine thermal development, vibratory stress response, and detailed stress analyses. Pertinent details of these ingredients are provided in the following sections.

Mission Usage

A substantial operational data base was available for use in formulating the usage spectrum. The technical approach employed was unprecedented and used measured engine time history data for developing the usage profiles. The mission analysis efforts resulted in the definition of eight operational profiles, a scheduled maintenance cycle and associated mission mix.

Approximately thirteen percent of the aircraft in the A-10 fleet are equipped with MXU-533 tape recorders. In addition to recording time histories of normal airframe parameters, those installed after 1976 recorded engine Power Lever Angle (PLA) activity. The PLA time history was of prime interest however airspeed and altitude information

were also processed and used in defining engine operating characteristics. A sample flight time history is shown in Figure 5.

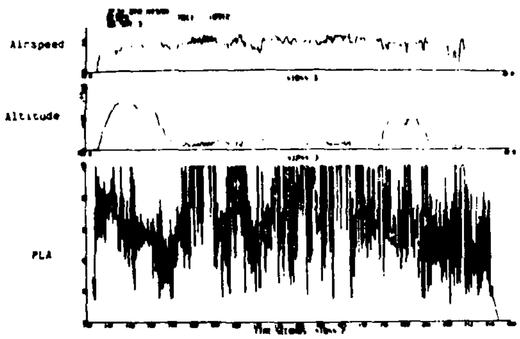

Figure 5 - Sample Time History

A computer routine was used in segregating the PLA activity into cycles of defined bands of extremities (peaks and valleys). The bands were set at 10° PLA intervals from idle (20°) to maximum PLA (100°) in order to allow for grouping of common cycle types. After examination of time history plots of many flights, three types of cycles dominated the operation. These types, when the bands for the peaks and valley were defined, encompassed essentially all cycles of significant range. Figure 6 illustrates these partial cycle types. The 0° to 100° to 0° PLA, (zero-max-zero or "LCF" cycle) was treated in addition to the three types shown.

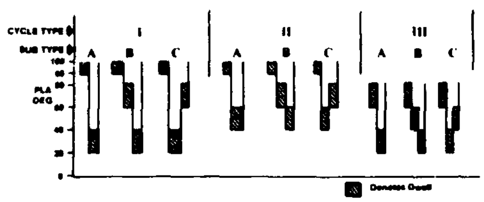

Figure 6 - Cycle Definition

The output of cycle counts and flight lengths were stored in a computer databank. A databank manipulation program grouped the counts by cycle type and determined the average number of occurrences of each cycle type per flight for each mission type.

An engine rotational speed-PLA relationship was needed for input to the life analysis program and was obtained from a statistical treatment of the performance deck values for various airspeeds and altitudes. The tapes were interrogated to determine the percentage of time spent in each airspeed/altitude corridor when the PLA was at maximum (90-100) and idle (10-20) settings.

The fan and core speeds associated with these airspeed/altitude corridors were tabulated and histograms of the maximum values encountered in each flight were developed for each base. A weighted maximum value was derived based on the number of engines at each base and used for every type I and III cycle. Values of speeds for partial PLA settings were also developed from the performance deck values.

The length of time (dwell) spent at the cycle peak, valley, and throughout the excursion was determined by examining the PLA time history plots of all flights. Although there was a large flight-to-flight variation in PLA activity and dwell times, typical profiles could be discerned. Typical profiles from each base/mission type were selected and the dwell times extracted graphically.

Each cycle type was subdivided according to when the acceleration/deceleration dwells occurred. The proportion of the three cycle subtypes were also determined. The dwell times were averaged for each base/mission type. The resulting eight mission profiles were created by taking the previously determined cycle counts, dwell times, and subtype proportionalities and placing them in a quasi-random sequence similar to the flight profiles from which they were statistically derived. A typical profile is shown in Figure 7. As with the dwell time determination, examination of many computer generated profiles allowed graphic determination of realistic sequencing; including proper placement of inactive (cruise) time. A sensitivity study treating cycle sequencing was conducted in Task V.

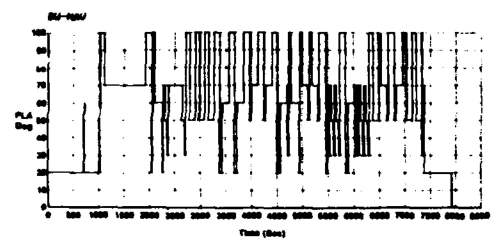

Figure 7 - Bentwaters RAFB Navigation Mission

Mission Mix

Seven predominant types of missions were identified in the A-10 application.

- Basic Flight Maneuvers (BFM)
- Conversion (CV)
- Ground Attack (GA)
- Ground Attack Tactics (GAT)
- Navigation (NAV)
- Surface Attack (SA)
- Surface Attack Tactics (SAT)

The percent of total operation of each mission type (the mission mix) was derived from the A-10 Operation Usage Program Quarterly Report. The mix was based on 185,000 hours of logged flying; essentially all A-10 operation to that date. It was determined through discussions with Tactical Air Command (TAC) operations personnel that past A-10 operation was representative of planned future activity. The tabulated mix was consolidated into the eight missions plus a maintenance cycle as shown in Figure 8.

Data on engine operation for scheduled maintenance, consisting primarily of trim and gun gas washing was

obtained from maintenance personnel at several bases. An unscheduled maintenance cycle was also defined through discussions with field representatives.

Mission	Hours/Flight EFH	Hours/Flight TOT	Cycles/Flight I	Cycles/Flight II	Cycles/Flight III	Operational (Percent)
BENTWATERS -- 45%						
BFM	1.42	1.87	5	10	10	10%
GAT	1.78	2.23	14	18	10	9%
NAV	1.75	2.20	8	12	17	15%
GA	1.62	2.07	9	14	8	16%
SAT	1.70	2.15	10	13	12	50%
	1.67	2.12	9.4	13.2	11.7	100%
MYRTLE BEACH - 26%						
BFM	1.42	1.87	5	10	10	12%
GAT	1.78	2.23	14	18	10	8%
NAV	1.75	2.20	8	12	17	17%
GA	1.62	2.07	9	14	8	27%
SAT	1.70	2.15	10	13	12	36%
	1.66	2.11	9.1	13.1	11.4	100%
DAVIS MONTHAN 29%						
SAT	1.72	2.17	16	20	11	36%
CV	1.62	2.07	14	17	10	36%
SA	1.72	2.17	12	19	10	28%
	1.68	2.13	14.2	18.6	10.4	100%
MAINTENANCE CYCLE						
		1.34	4	—	—	1 per 200 EFH

Figure 8 - Mission Analysis Summary

Mission/Base Integration

Low Cycle Fatigue (LCF) and crack growth analyses were conducted separately for all nine profiles. These results were mathematically combined based on the mission mix and the percentage of engines residing at each base.

Temperature Development

Component temperatures required in support of the stress and life analyses were generated using the Transient Heat Transfer (THT) analysis model. Measured engine and core thermocouple data provided the empirical base for calibrating the mathematical model.

The THT models for the different rotors were continuously updated/refined as new data became available. Figure 9 shows the locations of measured data employed in the update for the LPT rotor. Similar data was available for the HPT and two compressor rotors. Little measured thermocouple data existed for the combustor frame and HPT casing. As such, an instrumentation plan was defined and incorporated into the CIP program.

Vibratory Stress Review/Safety Limit Analysis

A review of both predicted and measured vibratory stress data was conducted for all fan, compressor, and turbine airfoils. Particular attention was paid to the fracture critical components (fan and stage 1 compressor blades). The results were directly employed in determining the vibratory margins and threshold crack sizes which could be tolerated without risk of failure. Stress levels utilized in conducting the life assessments were derived based upon normal engine operation.

The vibratory stresses were included in the safety limit calculation by allowing the as- -ed initial defect to grow until it reached a point where the vibratory ΔKEFF exceeded the threshold ΔK for the material. The rationale for truncating the safety limit at this point stems from the high rate of cycle accumulation and corresponding growth that would be expected to ensue during resonance response.

For all cases the steady stress corresponding to the speed(s) where vibratory response occurred were used for performing the crack growth analyses.

Adequate margins of safety existed for all airfoils during normal engine operation. In addition, the two fracture critical blades showed ample residual life for all operating conditions where vibratory stresses were considered.

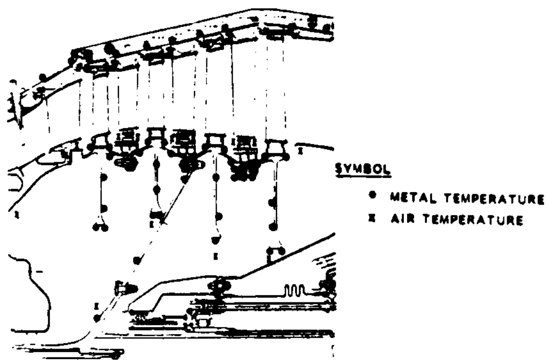

Figure 9 - Temperature Data Acquisition for Low Pressure Turbine

Stress Analyses

Stress spectrums were derived for fracture and durability critical turbomachinery and static cases/mount structures. A simplified stress cycle was defined and utilized in the control and accessory crack growth analyses.

A revised aircraft maneuver load spectrum, officially denoted Spectrum 3, was incorporated into the analysis. The translation of maneuver load spectrums into mount loads was accomplished through the application of a Mount Load computer program. Operating stresses in the forward and rear mount/load paths were obtained using a combination of empirical and analytical factors. For locations where measured data was available, empirical load-to-stress relationships were developed. These ratios, in conjunction with the load spectrum data and applicable analytical stress equations, served to define the stress spectrum for the primary areas of interest.

With the exception of the fan rotor assembly, nominal stress determination for the turbomachinery was accomplished using a shell analysis program. Nominal and concentrated stress values for the fan disk, blade, pin, and shaft were arrived at through a series of finite element models.

Thermal and mechanical nominal stress values were obtained at numerous points along the burst and chop

profiles. Rotor speeds, bolt loads, cavity pressures, and blade loadings were varied as a function of operating conditions in order to simulate the appropriate boundary conditions. Surface constraints between adjacent structures were modified to account for the relative growth experienced at different points in the transients.

Stress concentration factors and gradients for representative geometric details were arrived at through a combination of handbook, 2-D and 3-D finite element analyses. Displacement or stress fields as extracted from the axisymmetric finite element or shell analyses served as the boundary conditions for the localized 2-D and 3-D models. Once defined, Kts were treated as invariant quantities, independent of the thermal/mechanical make-up of the nominal stresses.

The HPT rotor analyses serves as an example of the nature and complexity of the various analyses undertaken. A 2-D axisymmetric finite element analysis of the entire rotor assembly was performed. The results supplemented the nominal shell analyses through definition of fillet Kts, gradients, and boundary conditions for use in refined Kt models of boltholes and dovetails. In an effort to better define the Kts and stress gradients for certain critical features, various 2-D and 3-D analyses were conducted. The stage 1 aft cooling plate and the inner bolthole region of the stage 2 disk were analyzed in 3-D. These analyses were undertaken in lieu of the simplified 2-D analyses due either to geometry complexity (cooling plate) or non-uniform loading (bolthole) making concentrated stress determination difficult. Boundary conditions for these models were derived from the axisymmetric analysis. A 2-D analysis was employed in defining dovetail Kts and stress gradients. Figure 10 depicts some of the HPT models employed.

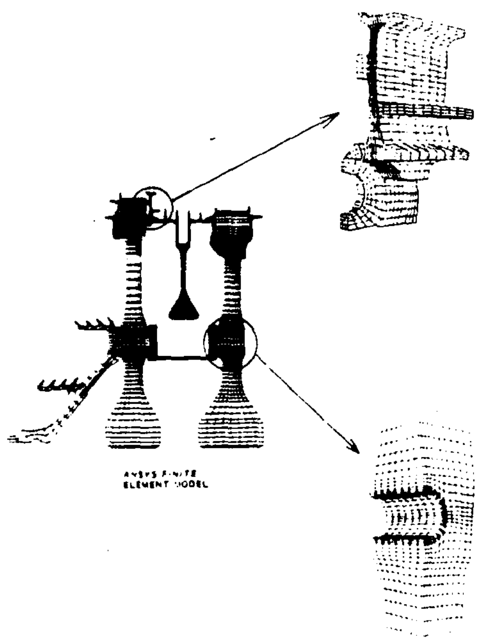

Figure 10 - HPT Finite Element Models

As previously discussed extensive use was made of finite element analysis in definir... ...e stresses and gradients for use in Task IV. An accurate determination of the stress state is fundamental to assessing a component's life (LCF or crack growth). This underlaying hypothesis provided the basis for conducting the many and varied analyses. However, it was recognized that even with the aid of significantly improved pre and post processors, 3-D analyses continued to be a luxury. Therefore, every effort was made to curtail the extent of 3-D work in an effort to allow sufficient resources for completing more rigorous analyses on a larger group of components/geometries. To supplement the analysis work for selected locations, verification activities were defined and initiated for certain high stress areas and included the following:

o 3-D Photoelastic test of LPT Stage 3-4 flange geometry. Initiated to empirically define the synergistic effect of two mutually exclusive stress risers; scallop and airslot. The test model was also designed to provide Kts for a variety of possible flange scallop/airslot reworks.

o Strain gage testing of the LPT Stage 4 disk. Initiated to provide an alternate source of verification for the combined effect of the scallop and airslot.

o Construction of a 3-D finite element model for a "representative" compressor loading slot. The model was developed during the course of the assessment; however, other priorities prohibited completion of the stress analysis. A residual task was defined and presented for follow-on completion.

o Identification of the fan drive shaft for follow-on analytical and empirical stress environment determination. An Engineering Program aimed at refining the stress spectrum was defined.

TASK III - QUALITY ASSESSMENT

The intent of this task was to establish a realistic flaw, defect, or fatigue crack size which would be utilized in subsequent tasks to establish Safety Inspection Intervals (SII's). The DADTA efforts were aimed at establishing realistic inspection intervals such that if a flaw of a barely non-detectable size was present in the worst location it would not have sufficient time to propagate to failure between inspections. Part replacement was still based on the conventional LCF or durability limit.

Efforts in this task concentrated on two separate areas:

a. Initial Quality Assessment . . . quality of the individual components as they left the factory.

b. Recurring Quality Assessment . . . defect detection capability as the components proceed through Depot inspection.

In the initial quality area the goals of this task were basically two-fold: (1) to establish for each critical location, the maximum flaw (defect) size which could go undetected as the part passes through production inspection, and (2) to assess the

likelihood that a flaw of a specific type, size, shape, location, and orientation exists in the material. For the recurring quality aspect, the intent was to ascertain the maximum flaw size possible to escape detection as the part passes through Depot inspections.

The approach taken was to review available Non Destructive Inspection (NDI) reliability and capability data for the appropriate environments, i.e., laboratory, production, field and/or depot. The inspection methods of interest included ultrasonic, eddy current (EC), Fluorescent Penetrant Inspection (FPI), magnetic particle and visual. In order to review the initial hardware quality, representative components from the major rotating assemblies were selected for extensive review of their respective evolution cycles. Process and engineering drawings, manufacturing methods, in-process inspections, quality controls on critical characteristics, and vendor NDI/NDE inspection results were reviewed. Finally, an assessment of the Depot's inspection capability was undertaken by team members with the assistance of NDI advisors from industry and government.

In general, the quality of TF34 hardware was assessed to be good. The key to establishing consistently good initial quality was to develop a sound process control system, qualify it and essentially "freeze" the process allowing deviation only with expressed approval after an extensive review. The key to maintaining quality control was through periodic audits of inplant processes as well as vendor operations to assure the qualified process was being followed.

Quantitatively the distribution of initial defects and the reliability of the inspection associated with finding these defects in TF34 parts could not be directly ascertained. The data required to make a quantitative assessment simply did not exist. Consequently, judgment was utilized in selecting flaw sizes for use in the crack growth calculations. This judgment was consistent with previously established precedence. In order to quantify the maximum benefit attainable through the use of enhanced inspection systems, smaller flaw sizes than can realistically be found by FPI were selected for initial computations.

As to recurring quality, the team determined that an upgrading of the depot's FPI inspection facility, while augmenting it with enhanced inspection systems (eddy current, ultrasonics, etc), was necessary to support the institution of the assessment team's Structural Maintenance Plan, developed under Task VII.

TASK IV - LIFE LIMITS AND INSPECTION INTERVALS

Introduction

Life limits and inspection intervals for the critical parts identified in Task I were established in Task IV. The results of Tasks II-III efforts served as the primary source of input to these life analyses. Development of Baseline Crack Growth Material Data and Configured Specimen Residual Life Testing were integral parts of Task IV activities. In addition, component empirical crack growth verification testing was initiated for several critical components. Testing was conducted through amplification of the on-going cyclic endurance Whirligig (atmospheric spin pit) programs. Extensive use was made of the Contractor's LCF and crack growth analysis programs. Refinements/improvements to these codes and procedures were incorporated as needed during the course of the assessment. The nature and extent of analysis activities for individual components were determined based upon classification category (Fracture or Durability Critical). Fatigue and fracture analyses were conducted on all fracture critical components. Limited fracture studies undertaken in support of Task VI efforts supplemented fatigue analyses for durability critical components.

Component Durability and Safety Limits

Residual life analyses complemented LCF updates for the majority of components considered by the Assessment Team. Selection of specific component locations was accomplished using the stress analysis results obtained in Task II. Revisions to the baseline list were made as results for selected geometric features revealed possible defect sensitivity in other areas. In total, over 287 analyses inclusive of LCF and crack growth were conducted on the turbomachinery (fan, compressor, high and low pressure turbines) components.

Consistent with the established Contractor's philosophy, part replacement limits were arrived at through conventional LCF analyses. However, for certain critical components where the LCF life was derived using generic[1] material data, operation beyond the lower bound predictions was considered. For these cases, a specific[2] part materials program was identified for generating the data required to support the life extension. Continued operation beyond the lower bound limit was predicated on the part receiving enhanced (eddy current) inspections at prescribed intervals.

Consideration was also given to adopting a Retirement for Cause (RFC) approach to life management for components exhibiting a reasonably long crack growth life but less than desirable LCF capability. The distinction between life extension and RFC hinges on whether the LCF life is anticipated to be greater than the 8000 hour goal. In the former case, a realistic projection based upon life improvement noted in generic-to-specific part LCF comparisons would suggest that the "actual" life was upwards to 8000 hours. For RFC, the anticipated improvements in predicted LCF life with specific part data are not expected to reflect the desired LCF capability. Thus, operating beyond this established LCF limit was considered. A prerequisite to adopting RFC was the institution of enhanced inspections.

1 Generic: Comprised of multiple sources of LCF data compiled from various commercial and military engine components/forgings.

2 Specific: LCF material data generated from unique TF34-100 component forgings.

Criteria

As previously indicated, two related but separate criterion were employed in setting structural limits. The first, and conventionally recognized as the component's replacement interval, is the durability or LCF limit. Defined as "the point in time where it is more prudent to replace the part than to continue in service" the LCF limit is traditionally associated with time to crack initiation. The second principle and normally thought of as life remaining from a given crack size to part dysfunction is the safety limit. Defined as "the time beyond which the risk of part failure is considered to be unacceptably high" the safety limit is related to the time for a small crack like flaw to propagate to failure. As applied to the baseline assessment activities this life was used in establishing the time between inspections and was an adjunct to the durability limit. Figure 11 graphically depicts the criterion as discussed.

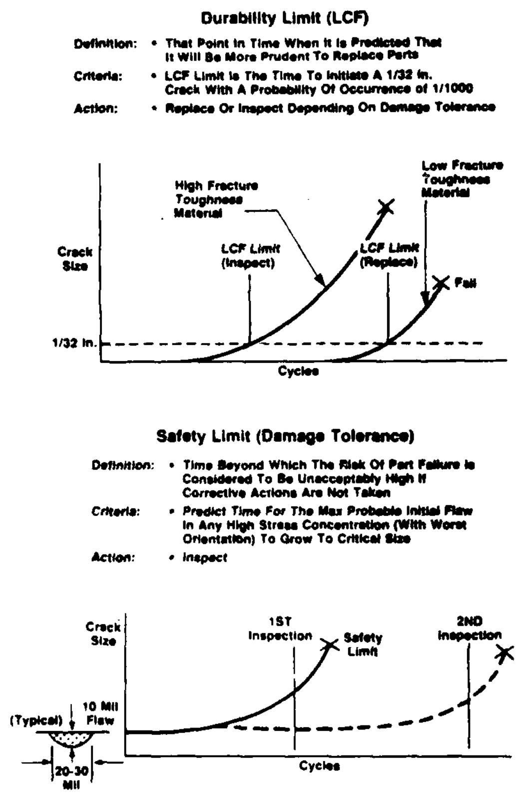

Figure 11 - Durability and Safety Limit Criterion

Embedded flaw residual life analyses were conducted for every bore where a corresponding surface flaw analysis was performed. A probabilistic approach to assessing the intrinsic defect areas was adopted in setting the initial embedded flaw size. The probabilistic analysis relies on the ability to statistically treat and translate the defect distribution as determined through examination of failed fatigue test specimens, to actual component hardware. Volume corrections are applied to account for the size differences between material test bars and actual component bore geometries. Residual life analyses for the derived flaw sizes approached the 8000 hour life goal. As such no replacement intervals were driven by an embedded flaw criterion.

Amplifications to this philosophy inclusive of life extension and RFC were treated on a case-by-case basis. Task VI details the rationale for applying these approacnes to specific components and lists the conditions under which adoption of such an approach was practical.

LCF Life and Crack Growth Prediction Procedures

Task II described the basic approach used in developing the stress spectrums. A pagoda rainflow cycle counting algorithm was used in extracting the stress cycles from the nine individual missions. Once compiled the stress cycles were modified for differences in temperatures between the minimum and maximum stress points and corrected, for use in the LCF analyses, to a stress ratio of A=1 using the Walker equivalent stress equation.(1) Minimum LCF life for each stress excursion was determined from a log life interpolation between applicable temperature load or strain control pseudo stress/strain range curves. In cases where specific part LCF data was unavailable, a "lower bound" limit was developed from the generic data base. Once established, the minimum life for each stress cycle within the mission was combined using Miner's Rule(2) to arrive at a life for the entire mission.

Life derivations for each of the nine missions were treated in a similar manner. A composite life, derived through a weighted mix of all missions for each base was developed.

Crack growth life for each critical geometric feature were derived treating the same stress spectrum developed for the LCF analyses. Since the local operating stresses in most life limiting areas exceeded the yield strength, emphasis was placed on obtaining the true stress gradients. Costs and timing prohibited the extensive use of elastic-plastic analyses to accurately predict local inelastic material response for the multitude of engine components. A simplified approximation using the Neuber equation(3) was adopted to estimate the real inelastic behavior of stress concentrations. Figure 12 depicts the results of an assessment of this simplified approximation.

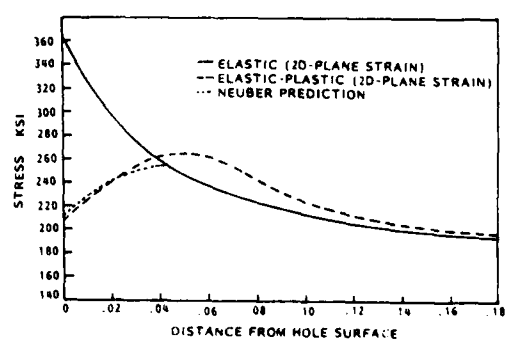

Figure 12 - Comparative Analysis of Plastic Stress Correction Methods

Stress intensity factor solutions for the majority of flaws treated in the crack growth analyses were developed using either influence function theory or a weight function method to account for complicated loading conditions. A specific stress intensity solution for the fan blade pinhole lug was developed using information supplied by the Lockheed-Georgia Company under Air Force contract.

The type of flaw analyzed in each location was determined by the geometry, stress condition, manufacture technique, and available experience. In order to encompass improvements in detection level through enhanced inspection systems (eddy current, ultrasonic, laser mapping, etc) all crack growth analyses were started from a crack depth of .005 inch. For locations where a .005 inch flaw caused the initial stress intensity to be below the material threshold, the crack growth analysis would be performed with a large enough flaw for propagation to occur. Typically, surface flaws with a 2:1 aspect ratio and corner flaws with a 1:1 aspect ratio were used in the analyses.

In locations such as flange boltholes, where cracking of the small ligament associated with the primary site did not cause failure, a crack growth analysis was performed on the secondary site using an initial 0.005 inch corner flaw. For disk bores which are susceptible to an embedded flaw type of defect, the initial flaw size was based on the ultrasonic inspection capability. The embedded flaw residual life calculations were also started with an 0.005 inch radius penny shaped flaw or the smallest size which would propagate under the DADTA defined missions. An alternative approach using a fracture analysis back calculation of failed fatigue test specimens was investigated. The results served to define the initial flaw sizes present in the material based upon volume and operating stress levels. Using this technique, a statistical estimate of the initial flaw distributions was possible. The flaw sizes determined by this method were used for residual life studies on selected critical bores.

Residual Life Methodology Verification/Development

A parallel comprehensive analytical and experimental residual life verification program was undertaken in support of the analysis performed by the assessment team. The program entailed conducting a series of analyses and laboratory testing on configured specimens. The nature and extent of work performed included:

- o Elastic-Plastic Analyses
- o Neuber Analyses
- o Crack Propagation Analyses
- o Cyclic Stress-Strain Curve Generation
- o Crack Growth Testing (Nickel and Titanium)
- o Metallurgraphic Examinations

Configured specimens representing typical turbomachinery features (i.e., boltholes, rabbets, dovetails/scallops, bores) were employed in verifying the analytical methodologies as they apply to residual life assessments.

The primary conclusion reached from the Verification efforts was that the residual life prediction methodology is a viable approach. Figure 13 represents a comparison between predicted and observed residual lives for the entire test matrix. As shown, nearly all results fall within a $\pm 2x$ scatter band. A tendency towards conservatism at short life (high stress) and nonconservatism at long life (low stress) is apparent. These results reflect good correlation considering the extent of analyses and tests conducted.

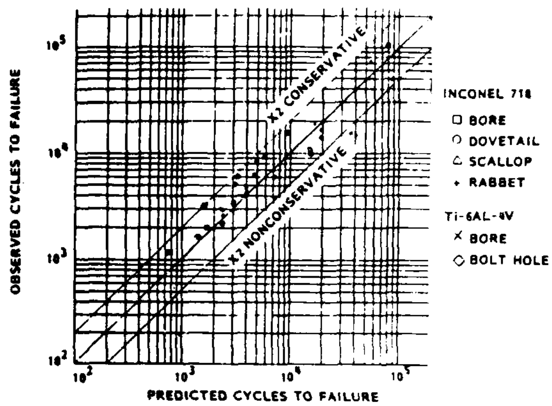

Figure 13 - Comparison of Predicted and Observed Residual Lives for Configured Specimens

Baseline Crack Growth Test Program

In an effort to accurately characterize the material response for the TF34-100 components an extensive material crack growth test program was conducted. The test program focused on defining the cyclic crack growth behavior of the fracture critical component materials (Inconel 718, Ti 6-4 DSTA, Ti 6-4 Annealed, and Ti 8-1-1). Variables within the test matrix included temperature, stress ratio, and tensile hold time. Variable ranges were selected to cover engine operating conditions while providing sufficient information for establishing an interpolative model. Table 1 summarizes the test matrix for each of the different materials characterized. Where appropriate, data from prior crack growth testing of TF34-100 unique materials supplemented the results obtained from the baseline program. The surface flawed K_B bar was selected as the specimen configuration for developing the baseline da/dn vs ΔK curves.

Crack Growth Data Reduction

The inferred crack depth versus accumulative cycle data were reduced to cyclic crack growth rate by the seven-point sliding polynomial technique recommended by ASTM. The stress intensity formulation consists of an elliptical flaw contained in an infinite solid, developed by Irwin, with appropriate modifications for specimen geometric features. The modified stress intensity solution in the crack depth direction is given by:

$$K_I = \sigma F_1 F_2 F_3 F_4 \sqrt{\pi a / \Phi}$$

where, F1, F2, F3 and F4 are correction factors which account for front surface, back surface, loss

Table 1 - Continuously Cycled Crack Growth Rate vs Stress Intensity Data

TEMPERATURE, °F	STRESS RATIO, A_σ					
	∞	2.0	1.0	.95	.6	.3
R.T. (70)	◇2		◇2	○2	○2	○2 ◇2
300°		△2	□2	△2 ▽2 ○2	△2 ○2 □2	△2 ○2 □2
600°		△2	□2	△2 ○2	△2 ○2 □2	△2 ○2 □2
800°		△2		△1 ▽2	△2	△2 ▽2
1000°		△2		△2 ▽2	△2	△2
1200°		△2		△2 ▽2	△2	△2

△ INCONEL 718 - COARSE GRAIN
▽ INCONEL 718 - FINE GRAIN
○ Ti 6-4 DSTA
□ Ti 6-4 ANNEALED
◇ Ti 8-1-1

Number within symbol indicates the quantity of specimens tested.

of load bearing area, and plastic zone. Φ represents the elliptical integral.

The sigmoidal equation was utilized for fitting a functional relationship to the da/dn vs ΔK data. The form of the equation is presented below:

$$\frac{da}{dN} = e^B \left(\frac{\Delta K}{\Delta K^*}\right)^P \left(\ln \frac{\Delta K}{\Delta K^*}\right)^Q \left(\ln \frac{\Delta K_c}{\Delta K}\right)^D$$

ΔK^*, and ΔKc represent the threshold and critical stress intensity ranges respectively. Stress ratio influences were treated using the Walker equation.(4) Figure 14 contains a typical sigmoidal curve fit to a set of corrected data.

Time Dependent Effects

Crack growth testing inclusive of a hold period at max tensile load was conducted to determine the time dependent effects on Inconel 718. Testing was performed at various temperatures and hold times in order to establish the temperature-dwell time dependency. Table 2 contains a summary of the nature and extent of test conditions considered. As noted, a single hold time test was conducted on Ti 6-4 DSTA. The results revealed that hold time effects at these conditions do not exist. As for Inconel 718, time dependent effects were observed at all three temperature conditions. Varying degrees of dependency ranging from insignificant to tenfold were evident. At 1000°F, the reduction in residual life was independent of starting crack size whereas at 1100°F and 1200°F the starting crack size played a major role in the resulting reduction factor. The factor was shown to increase with increasing crack length.

In order to utilize the test results, a procedure was developed to handle the time dependent effects. A cross plot of the residual life reduction factor vs tensile hold time, on a log-log scale resulted in a linear relationship being obtained. This relationship was employed in establishing a functional reduction factor for different hold times and temperatures. The resulting factor was indirectly applied to the residual life via a modification to the vertical translation coefficient, B, in the sigmoidal equation. This approach provided a simplified means of accounting for hold time effects. Further work is considered necessary to qualitatively and quantitatively determine the limitations of such an approach.

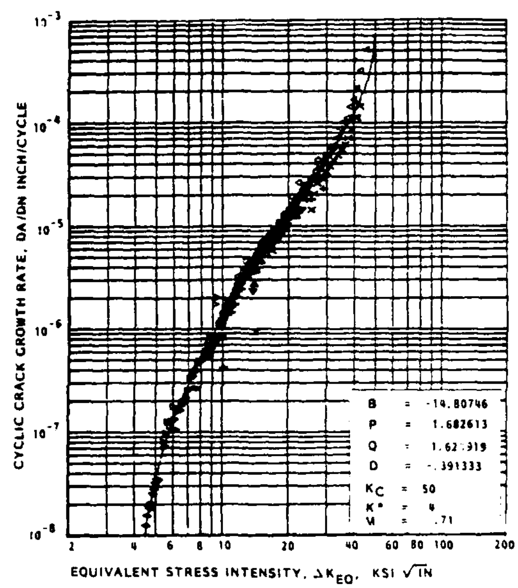

Figure 14 - Cyclic Crack Growth Rate versus Equivalent Stress Intensity for Ti 6-4 DSTA at Room Temperature

Table 2 - Hold Time Crack Growth Rate vs Stress Intensity Data

TEMPERATURE, °F	TENSILE HOLD TIME	
	90 SEC	300 SEC
600°	○1	
1000°	△1 ▽1	△1
1100°	△1 ▽1	△1
1200°	△1 ▽1	

△ INCONEL 718 - COARSE GRAIN
▽ INCONEL 718 - FINE GRAIN
○ Ti 6-4 - DSTA

Number within symbol indicates the quantity of specimens tested.

Component Empirical Crack Growth Verification

As an adjunct to the cyclic endurance whirligig testing for the major assemblies; Fan, Compressor, High Pressure Turbine and Low Pressure Turbine, crack propagation data generation was initiated for several critical geometric features. The approach consisted of monitoring crack growth from either cyclic induced (natural LCF) or fabrication (EDM notch) cracks in the selected geometric details as listed below:

- o Fan disk and blade pinhole
- o Compressor disk dovetails
- o Compressor shaft and spool boltholes
- o HPT disk boltholes
- o LPT disk and seal scallops and boltholes

Results obtained on the fan disk and blade pinholes reflected good agreement between the analysis and empirical results. Post mortem analyses of the fracture surfaces were conducted for ascertaining the crack front shape. This information was employed in back predicting the crack growth once a precise definition of the initial crack geometry was established.

The test data generated from the Baseline program provided the necessary information for developing a strong interpolative da/dn vs ΔK model. With the exception of time dependent behavior, good agreement was obtained between the empirically based model and actual test results. The hold time data provided a unique situation relative to the treatment and application of the results. The most significant conclusion reached after an exhaustive review of the hold time data was that every attempt should be made to keep the stress intensity of an inspectabe flaw below threshold when operating above 1100°F. Finally, the sigmoidal relationships as developed in the Baseline program provided the material foundation upon which all residual life analyses undertaken in Task IV were based.

TASK V - SENSITIVITY STUDIES

Introduction

While conducting the various analyses associated with Tasks II and IV a series of sensitivity studies were undertaken. These studies provided the information necessary to ascertain the influence of different variables/analysis techniques on component life. A summary of the specific topics and task(s) which they support is listed below:

TOPIC	SUPPORTING TASK
Partial Cycle Sequencing	II
Assessment of Thermal Stress Estimating Algorithm Accuracy (COMALL Validity Assessment)	II & IV
Influence of Initial Flaw Size on Partial Cycle Effective Damage	IV
Miner's Rule-Complex Cycle Residual Life Comparative Study	IV
Crack Growth Retardation Study	IV
Influence of Inert Atmosphere on Crack Growth	IV
Sensitivity of Residual Life to Initial Flaw Shape (Aspect Ratio)	IV

Partial Cycle Sequencing

A study aimed at determining the sensitivity of stress spectrum development to partial cycle (Type I, II and III) sequencing was conducted. The study, prompted by the DADTA team's quasi-random ordering of partial cycles within a mission, consisted of arbitrarily repositioning the cycles within a baseline mission and performing stress and LCF life analyses. Three thermally sensitive components were analyzed to both cycle sequences. A review of the results revealed less than a 5% variation in predicted LCF life. Therefore, the original baseline missions, as presented in Task II, were utilized for all stress and life analyses.

COMALL Validity Assessment

To establish confidence in the Complex Mission Analysis for LCF Life (COMALL) program a comprehensive comparative study was conducted. The study, structured to determine the accuracy of the transient thermal stress calculation algorithm, consisted of performing back-to-back stress/life analyses. Discrete stress analyses were conducted at selected time steps encompassing both sides of the COMALL V predicted extremums. The discrete step results, combined with appropriate material curves, provided the basis for direct comparison to COMALL V LCF life predictions. The assessment served to indicate that although exact agreement was not obtained, sufficient accuracy was determined to exist to justify the use of the COMALL program for the bulk of the stress/life analysis work.

Influence of Initial Flaw Size on Partial Cycle Damage

A study was undertaken to establish the sensitivity of residual life partial cycle damage ratios (K factors) to the assumed initial flaw size. The study concentrated on defining the relative damage of the Type I, II and III cycles as a function of the assumed initial flaw size. The results of a residual life analysis on the fan disk pinhole revealed that a 10% difference in effective damage (Type I, II and III cycles combined) occurred between the two initial flaw sizes (.005" and .015") considered.

This difference is attributed to the effect of a threshold ΔK regime on crack growth for the smaller K's of the partial cycles. As the initial flaw size is decreased the calculated ΔK's for the partial cycles approach the threshold region and their contribution to total crack growth becomes proportionally less.

With the majority of initial inspection intervals established from a 0.015" defect, adoption of the corresponding K factor was considered appropriate. For locations where either the initial or recurring flaw size is less than 0.015" the K factor, as developed from the 0.015" case was employed. This is conservative since the relative damage, KEFF, is greater for the larger defect size.

Miner's Rule-Complex Cycle Residual Life Study

The potential error introduced by combining the residual life capabilities of different partial cycles using Miner's Rule was investigated. A study comparing the simple cycle Miner's Rule summation

crack growth life calculation to the complex mission approach (cycle-by-cycle) revealed less than a four percent difference between the two methods for the fan disk pinhole. As expected Miner's Rule provided slightly longer predicted lives. Both approaches were therefore considered interchangeable provided care was exercised when performing analyses using the simple cycle technique to assure that the partial cycle ΔK was above ΔK threshold. The majority of locations analyzed employed the more rigorous cycle-by-cycle approach.

Crack Growth Retardation Study

In order to obtain the life benefits/safety margins possible through the consideration of retardation a series of residual life studies were conducted. The Willenborg model,[5] modified with an appropriate empirical material factor, was employed in the retardation studies. Table 3 lists a summary of results obtained for selected critical components. As anticipated, minimal benefit was obtained for the stage 2 and 3 dovetails, where the maximum stresses from the partial and major cycles were nearly equal. The tabulated values represent composite residual lives derived from a weighted average of individual base-mission analyses.

Although crack growth retardation is known to be dependent upon mission cycle sequencing no effort was made to interrelate the various missions prior to conducting the analyses. Therefore, the numbers presented were taken to represent possible "levels of improvement" gained through retardation.

Table 3 - Effects of Retardation for Critical Components

LOCATION	RESIDUAL LIFE MISSION HRS. (FROM .015" FLAW) BASELINE	RETARDATION	% INCREASE
OLD COMPRESSOR			
STG. 2 DOVETAIL	1.0	1.11	11.0
STG. 3 DOVETAIL	1.0	1.1	10.4
STG. 9 BOLTHOLE	1.0	1.83	53.3
STG. 10 SCALLOP	1.0	1.2	20.3
NEW COMPRESSOR			
STG. 10 LOAD SLOT	1.0	1.36	38.1
STG. 14 BORE (SURF)	1.0	1.34	34.7
HPT			
INNER T/C AFB B.H.	1.0	1.41	40.9
STG. 2 DISK OUTER B.H.	1.0	1.46	46.1
STG. 2 DISK FWD FILLET	1.0	1.89	89.3
LPT			
STG. 3 SCALLOP	1.0	1.37	37.1

Influence of Inert Atmosphere on Crack Growth

The baseline residual life analyses for embedded flaws were performed using da/dn data generated in laboratory air environment. To assess the influence of an inert environment on embedded flaw crack growth, estimated curves were derived from the baseline air data. The basis for the estimated curves was a review of available inert and air environment da/dn curves for similar materials.

Table 4 contains the results of residual life analyses conducted on disk bores of the fan, compressor, and turbine assemblies. As shown, results have been compiled for a variety of analyses (baseline, retarded, inert, and inert + retarded) performed on the different locations. A reference

Table 4 - Influence of Inert Atmosphere on Embedded Flaw Crack Growth

LOCATION	BASELINE (MSSN. HRS.*)	BASELINE + RETARDATION	BASELINE + INERT	BASELINE + INERT + RETARDATION
FAN				
DISK BORE	1.0		1.15	[illegible]
COMPRESSOR				
STG 10 BORE (NEW)	1.0	[illegible]		
STG 14 BORE (OLD)	1.0	[illegible]		
STG 14 BORE (NEW)	1.0	[illegible]	1.7	[illegible]
HPT				
STG 1 BORE	1.0	[illegible]		
STG 2 BORE	1.0	[illegible]		
OUTER T/C BORE	1.0	[illegible]		
LPT				
STG 3 BORE	1.0	[illegible]		
STG 4 BORE	1.0	[illegible]		
STG 5 BORE	1.0	1.6	[illegible]	[illegible]
STG 6 BORE	1.0	[illegible]	[illegible]	[illegible]

*RESIDUAL LIFE FROM A 1700 SQ MIL FLAW (.046" DIA)

defect size of 1700 sq mils was selected for these studies. Predicted improvements in residual life range from less than 15% to greater than 530% depending upon location and conditions treated. These results were indirectly employed in the development of the Task VI intervals and replacement times.

Sensitivity of Residual Life to Initial Flaw Shape (Aspect Ratio)

At the outset of the assessment program initial flaw geometries were established based primarily upon past experience. However during the course of the assessment, interest was expressed in determining the relative influence of flaw geometry (aspect ratio) on residual life. As such, a study aimed at determining the sensitivity of residual life to the initial flaw geometry was initiated. Table 5 presents the results from a study where two different aspect ratios (2:1 and 3:1) were considered. As is apparent from the summary a significant difference in residual life, upwards to 25% can be expected for reasonably possible variations in the initial defect geometry. The more conservative value of 2:1 was selected for use on all baseline crack growth analyses.

Table 5 - Dependency of Residual Life on Initial Flaw Aspect Ratio

LOCATION	2:1 ASPECT RATIO* Keff	2:1 ASPECT RATIO* MSSN. HRS.	3:1 ASPECT RATIO* Keff	3:1 ASPECT RATIO* MSSN. HRS.	% DIFFERENCE
FAN DISK BORE	1.1400	1.0	1.1241	1.16	15.9
FAN DISK AFT WEB	1.1864	1.0	1.1600	1.14	14.2
COMPRESSOR (OLD)					
STAGE 3 DOVETAIL	0.4043	1.0	0.3867	1.12	12.2
HPT					
STAGE 1 BORE	.3014	1.0	.2851	1.18	[illegible]
LPT					
STAGE 4 BORE	.3879	1.0	.3636	1.22	21.7

*LIVES ARE COMPOSITES FROM .015 x .030 (2:1) AND .010" x .030" (3:1) INITIAL FLAWS.

TASK VI - MODIFICATION/REWORK OPTION STUDY

In an effort to extend the safe utilization of the durability and fracture critical components to 8000 hours, various modification/rework studies were undertaken. All parts were screened to determine the life limiting characteristics and possible resources for extending or enhancing the durability and/or safety inspection limits. Different options were considered and a preferred option selected based upon the following criteria.

- o Life cycle cost benefits
- o Difficulty of introduction at depot facility
- o Potential enhancement and risk factor

Methods for extending the useful service life of 18 critical components upwards to 8000 hours were defined and grouped into three categories: rework, life extension, and RFC. Reworks included bushing of pinholes, overstressing through cold working or overspeeding, and reduced stress concentration by increased radius of geometric discontinuities.

Component life extension through controlled operation beyond the "lower bound" LCF life was recommended in situations where historical data for the particular material suggested that a significant improvement in LCF life could be realized through use of specific part data. A further prerequisite for adopting the life extension option was a relatively long residual life capability for the prescribed limiting location(s). The selection of this approach to life management while working to develop specific part LCF data required institution of enhanced depot inspections.

The RFC option was recommended for components where the LCF life developed using specific part data was less than the desired 8000 hour goal. A corresponding high crack growth capability was necessary prior to considering RFC a viable option. As with life extension, incorporation of enhanced inspections for the critical features was a prerequisite for implementing an RFC approach to life management.

TASK VII - FORCE STRUCTURAL MAINTENANCE PLAN

Introduction

This task served to integrate the results of Tasks IV-VI into an overall maintenance plan for the TF34-100. Contained within the plan is the specific scheduled maintenance requirements inclusive of component inspection and replacement intervals. Where appropriate a progressive program was defined to reflect anticipated revisions to the plan. These revisions are expected as results from planned/on-going activities (stress refinements, specific part material data, component residual life testing, etc) become available. The unscheduled and opportunistic maintenance items were indirectly treated in definition of windows on the scheduled intervals. The final version of the Force Structural Maintenance Plan inclusive of all scheduled and unscheduled maintenance activities was established through the efforts of the Maintenance Planning Group (MPG).

A summary of the specific items addressed in the development of the Force Structural Maintenance Plan is presented below.

- o Life Assignment Methodology/Criteria
- o Proposed Life Limits and Inspection Intervals
- o Inspection Requirements
- o Progressive Life Growth Plan
- o Structural Maintenance Plan Development
- o Life Tracking System

Life Assignment Methodology/Criteria

Life limits and inspection intervals for the fracture and durability critical components were established with the following guidelines.

Component replacement was based upon the conventional LCF life. Both generic strain controlled and generic load controlled LCF data were used in establishing the life, the lower of which established the "lower bound" limit. Due to the nature of the data base this "lower bound" value was considered a worst case minimum LCF life. In situations where the "lower bound" limit reflected less than desired life a specific part material program was identified. As an interim measure the generic load controlled results were used in conjunction with the generic strain control limits to arrive at a recommended minimum LCF life for field application. Therefore, continued operation beyond the "lower bound" LCF limit was considered an interim measure while specific part data was generated. As discussed in Task IV embedded flaw residual lives using a probabilistic based initial defect size results in lives (1/10000) beyond the 8000 hour goal. Therefore, all replacement intervals reflected LCF limits.

Fracture critical component safety inspection intervals were arrived at using the results from the residual life analyses previously discussed in Task IV. The assigned interval was initially established treating a 0.015" deep flaw. For locations where the assumption failed to provide a reasonable inspection interval consistent with the majority of components in the major assembly, an assumed initial flaw downwards to 0.005" in depth was considered. An enhanced (eddy current) inspection requirement was identified for any area where a small flaw ($<$ .015") assumption was employed in defining the interval. Additionally, for critical areas where visual detection of a 0.015" flaw was hindered due to the geometric complexity of the feature or compromised by high residual stress levels, an eddy current inspection was established.

In defining the recommended safety and durability limits, consideration was given to all aspects of the analytical and empirical base upon which they were derived. Particularly, results from the sensitivity studies, factory and field experience, empirical verification efforts, and material data sources served to provide varying degrees of flexibility in assignment of life limits.

Proposed Life Limits and Inspection Intervals

The maintenance philosophy developed for the TF34-100 contains an integration of the two criteria discussed above. The durability limit represents the point in time when the part is retired from service. As previously stated, operation beyond the "lower bound" limit was recommended for selected

components as a means of life extension while specific part material data was generated. A projection of the expected improvement in the durability limit when using specific part LCF data was performed to aid in piece-part provisioning.

The safety inspections as derived from the fracture analyses supplemented the durability limits. These inspections are instituted when the component enters service and serve to insure structural integrity of the fracture critical components. Figure 15 illustrates the methods by which these two elements interact. As shown, if operation beyond the durability limit was desired a reduction in the inspection interval was required. Where life extension beyond the "lower bound" durability limit was proposed a corresponding reduction in the inspection interval and/or the identification of an eddy current inspection requirement was defined.

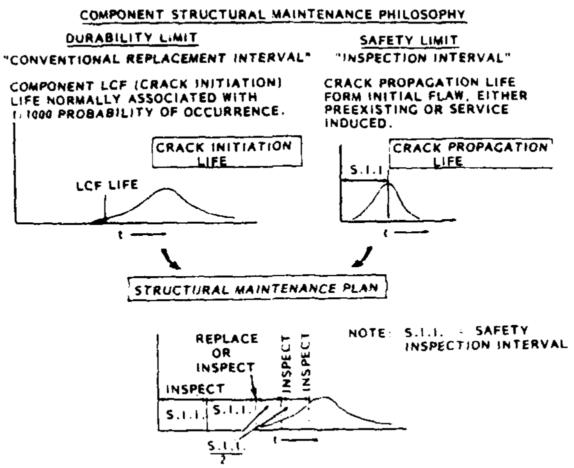

Figure 15 - Elements of Structural Maintenance Development

Inspection Requirements

The basis for selecting particular inspection methods was briefly described in the aforementioned paragraphs. Table 6 summarizes the nature and extent of the inspections defined for the critical components. As noted, implementation of eddy current inspections at production was not recommended. With TF34-100 production nearly complete minimal benefit could be derived in equipping production for enhanced inspections. Where required, spare parts forwarded to depot were inspected prior to installation.

Table 6 - Summary of Inspections for Critical Components

MAJOR ASSEMBLY	CRITICAL PARTS	FPI* CONTRACTOR	FPI [illegible]	EDDY CURRENT [illegible]	EDDY CURRENT [illegible]	[illegible]	[illegible]
FAN	[illegible]	[illegible]	[illegible]		[illegible]		
COMPRESSOR	[illegible]	[illegible]	[illegible]		[illegible]		
HPT		[illegible]	[illegible]		[illegible]		
LPT	[illegible]	[illegible]	[illegible]		[illegible]		
STATIC	[illegible]	[illegible]	[illegible]		[illegible]		
	[illegible]	[illegible]	[illegible]		[illegible]		

*All parts receive FPI in production.

Progressive Life Growth Plan

As alluded to, the recommended component inspection and replacement intervals as defined through the assessment team's activities represented the proposed scheduled maintenance plan for the TF34-100. However, for those areas where the recommmended inspection/replacement intervals were less than the desired goals, a follow-on life enhancement program was identified. Normally these programs contained specific part material characterization and were aimed at extending the durability or replacement limits.

Structural Maintenance Plan Development

Development of the Structural Maintenance Plan served to culminate the efforts completed in Tasks IV-VI. A multi-faceted approach was defined for establishing the recommended life limits, depot return intervals, and field/depot inspections. The essential ingredients included:

- o Durability and safety limits as defined through analysis and testing.
- o Definition of variability in life limits due to material data base and assumptions employed in establishing the limits.
- o Planned/on-going activities that were likely to influence the particular limits.
- o Demonstrated sensitivity of primary components to cracks in critical features.
- o Supportability picture in terms of manpower, facilities, and piece part requirements.

A set of life goals consistent with the Component Improvement Program's charter were adopted when determining the extent of follow-on efforts required. Locations where the existing life was derived from generic LCF data a specific part program was defined and incorporated into the Life Management program. The individual results were integrated, taking into account existing field and depot maintenance items, into a major assembly depot/field maintenance plan. Figure 16 illustrates the resulting Maintenance Plan, by assembly, proposed for the TF34-100.

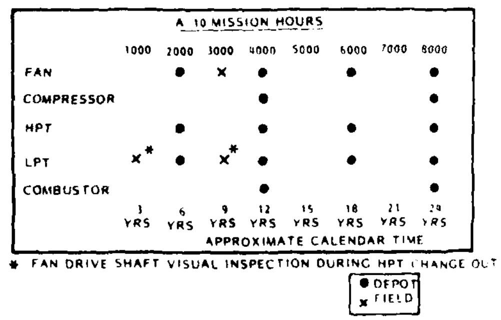

Figure 16 - Scheduled Depot and Field Inspection Maintenance Plan

In the final analysis a Maintenance Plan optimized on relative impact, cost, and safety was defined for the TF34-100/A-10 weapon system. The plan as

developed, served as the starting point upon which a more comprehensive program continued to evolve.

Life Tracking System

With the current life tracking system (Engine Time Temperature Recorder at engine level, MIMICS at base level and G337 at depot level) capable of recognizing limits based upon either cycles or engine operating hours an either/or criteria was established for tracking the inspection intervals. Rather than assigning a discrete inspection limit to each component a grouping according to major assembly was considered. An inspection limit (cycle/hours) associated with the most limiting (safety limit) component within an assembly was assigned to the assembly. Exceedance of either limit was cause for module removal and return to depot for inspection.

Replacement/refurbishment intervals will continue to be tracked on an individual component basis. Durability limits as prescribed in Task VII, coupled with corresponding K factors, served to control part replacement. Optimum inspection/replacement windows are dependent upon other logistic factors and were treated as part of the Maintenance Planning Group (MPG) activities.

TASK VIII - IMPLEMENTATION ACTIONS

Effort associated with this task focused on defining the actions and organizations responsible for implementing the DADTA's recommended Force Structural Maintenance Plan. In addition, tasks necessary to support the Life Growth Plan developed in Task VII were defined for inclusion into the on-going Component Improvement Program. A strong commitment by the contractor, depot and DADTA team personnel was essential to the successful and timely execution of the implementation action plan.

SUMMARY

The TF34-100 Durability and Damage Tolerance Assessment activities culminated in a comprehensive Force Structural Maintenance Plan that identified both current and future maintenance actions necessary for insuring maximum flight safety of the TF34-100/A-10 weapon system. The plan contained component inspection intervals, replacement times, inspection systems, preferred modifications/reworks, and definition of a life growth plan for extending the useful life upwards to 8000 hours. A Life Cycle Cost (LCC) analysis of the assessment's recommended preferred options reflected sizable monetary savings through adoption of the proposed inspection/replacement schedules for the various components. Although philosophically different from the Life Management plan in place at the outset of the assessment, with the inclusion of a fracture mechanics based inspection criteria, the essential elements of the plan are consistent with the overall goal of obtaining the maximum affordable operational safety possible.

The nature and extent of analysis efforts which were undertaken was unprecedented for the TF34-100. The analyses served to identify locations which were particularly sensitive to LCF generated, intrinsic, or induced (handling, machining, etc.) flaws. As such, upgraded and enhanced inspections were defined and recommended for these areas. An extensive and aggressive program was defined and executed for the implementation of improved FPI and eddy current inspections at depot. The full support of the Navy (direct depot responsibility), Air Force, and the Contractor was required for implementation to follow a timely course.

The TF34-100 Structural Assessment provided further insight into the structural requirements necessary for ensuring maximum flight safety. It marks the second engine in the Air Force inventory to receive an assessment of this type. The assessment continues to demonstrate the applicability of a Damage Tolerance criteria to the life management of engine components. The uncertainties related to component life, operational environment, and manufacturing variations highlights the importance of establishing Safety Inspections on the critical parts. The underlying philosophy is one of Prevention rather than Reaction. Growth of this concept is strengthened with the successful completion and demonstration of the benefits, vis-a-vis F100 and TF34 experience. It is an approach which is rapidly taking hold in the design stage of new engines, as apparent from the incorporation of Damage Tolerance requirements in the Engine Structural Integrity Program (ENSIP) specification. The specification mandating this design methodology was put in force on the 45th anniversary (1984) of the first gas turbine engine powered flight. The influence of this specification is already apparent in the design of the Next Generation Trainer (NGT) and Advanced Tactical Fighter (ATF) engines. The NGT and ATF engines will constitute the first structural designs to fully comply with the damage tolerance design approach set forth in the Air Force's ENSIP Mil Standard 1783. As with the Structural Assessments they are not destined to be the last.

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(1) Walker, K., "The Effect of Stress Rates During Crack Propagation and Fatigue for 2024-T3 and 7075-T6 Aluminum," ASTM STP 462, 1970

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(3) Neuber, H., "Theory of Stress Concentration for Sheer-Strained Prismatical Bodies with Arbitrary Nonlinear Stress Strain Laws," Trans, ASME, Journal of Applied Mechanics, Vol 28, Dec 1961, pp 544

(4) Irwin, G., Journal of Applied Mechanics, Vol 29, 1962, pp 651

(5) Willenborg, J., Engle, R. M., and Wood, H. A., "A Crack Growth Retardation Model Using an Effective Stress Concept"

CRYOGENIC PROOF TEST

A positive inspection technique

Troy King and F100 Engine Projects
Engineering-Operations and Government
Pratt & Whitney

Herb Rippa
Jet Engine Overhaul Division
San Antonio Air Logistics Center

AVB322095 870311

CHART 1

GOOD MORNING EVERYONE! HERB RIPPA AND I ARE GLAD TO HAVE THIS OPPORTUNITY TO TELL YOU ABOUT THIS IMPORTANT JOINT VENTURE BETWEEN THE UNITED STATES AIR FORCE AND PRATT & WHITNEY - CRYOGENIC PROOF TEST OF F100 ENGINE FAN DISKS.

CRYOGENIC PROOF TEST BECAME AN OBJECTIVE IN 1979 AND HERB AND I HAVE BEEN CLOSELY INVOLVED WITH THIS EFFORT SINCE THEN. MY INVOLVEMENT HAS BEEN WITH THE TECHNICAL NEED, DESIGN AND SUBSTANTIATION WHILE HERB HAS HAD THE DIFFICULT JOB OF GETTING THE FACILITY BUILT. AND AS YOU'LL SEE, HIS EFFORTS HAVE BEEN VERY SUCCESSFUL AND THE FACILITY BECAME OPERATIONAL IN MID-1984.

OUTLINE

F100 fan disk cryogenic proof test

- Background
- Theoretical considerations
- Benefits of cryogenic proof test
- Substantiation program
- Cryogenic spin test facility - SA-ALC
- Spin test results to date
- Summary

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CHART 2

THE OUTLINE WE'LL COVER WITH YOU IS SHOWN HERE AND IT COVERS THE COURSE OF EVENTS SINCE THE TECHNICAL NEED WAS ESTABLISHED, INCLUDING THE STATUS OF SPIN TEST EXPERIENCE FROM 1984.

BACKGROUND

F100 fan disk cryogenic proof test

F100 engine durability and damage tolerance assessment, 1978-1979

- Identified revised replacement limits, safety inspection requirements and preferred mods for F100 engine

Fan disk design identified as sensitive

- Detection of small flaws required in critical locations
- Inspection intervals less than mature maintenance goals

Two options studied

- Cryogenic overspeed proof test
- Alternate material

1979 decision

- Eddy current short-term (1980 and 1981)
- Cryogenic proof test long-term (1982 and subs)

1985 CIP task

- Redesign

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CHART 3

- AS SOME OF YOU ARE AWARE, A DURABILITY AND DAMAGE TOLERANCE ASSESSMENT WAS CONDUCTED ON THE F100 ENGINE BY AN ON-SITE, JOINT AIR FORCE, MCAIR, GD AND P&W DURING 1978 AND 1979. THIS ASSESSMENT OF THE F100 ENGINE BY THE AIR FORCE REPRESENTED THE FIRST FOR ENGINES ALTHOUGH THIS INITIATIVE HAS BEEN AN IMPORTANT ONE BY THE AIR FORCE; AND, SINCE 1970, HAS COVERED ESSENTIALLY EVERY AIRFRAME SYSTEM IN THE INVENTORY.

  THE OBJECTIVE OF THESE ASSESSMENTS HAS BEEN TO IDENTIFY REVISED REPLACEMENT LIMITS, SAFETY INSPECTION REQUIREMENTS AND PREFERRED MODIFICATIONS FOR CRITICAL, LIFE LIMITED COMPONENTS. THE MAIN INGREDIENT HAS BEEN TO INTRODUCE AND IMPLEMENT DAMAGE TOLERANCE REQUIREMENTS FOR LIFE MANAGEMENT.

- AS A RESULT OF THE F100 ENGINE STRUCTURAL ASSESSMENT, THE FAN DISKS WERE IDENTIFIED AS SENSITIVE WHEN EVALUATED BY THE DAMAGE TOLERANCE CRITERIA. THE ANALYSIS INDICATED THAT RESIDUAL LIFE IN THE PRESENCE OF ASSUMED FLAWS REQUIRED THAT INSPECTION SYSTEMS BE CAPABLE OF RELIABLY DETECTING SMALL FLAWS. ALSO, THE ANALYSIS INDICATED THAT THE INSPECTION INTERVAL WOULD BE LESS THAN THE MATURE MAINTENANCE GOAL FOR THE F100 ENGINE FAN MODULE.

- TWO OPTIONS WERE REVIEWED TO ACHIEVE OPTIMUM INSPECTION INTERVAL:

  1) CRYOGENIC OVERSPEED PROOF TEST, AND, 2) ALTERNATE MATERIAL AND REDESIGN.

- IN 1979, THE JOINT DECISION WAS TO PURSUE IMPLEMENTATION OF CRYOGENIC PROOF TEST AS THE MATURE LIFE MANAGEMENT METHOD WITH EDDY CURRENT INSPECTION TO FILL THE SHORT TERM IMMEDIATE NEED. AT THIS POINT (08/79), CRYOGENIC PROOF TEST WAS THE GAME PLAN BASED ON LIMITED FEASIBILITY ANALYSIS AND TEST, BUT SUBSTANTIATION WAS STILL AHEAD OF US.

F100 ENGINE FAN ROTOR SECTION

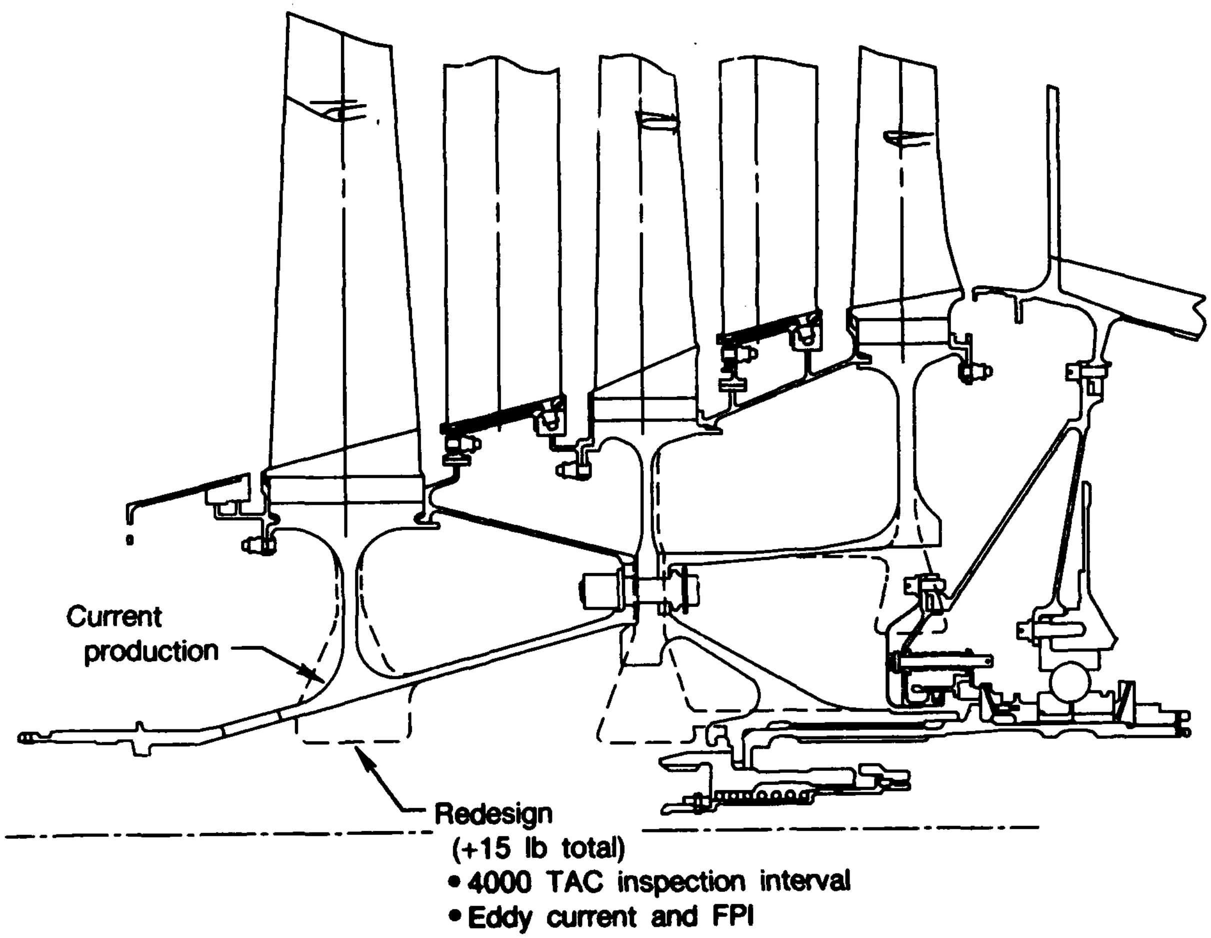

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CHART 4

THIS IS A CROSS SECTION VIEW OF THE FAN ROTOR AND, IN PARTICULAR, THE 1ST, 2ND, AND 3RD STAGE FAN DISKS WHICH ARE THE CENTER OF INTEREST IN THIS BRIEFING.

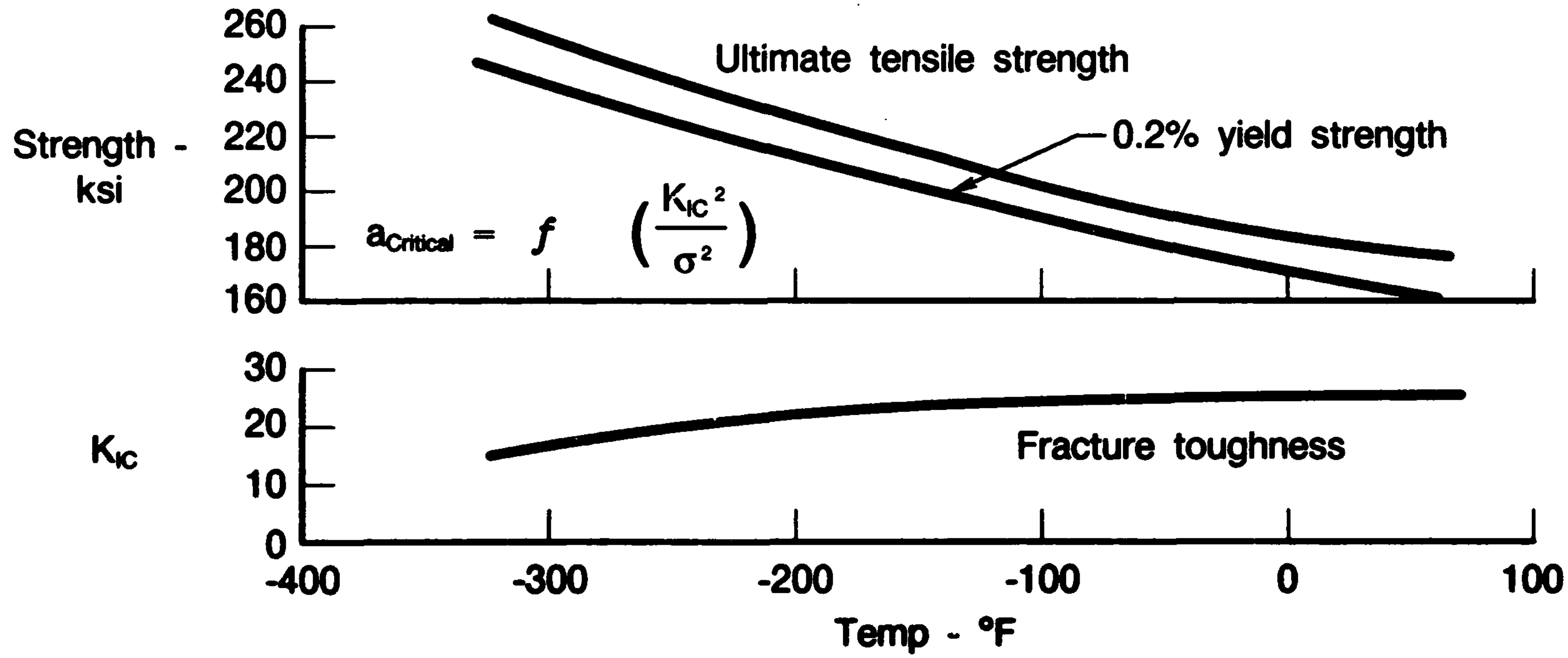
THE DISK MATERIAL IS HIGH STRENGTH TITANIUM WITH STRENGTH AND LOW CYCLE FATIGUE CRACK INITIATION CAPABILITIES MUCH SUPERIOR THAN CONVENTIONAL TITANIUMS LIKE 6-4 AND 8-1-1. HOWEVER, AS IS OFTEN THE CASE WITH A DESIGN UTILIZING HIGH STRENGTH ALLOYS, THE CRACK GROWTH RATE IS INCREASED FOR A GIVEN STRESS INTENSITY AND DUE TO THE HIGHER OPERATING STRESS AFFORDED BY THE HIGHER YIELD STRESS ALLOWABLE.

THE RECENT REDESIGN USES DAMAGE TOLERANCE DESIGN CRITERIA WITH NO MATERIAL CHANGE AND ACHIEVES AN INSPECTION INTERVAL EQUAL TO 1/2 THE DESIGN LIFE WITH MARGIN; OR PUT ANOTHER WAY, ACHIEVES A RESIDUAL LIFE EQUAL TO THE FULL DESIGN LIFE.

- IT IS INTERESTING TO NOTE THE FAN DISKS HAVE BEEN REDESIGNED UNDER A CIP TASK AND ARE SCHEDULED TO BECOME A PART OF -220 ENGINE PRODUCTION IN THE FIRST HALF OF 1988. WE LEARNED HERE WHAT WAS PREVIOUSLY LEARNED ON SOME OTHER DESIGNS UTILIZING HIGH STRENGTH MATERIALS - THE DEFICIENCY OFTEN RESTS WITH THE DESIGN CRITERIA AND NOT THE MATERIAL.

EFFECT OF CRYOGENIC TEMPS

Lower temperature increases strength and increases brittleness for Ti 6-2-4-6



Note: $a_{Critical}$ = Limiting crack depth, K_{IC} = Fracture toughness, σ = Operating stress

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CHART 5

THE FAN MATERIAL IS AN IDEAL CANDIDATE FOR CRYOGENIC OVERSPEED PROOF TEST AS A METHOD FOR SCREENING FOR INITIAL FLAWS. THAT IS, THE STRENGTH INCREASES AND THE TOUGHNESS DECREASES AS THE TEMPERATURE IS LOWERED.

FLAW DETECTION CAPABILITY

Low temperature and high stress proof for small flaws

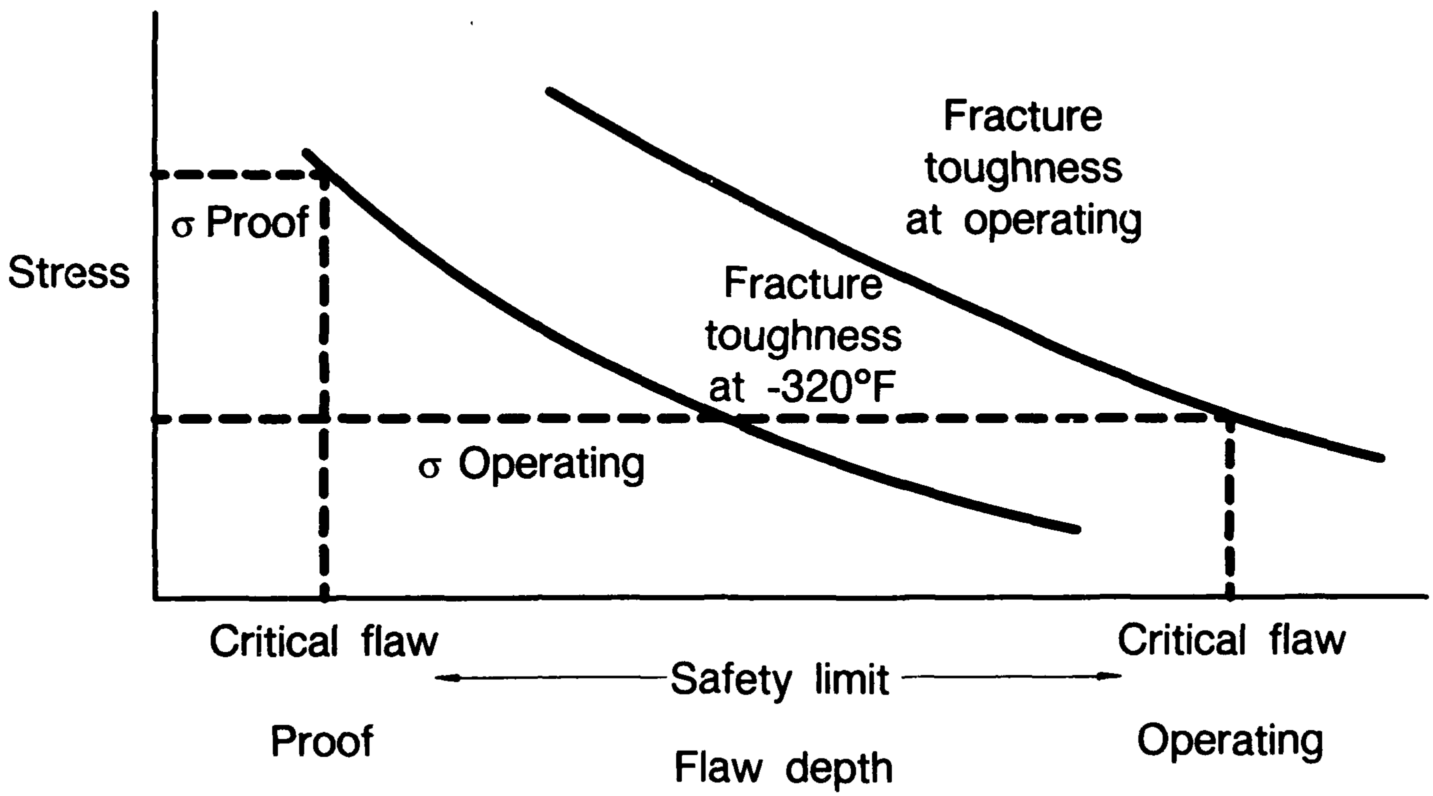

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CHART 6

THE ABILITY TO PROOF TEST AT A STRESS LEVEL SIGNIFICANTLY HIGHER THAN THE OPERATING STRESS AND AT A TOUGHNESS LOWER THAN EXISTS AT OPERATING TEMPERATURE ALLOWS US TO SCREEN FOR SMALL FLAWS. THIS, IN EFFECT, PROVIDES A MAXIMUM SEPARATION BETWEEN MAXIMUM UNDETECTABLE INITIAL FLAW SIZE AND CRITICAL FLAW SIZE AND RESULTS IN A MAXIMUM RESIDUAL LIFE.

RESIDUAL STRESS

Beneficial residual stress from local yielding retards crack growth

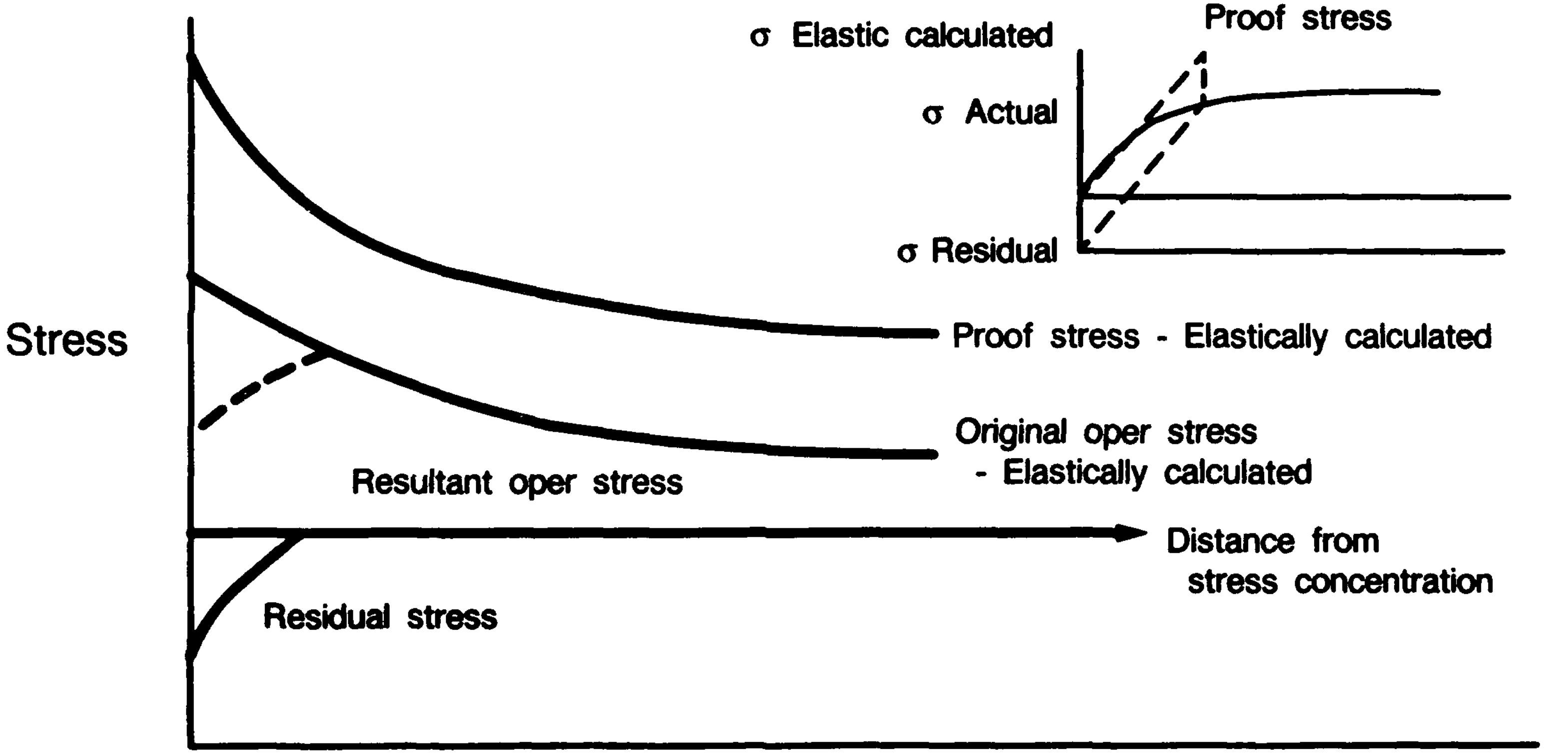

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CHART 7

A MORE SUBTLE BENEFIT ACHIEVED BY OVERLOAD PROOF TEST IS THE INCREASED BENEFICIAL RESIDUAL COMPRESSIVE STRESSES THAT ARE INTRODUCED AT CRITICAL STRESS CONCENTRATION LOCATIONS. CRACK GROWTH SUBSEQUENT TO PROOF TEST IS SLOWED SINCE THE MEAN STRESS IS LOWERED AS THE STRESS RANGE OCCURS ON EACH CYCLE.

PROOF TEST CONCEPT

Cryogenic proof test accomplishes two things

- Screens for small initial flaws ($<$0.005 in. deep)
- Local yielding (Stress concentration regions) introduces beneficial residual stresses

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CHART 8

IN SUMMARY, THE TWO BENEFITS AFFORDED BY FAN DISK CRYOGENIC PROOF TEST ARE 1) SCREEN FOR SMALL FLAWS ON THE ORDER OF .005 INCH DEEP, AND, 2) LOCAL YIELDING AT STRESS CONCENTRATION LOCATIONS WHICH EXTENDS RESIDUAL LIFE.

FAN CRYOGENIC TEST BENEFITS/SAVINGS*

- INSPECTION INTERVAL INCREASED FROM 1800 TO 3000 CYCLES
- AVOIDS 3141 FAN INSPECTIONS AT THE DEPOT
- AVOIDS TWO DISK CHANGEOUTS
- TOTAL PROJECTED SAVINGS IS $179,118,000 FOR THE LIFE OF THE ENGINE

*BASED ON 2585 FAN MODULES, 20 YEAR LIFE CYCLE

CHART 9

AS SHOWN HERE, THE LIFE MANAGEMENT BENEFITS FOR THE F100 ENGINE ARE SIGNIFICANT BY SEVERAL MEASURES OF MERIT.

SUBSTANTIATION PROGRAM

Extensive analyses and tests conducted (1979-1983)

Material property characterization

- -320°F, RT, 400°F

Residual stress (Strain) measurement

- Specimen and full scale disk

Residual life benefits

Proof test strain gage verification

- Assured proper tooling loading

Cryogenic pit thermal survey

Cryogenic proof test failure verification

- Specimen and full scale disk

AMT engine tests

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CHART 10

BETWEEN 1979 AND 1983, EXTENSIVE ANALYSIS AND TESTS WERE CONDUCTED TO FULLY SUBSTANTIATE CRYOGENIC PROOF TEST FOR THE F100 ENGINE FAN DISKS. THE SPECIFIC TEST CATEGORIES OR AREAS OF INVESTIGATION ARE LISTED.

THE MOST DRAMATIC TEST WAS FAILURE VERIFICATION WITH A FULL SCALE DISK DURING CRYOGENIC PROOF TEST. WE ACCOMPLISHED THIS IN 1979 PRIOR TO THE CULMINATION OF THE STRUCTURAL ASSESSMENT ON-SITE TEAM EFFORT TO CONVINCE OURSELVES THAT THE THEORY WAS VALID. THERE WERE A FEW DISBELIEVERS AT THIS TIME. THESE FAILURE VERIFICATION TESTS ALSO DEMONSTRATED THAT CRYOGENIC PROOF TEST WOULD DISCERN BETWEEN DETRIMENTAL CRACKS/DAMAGE AND OTHER ANOMALIES.

PURPOSELY DAMAGED 2ND DISK

Spin pit test results matched predictions

- Tack weld in bolthole ID $\approx$ 0.002 in. X 0.034 in.
- Cryo proofed to 15,000 rpm at -220°F
- Screened for $\approx$ 0.003 in. X 0.034 in. crack
- Cycled disk at 12,000 rpm (RT) until confirmed crack propagation (1400 lab cycles, equivalent to >7000 TAC cycles)
- Predicted disk rupture between 12,500 and 13,000 rpm 0.010 in. deep crack
- Cryo proofed at -230°F and failure occurred at 12,850 rpm
- Failure originated in induced crack; crack depth 0.008 in.

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2ND STAGE FAN DISK

Disk burst during proof test at predicted conditions

Successful cryo-proof test of 2nd stage fan disk with built-in flaw LCF crack (0.008 in.) in bolthole

AVB196826 870311

FIELD DAMAGED 1ST FAN DISK

Damage did not act like cracks

- Cryo proofed to 13,000 rpm
- Room temp cycled to 12,000 rpm
- Cryo proofed to 15,000 rpm
- Room temp cycled to 12,000 rpm

→ Equivalent to 14,000 TAC's field usage

AVA196822 870311

FA181 THIRD DISK

Disk with live rim indication passed proof test

- Eddy current indication in live rim
- Cryo proof to 15,000 rpm at -315°F
- Screened for ≈ 0.004 in. corner crack
- RT cycled to 10,300 rpm for 3,000 cycles

→ Equivalent to 3,000 TAC's field usage

- Disk cutup revealed no indications

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CHARTS 11 - 14

THESE CHARTS SUMMARIZE THE FULL SCALE FAILURE TESTS THAT DEMONSTRATED THE VIABILITY OF CRYOGENIC PROOF TEST.

CRYOGENIC SPIN TEST FACILITY

CHART 15 - INTRODUCTION

THE FACILITY WAS COMPLETED IN 1983. BUILDING AND EQUIPMENT COSTS TOTALLED $16 MILLION. THE PROJECTED SAVINGS FOR 20 YEARS ARE $179 MILLION.

WE CRYOGENICALLY SPIN TEST 1ST, 2ND, AND 3RD STAGE TITANIUM FAN DISKS THAT ARE INSTALLED IN THE INLET FAN OF THE F100-100 AND -200 ENGINE.

CHART 16 - BUILDING

THE CRYOGENIC SPIN TEST FACILITY IS LOCATED ON THE EAST SIDE OF BUILDING 360.
JET ENGINE OVERHAUL FACILITY.

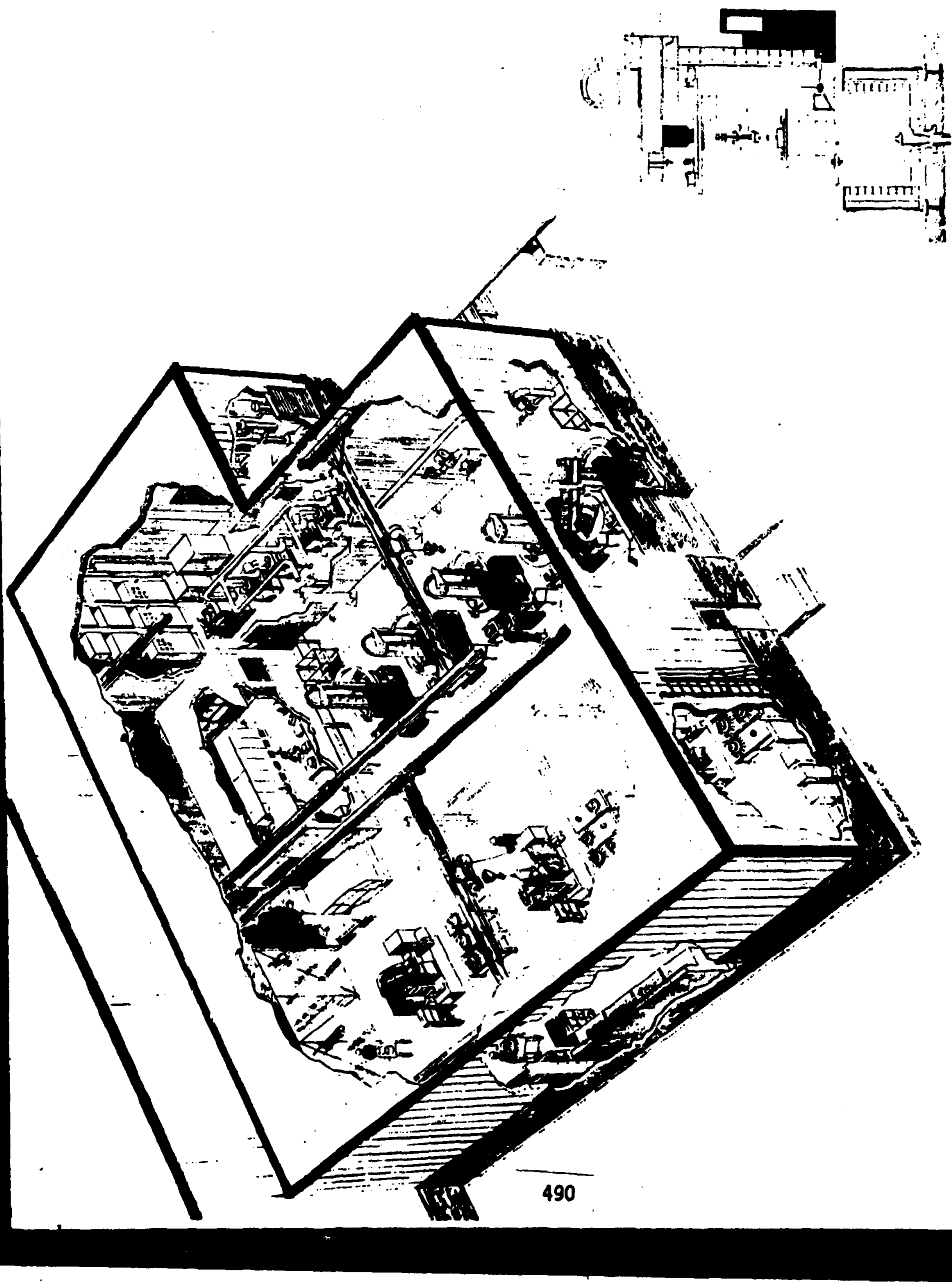


CHART 17 - FACILITY LAYOUT

THE FACILITY CONSISTS OF A BUILD UP AND BALANCING AREA AND THE TESTING OR PIT AREA. THERE ARE FIVE SPIN PITS DIVIDED INTO TWO SYSTEMS: PITS 1 & 2 ARE IN SYSTEM 1; PITS 3, 4 & 5 ARE IN SYSTEM 2.

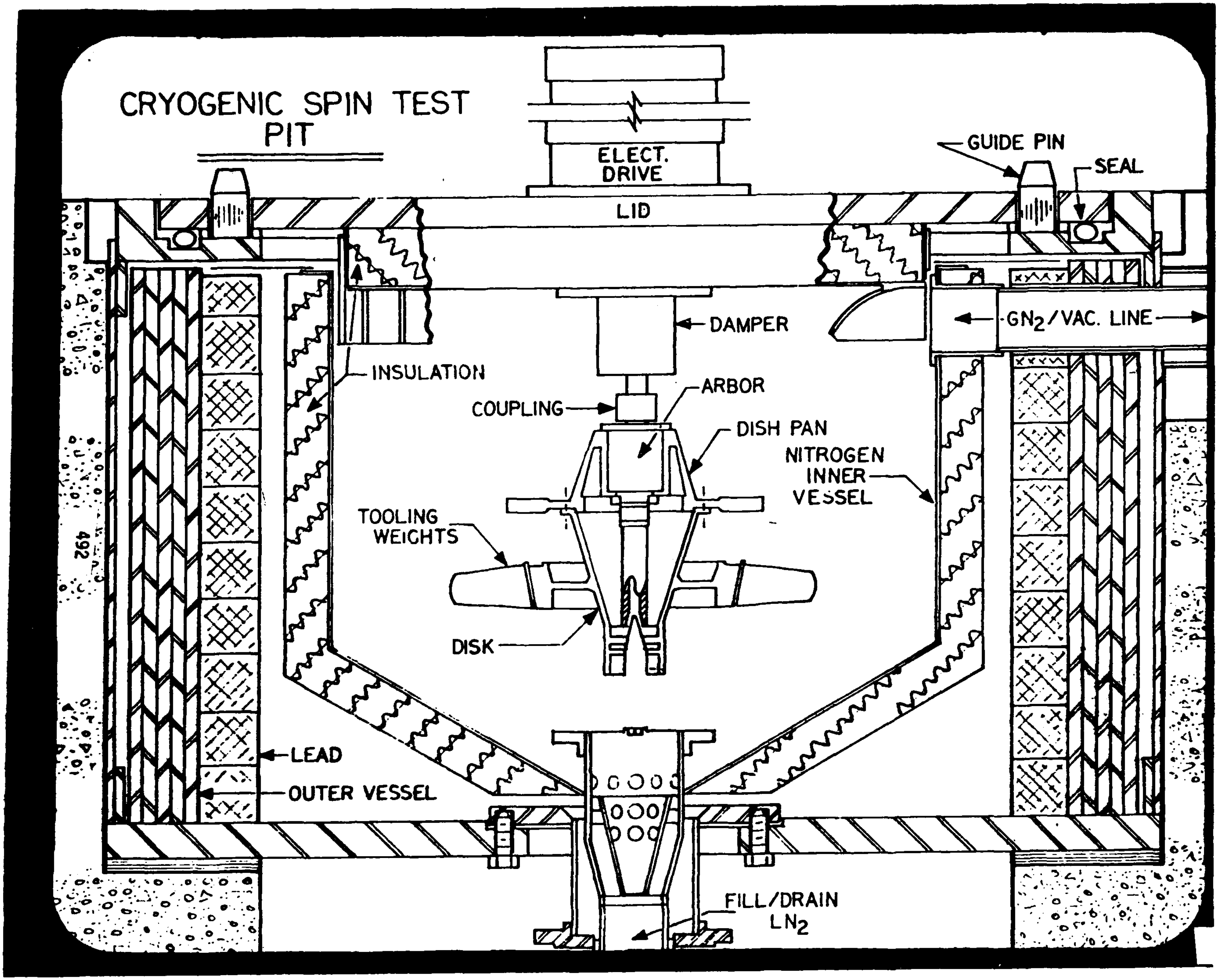

CRYOGENIC SPIN TEST PIT
ELECT. DRIVE
LID
GUIDE PIN
SEAL
DAMPER
GN$_2$/VAC. LINE
INSULATION
ARBOR
COUPLING
DISH PAN
NITROGEN INNER VESSEL
TOOLING WEIGHTS
DISK
LEAD
OUTER VESSEL
FILL/DRAIN LN$_2$

CHART 18 - CRYOGENIC SPIN TEST PIT

THE SPIN PIT CONSISTS OF A STAINLESS STEEL INNER VESSEL WRAPPED WITH INSULATION. THIS IS INSERTED INTO A STEEL OUTER VESSEL LINED WITH EXTRUDED LEAD BRICK DESIGNED TO ABSORB THE SCHRAPNEL FROM A BURST.

PRESSURE VESSEL

FABRICATION MATERIAL A-36 STAINLESS STEEL
WALL THICKNESS - 1/2" DIAMETER - 73" HEIGHT - 46-1/2"
BASE THICKNESS (BOTTOM PLATE) - 2"
5 LAMINATED CARBON STEEL RINGS WELDED TOGETHER SET ON TOP OF BOTTOM PLATE (FREE MOVING) MUST BE ROTATED BACK TO ORIGINAL POSITION AFTER BURSTS.

LEAD

FABRICATION PROCESS	EXTRUDED BRICK
DIMENSIONS	4" X 4" X 8"
BRICK WEIGHT	26 LBS
TOTAL NO. OF BRICK PER PIT	361

INNER VESSEL

FABRICATION MATERIAL	STAINLESS STEEL
STORAGE FOR LN2 TO SOAK PARTS	
WELDMENT ASSEMBLY	REUSABLE AFTER BURSTS
VESSEL INSULATION	FIBERGLASS, OWEN-CORNING 701 OR EQUAL
WALL THICKNESS	3"
LID INSULATION	FIBERGLASS, OWEN-CORNING 701 OR EQUAL
THICKNESS	VARIES 4" TO 8"

CHART 18 - CRYOGENIC SPIN TEST PIT (CONTINUED)

<u>DAMPER OPERATION</u>

IN ORDER TO KEEP THESE VIBRATIONS FROM DESTROYING THE COMPONET, PRATT & WHITNEY HAS DESIGNED A DAMPER TO EXERT A LATERAL LOAD ON THE DRIVE SPINDLE. THIS EXTERNAL LOAD ON THE SPINDLE CHANGES THE NATURAL FREQUENCY OF VIBRATION OF THE ROTATING MASS AND PREVENTS IT FROM COINCIDING WITH THE EXCITING FREQUENCY OF THE SPINDLE DRIVER AND PRODUCING RESONANCE. THIS ALLOWS THE SPINNING TEST PART AND DRIVE TO PASS THROUGH THE CRITICAL FREQUENCIES WITHOUT DESTROYING ITSELF, OR INVALIDATING THE TEST.

<u>DAMPER</u> - 2 DESIGNS

P & W - PRATT & WHITNEY
TDI - TEST DEVICES INC

COUPLED BETWEEN TEST PART AND HIGH SPEED MOTOR TO DAMPEN VIBRATION OF TEST PART. SENSITIVE DISPLACEMENT MEASUREMENT.

<u>HIGH SPEED MOTOR PERFORMANCE</u>

HIGH SPEED ELECTRIC MOTORS SHALL PROVIDE THE SPIN TORQUE REQUIRED FOR TESTING; ONE MOTOR TO BE LOCATED OUTSIDE OF EACH SPIN PIT ON TOP OF THE LID. LOADS ARE CONNECTED TO THE MOTOR DRIVE SPINDLE WHICH PENETRATES THROUGH THE LID, THUS SUSPENDING THE CONNECTED LOAD WITHIN THE COVERED SPIN PIT.

<u>DRIVE MOTOR</u> - CUSTOM DESIGN

30/15/3 H.P.
16500/8250/1800 RPM
460/230/50 VOLT
275/137/30 HERTZ
3 PHASE

FAN CRYOGENIC SPIN TEST PROCEDURE

- BUILD-UP DISK
- BALANCE
- MOVE BALANCED ASSEMBLY
- INSTALL ASSEMBLY
- PULL VACUUM
- COOL TO -300°F WITH LN_2
- SPIN AT 15,000 RPM
- HEAT DISK
- REMOVE DISK

82 513

CHART 19 - FAN CRYOGENIC SPIN TEST PROCEDURE:

THE 1ST, 2ND, AND 3RD STAGE TITANIUM FAN DISK OF THE INLET FAN FOR ALL F-100-100 AND -200 ENGINE ARE CRYOGENICALLY SPIN TESTED FOR FLAWS.

EACH OF THE THREE STAGES ARE SPUN INDIVIDUALLY, AFTER BUILD UP AND BALANCING.

THE BUILD UP PROCEDURE CONSISTS OF INSTALLING DUMMY TOOLING WEIGHTS THAT ARE 1.5 TIMES THE WEIGHT OF THE BLADES ACTUALLY USED IN THE ENGINE, A DISHPAN (FOR THE 1ST AND 3RD STAGES) AND AN ARBOR. THE 2ND STAGE DOES NOT REQUIRE A DISHPAN ASSEMBLY.

AFTER BUILD UP, THEY ARE BALANCED IN ONE OF THE THREE BALANCING MACHINES AND THEN TAKEN TO THE PIT AREA.

NEXT, THE DISK IS ATTACHED TO THE HIGH SPEED MOTOR DRIVE SHAFT AND THE LOWERED INTO THE PIT.

LN2 IS PUMPED INTO THE PIT AND THE TEST ASSEMBLY IS SOAKED IN THE LN2 FOR ONE (1) HOUR LOWERING THE TEMPERATURE TO -320°F.

THE LN2 IS PUMPED OUT OF THE PIT AND A VACUUM IS PULLED.

WHILE IN THE VACUUM AND WITH THE TEMPERATURE OF THE TEST ASSEMBLY (DISK) AT -320°F IT IS ACCELERATED IN 6 MINUTES TO 15,000 RPM THIS IS 1.5 TIMES THE ENGINE SPEED (10,000 RPM) AND HELD AT THIS SPEED FOR ONE (1) MINUTE. IN THE EVENT THERE IS A HIDDEN FLAW (LESS THAN 5 THOUSANDTHS OF AN INCH), THE FLAW WILL RAPIDLY PROPAGATE LEADING TO CATASTROPHIC FAILURE OF THE TEST ASSEMBLY. IF THE TEST PART DOES NOT FAIL BY BURSTING WHEN IT REACHES MAXIMUM SPEED, IT IS DECELERATED IN 6 MINUTES TO ZERO RPM.

AFTER THE SPIN TEST, THE TEST PIT IS FILLED WITH GASEOUS NITROGEN AT 25 PSIG AND 250°F $\pm$ 20°F TO GRADUALLY RAISE THE TEMPERATURE OF THE COMPONENT TO AMBIENT TEMPERATURE FOR HANDLING PURPOSES. THIS TAKES APPROXIMATELY 30 MINUTES.

THE TOTAL PROCESS TIME IS 2½ HOURS.

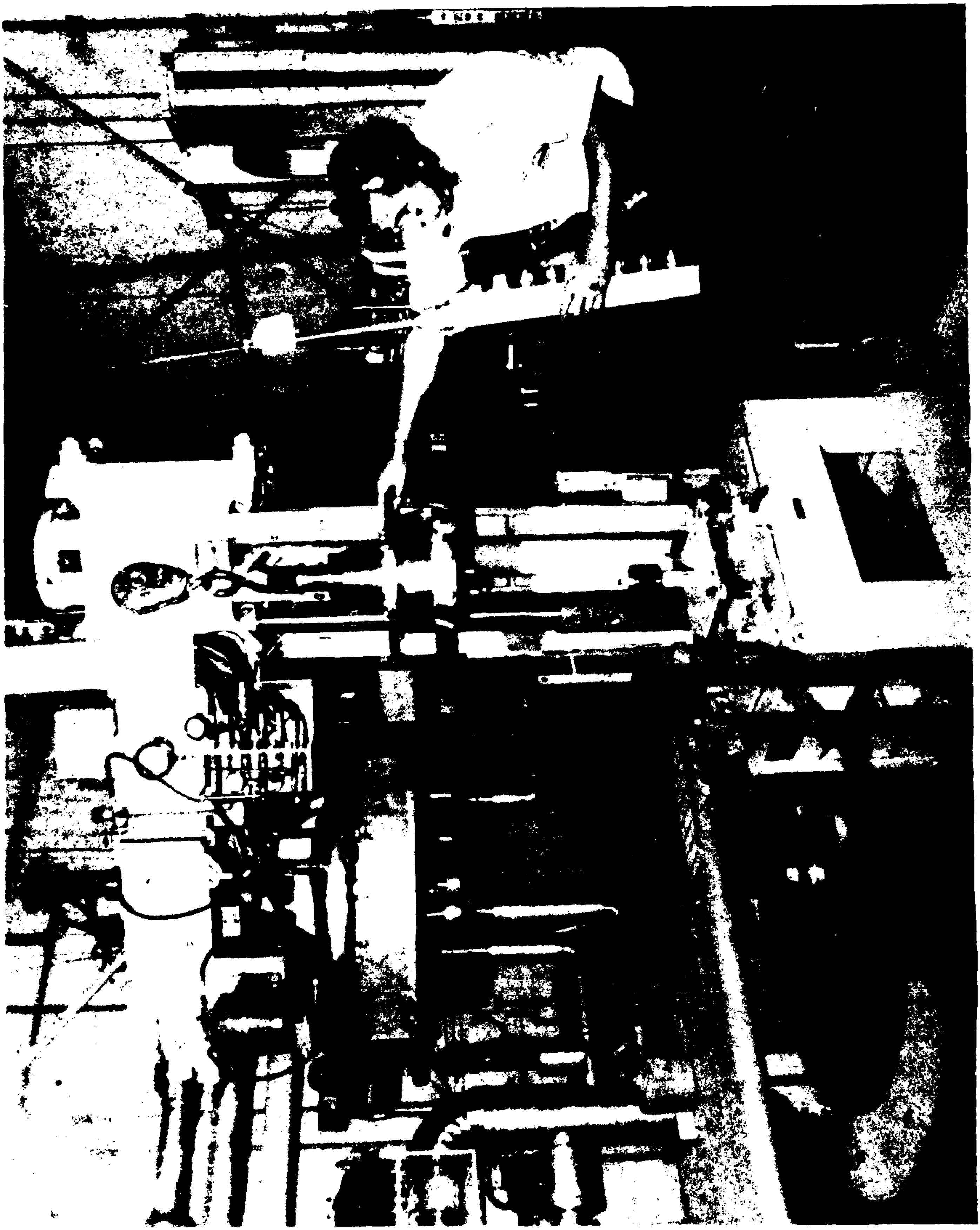


CHART 20 - DISK HANDLING FIXTURE

AN OPERATOR IS LOWERING A 3RD STAGE DISK ONTO THE DISK HANDLING FIXTURE. THE SPIN PIT LID WILL THEN BE ROTATED AND THE DRIVE SHAFT NUT WILL BE TIGHTENED READYING THE DISK FOR A HIGH SPEED SPIN TEST.

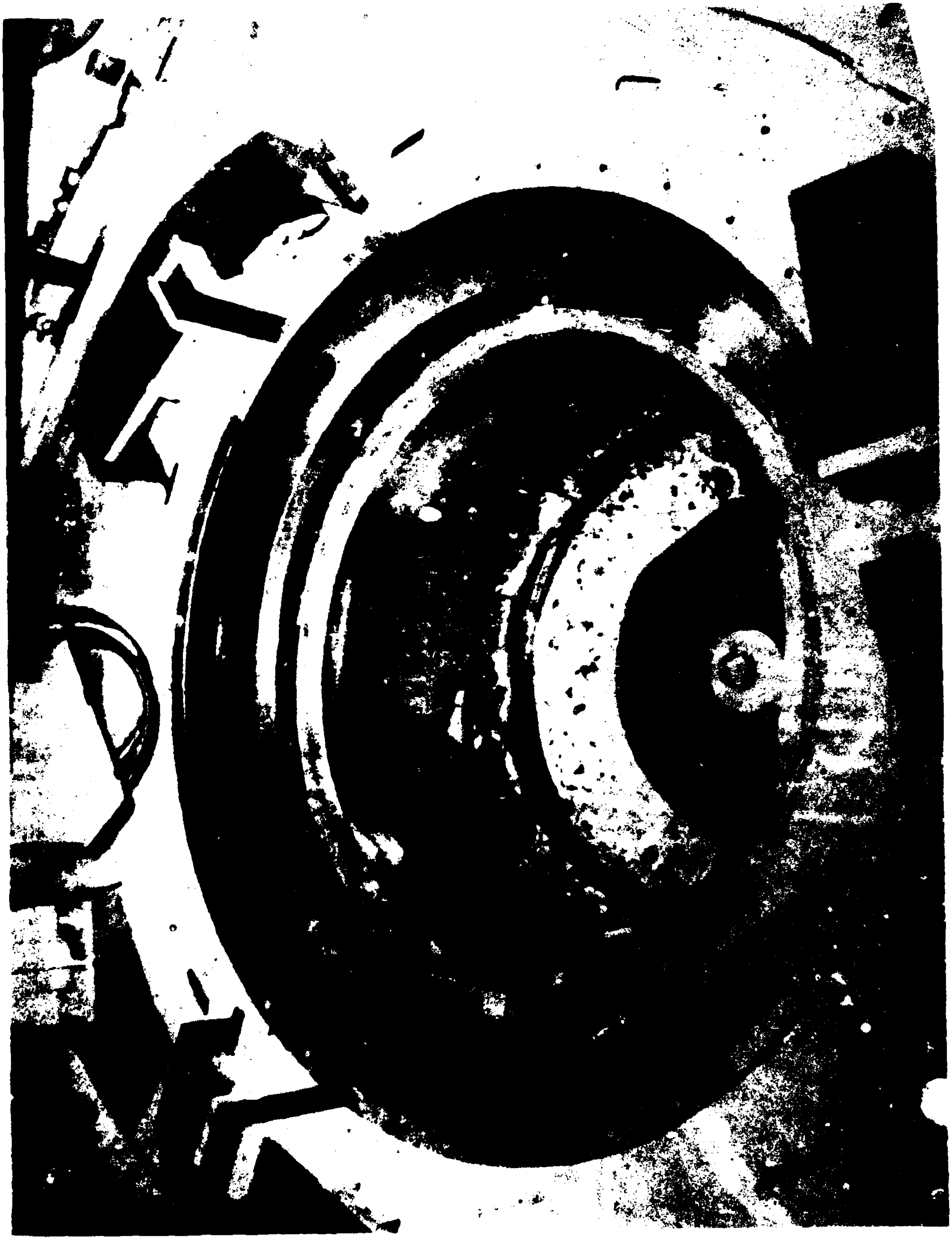


CHART 21 - BURST PIT

WHEN A BURST OCCURS, TOOLING WEIGHTS ARE PROPELLED AT A HIGH RATE OF SPEED, PIERCING THROUGH THE INNER VESSEL AND LODGING IN THE EXTRUDED LEAD BRICK. THE ENTIRE BURST IS CONTAINED UNDERGROUND. AS CAN BE SEEN THERE IS A DEFINITE "KILL ZONE" THAT IS DAMAGED IN A BURST. IT TAKES APPROXIMATELY 3-4 DAYS TO REBUILD A PIT AFTER A BURST AND COSTS AN AVERAGE OF $51,000.

CHART 22 - BURST DISK

THE FIRST BURST OCCURRED ON 1 MAY 1984. AS OF THE END OF SEPTEMBER 1987, THERE HAVE BEEN 103 BURSTS. ALL THESE FAILURES PASSED EDDY CURRENT INSPECTION. BESIDES SCREENING FOR FLAWS, WE GAIN SOME ADDITIONAL BENEFITS AS A RESULT OF THE SPIN TEST.

LOCAL YIELDING INTRODUCING BENEFICIAL RESIDUAL STRESSES (A GRAIN REFINEMENT PROCESS).

INSPECTION INTERVAL INCREASES FROM 3.5 YRS TO 6 YRS.

EXTENDS LIFE OF DISK FROM 7.5 YRS TO 20 YRS.

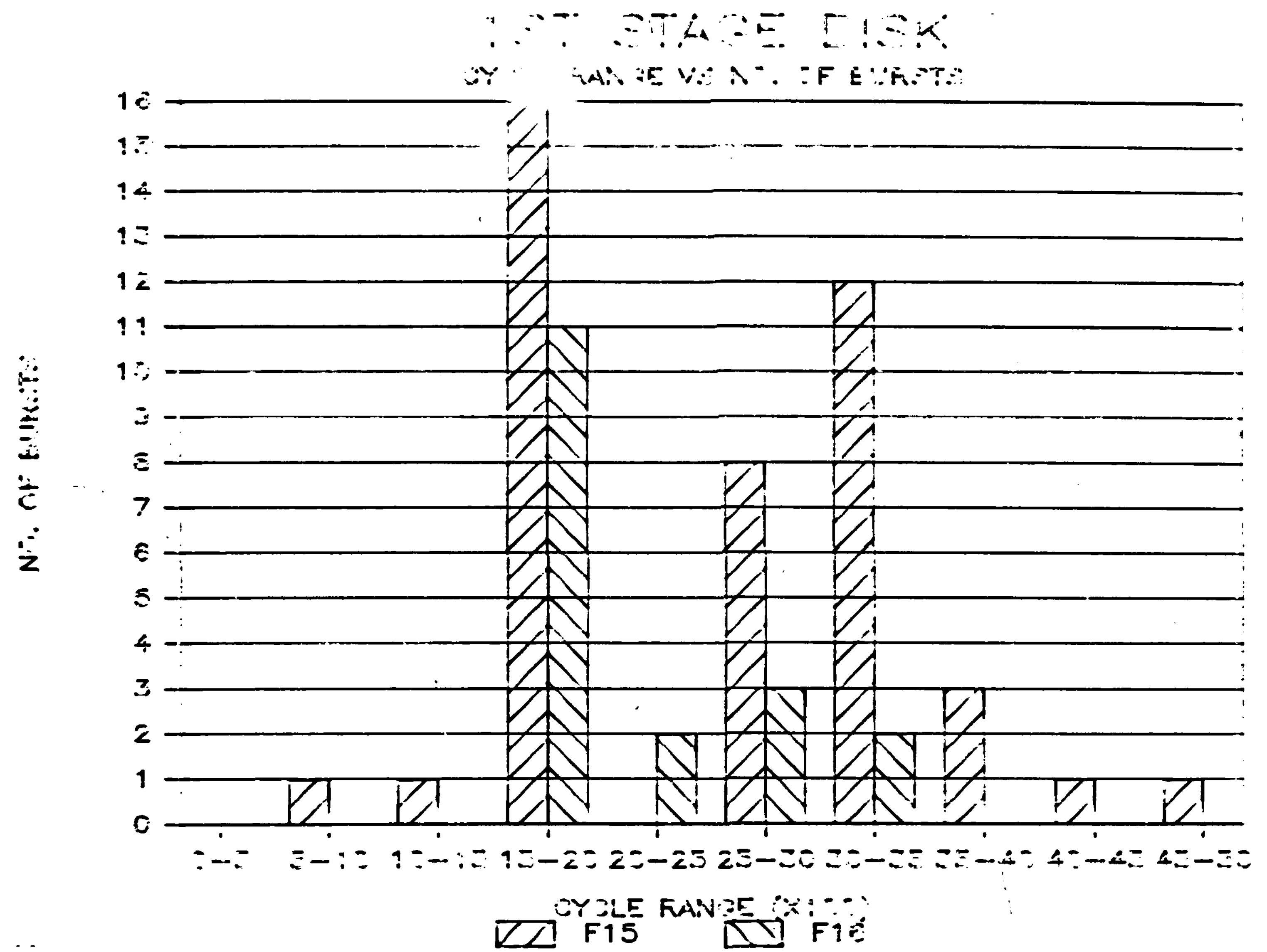

STAGE DISK
NO. OF BURSTS
16
15
14
13
12
11
10
9
8
7
6
5
4
3
2
1
0
0-5
5-10
10-15
15-20
20-25
25-30
30-35
35-40
40-45
45-50
CYCLE RANGE
F15
F16

CHART 23 - BAR   1ST STAGE DISK/CYCLE RANGE VS NO. OF BURSTS

			F15	(x 100)	F16	
CONCLUSION	0	BETWEEN	0	0-5	0	CYCLES
	1	BETWEEN	1	5-10	0	CYCLES
	1	BETWEEN	1	10-15	0	CYCLES
	27	BETWEEN	16	15-20	11	CYCLES
	2	BETWEEN	0	20-25	2	CYCLES
	11	BETWEEN	8	25-30	3	CYCLES
	14	BETWEEN	12	30-35	2	CYCLES
	3	BETWEEN	3	35-40	0	CYCLES
	1	BETWEEN	1	40-45	0	CYCLES
	1	BETWEEN	1	45-50	0	CYCLES
	61		43		18	

47.5% OF THE FIRST STAGE DISKS WHICH FAILED OCCURRED ON DISKS WITH 2000 CYCLES OR LESS.

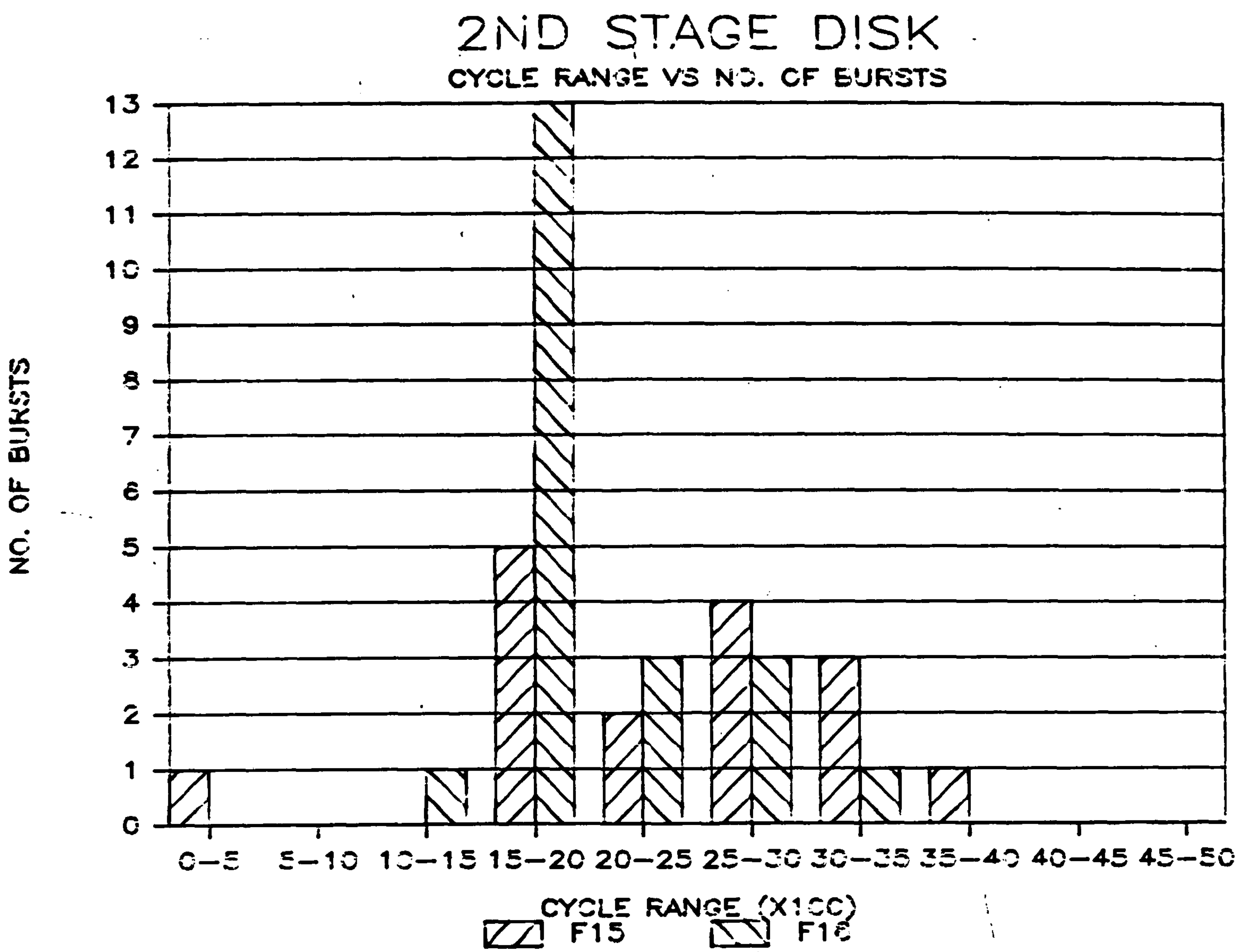
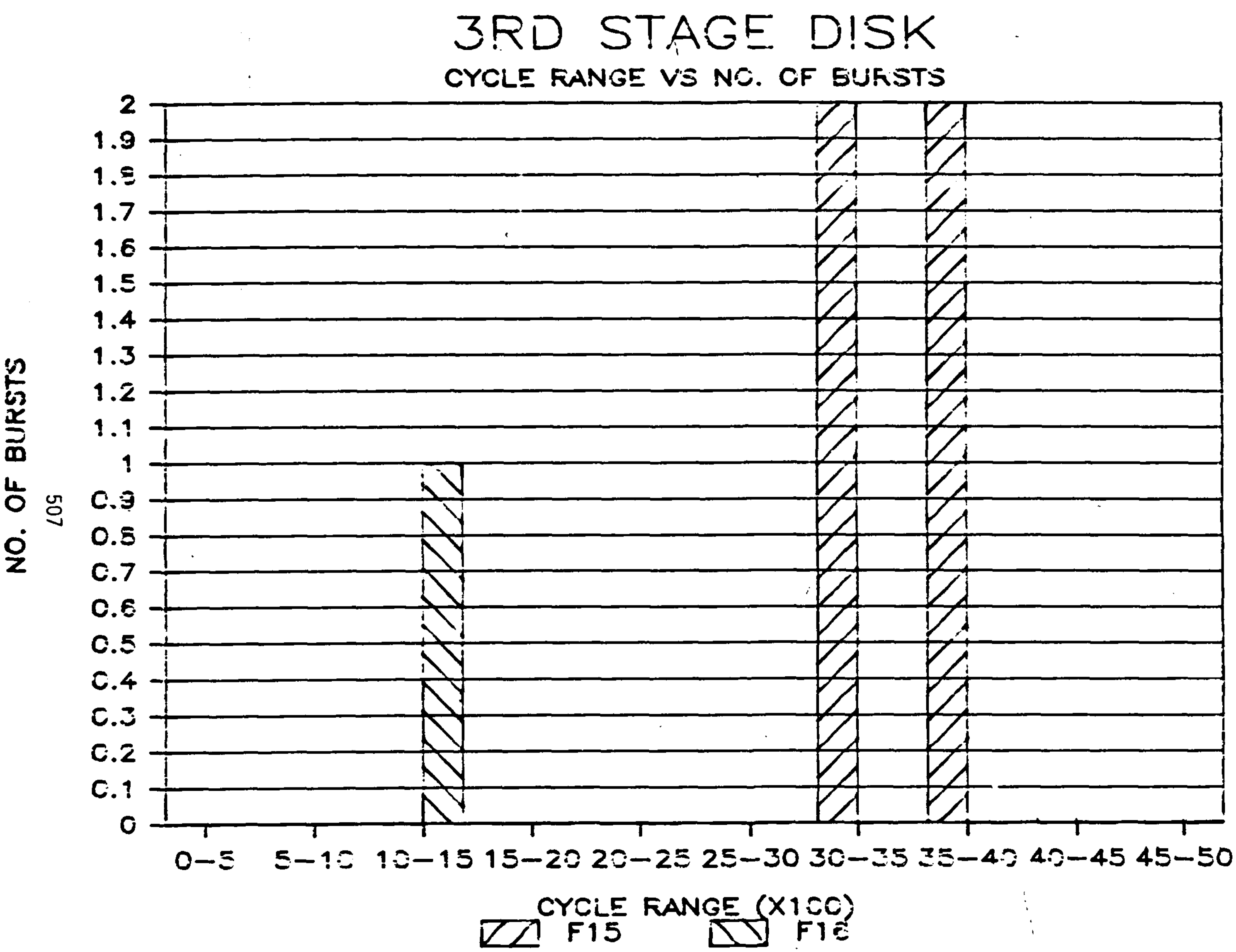

2ND STAGE DISK
CYCLE RANGE VS NO. OF BURSTS
NO. OF BURSTS
0
1
2
3
4
5
6
7
8
9
10
11
12
13
0-5
5-10
10-15
15-20
20-25
25-30
30-35
35-40
40-45
45-50
CYCLE RANGE (X100)
F15
F16

CHART 24 - BAR CHART 2ND STAGE DISK/CYCLE RANGE VS NO. OF BURSTS

			F15	(x-100)	F16	
CONCLUSION	1	BETWEEN	1	0-5	0	CYCLES
	0	BETWEEN	0	5-10	0	CYCLES
	1	BETWEEN	0	10-15	1	CYCLES
	18	BETWEEN	5	15-20	13	CYCLES
	5	BETWEEN	2	20-25	3	CYCLES
	7	BETWEEN	4	25-30	3	CYCLES
	4	BETWEEN	3	30-35	1	CYCLES
	1	BETWEEN	1	35-40	0	CYCLES
	0	BETWEEN	0	40-45	0	CYCLES
	0	BETWEEN	0	45-50	0	CYCLES
	37		16		21	

54.0% OF THE SECOND STAGE DISKS WHICH FAILED OCCURRED ON DISKS WITH 2000 CYCLES OR LESS.



3RD STAGE DISK
CYCLE RANGE VS NO. OF BURSTS
NO. OF BURSTS
2
1.9
1.8
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0–5
5–10
10–15
15–20
20–25
25–30
30–35
35–40
40–45
45–50
CYCLE RANGE (X100)
F15
F16

CHART 25 - BAR CHART   3RD STAGE DISK/CYCLE RANGE VS NO. OF BURSTS

			F15	(x 100)	F16	
CONCLUSION	0	BETWEEN	0	0-5	0	CYCLES
	0	BETWEEN	0	5-10	0	CYCLES
	1	BETWEEN	0	10-15	1	CYCLES
	0	BETWEEN	0	15-20	0	CYCLES
	0	BETWEEN	0	20-25	0	CYCLES
	0	BETWEEN	0	25-30	0	CYCLES
	2	BETWEEN	2	30-35	0	CYCLES
	2	BETWEEN	2	35-40	0	CYCLES
	0	BETWEEN	0	40-45	0	CYCLES
	0	BETWEEN	0	45-50	0	CYCLES
	5		4		1	

TOTAL   103

20% OF THE THIRD STAGE DISKS WHICH FAILED OCCURRED ON DISKS WITH 2000 CYCLES OR LESS.

CRYOGENIC SPIN TEST

Results (05/84 - 09/25/87)

Disk stage	NR spins	NR bursts*	Burst rate (%)*
1	2279	61	2.6
2	2235	37	1.6
3	2337	5	0.2
Totals	6851	103	1.5

Spin rate is 250 disks/month
*Burst frequency is result of test severity and is not quality related

AVA322091 870311

CHART 26

NOW, FOR THE RESULTS OF CRYOGENIC SPIN TEST OF F100 ENGINE FAN DISKS SINCE 1984. FIRST, 6851, 103 AND 1.5% ARE THE TOTAL NUMBER OF SPINS, NUMBER OF BURSTS, AND THE BURST RATE, RESPECTIVELY AS OF 09/25/87. THE BURST RATE IS WELL ABOVE THAT EXPECTED/PREDICTED BASED ON QUALITY CONSIDERATIONS.

THE BURST OCCURRENCES COME IN GROUPS WHICH CANNOT BE CORRELATED TO HEAT CODE, INTEGRAL TEST RING PROPERTIES, PROCESSING CHANGES, TIME OF DAY OR ANY OTHER PARAMETER. ONE MONTH WINDOWS OF DATA WILL REVEAL PREDOMINANTLY 1ST STAGE BURSTS, OR PREDOMINANTLY 2ND STAGE BURSTS, OR NO BURSTS, OR AN ISOLATED 3RD STAGE BURST, ETC.; ALL FOR A RELATIVELY CONSTANT SPIN RATE PER MONTH.

AN IMPORTANT OBSERVATION THAT I WANT TO MAKE IS THAT THIS BURST FREQUENCY IS A RESULT OF TEST SEVERITY AND IS NOT QUALITY RELATED.

F100 FAN CRYOGENIC SPIN TEST HISTORY

One hundred and three cryo fractures as of 09/25/87

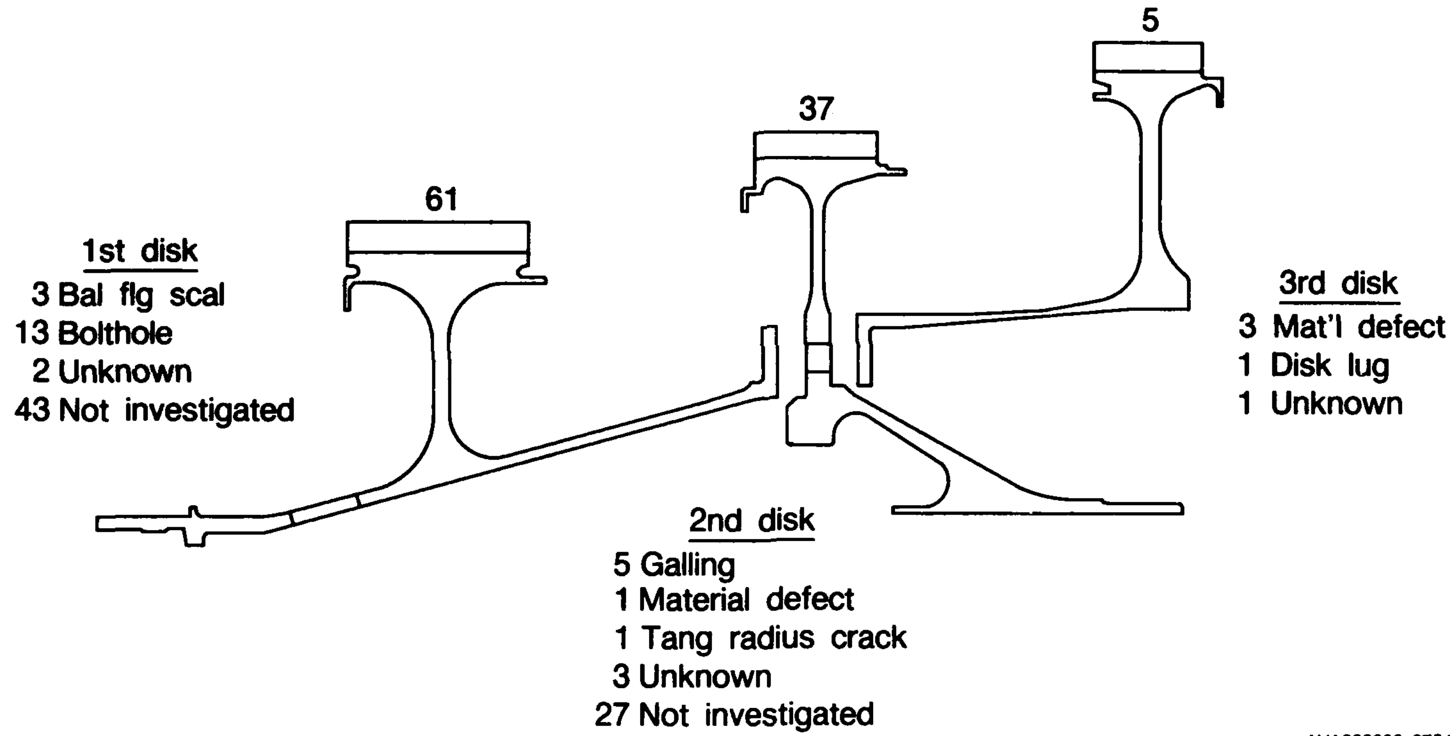


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CHART 27

AS PART OF THE INITIAL IMPLEMENTATION OF CRYOGENIC PROOF TEST, BURST DISKS WERE COLLECTED AND EXAMINED TO IDENTIFY FAILURE ORIGINS AND CAUSES. THIS PROCEDURE HAS GRADUALLY CHANGED TO THAT OF INVESTIGATING UNUSUAL BURSTS, SUCH AS BURSTS THAT OCCUR AT LOW SPIN SPEED OR 3RD STAGE DISK BURSTS.

GALLING/FRETTING OF THE BOLTHOLES IS INDICATED TO BE THE INITIATION CAUSE FOR MANY OF THE 2ND STAGE AND 1ST STAGE BURSTS.

SUBSURFACE MATERIAL DEFECTS HAVE BEEN CONFIRMED AS THE CAUSE FOR ONE 2ND STAGE BURST AND THREE 3RD STAGE BURSTS.

FATIGUE CRACKING AT A STRESS CONCENTRATION AREA DUE TO SURFACE CONDITION WAS THE BURST CAUSE FOR A 2ND STAGE DISK.

FRETTING AND FATIGUE AT A 3RD STAGE DISK LUG WAS THE BURST INITIATION SITE IN ANOTHER INCIDENT.

LOWERED NOTCH STRENGTH, INABILITY TO REDISTRIBUTE STRESS AND GENERAL SEVERITY OF THE TEST ARE ALSO THOUGHT TO BE CAUSES FOR BURSTS.

SUMMARY

F100 fan disk cryogenic proof test

- An integral part of F100 engine life management
- Critical quality and cyclic related distress has been culled
- Damage tolerance criteria is important for design of high strength materials
- Reflects super cooperative effort between U.S. Air Force and Pratt & Whitney

AVA322093 870311

CHART 28

- IN SUMMARY, FAN DISK CRYOGENIC PROOF TEST IS AN INTEGRAL PART OF F100 ENGINE LIFE MANAGEMENT AND HAS PERMITTED THE DEPOT RETURN INTERVAL FOR FAN MODULES TO BE EXTENDED FROM 1800 TAC CYCLES TO 3000 TAC CYCLES (FROM 4 YEARS TO 7 YEARS). PROJECTED LCC SAVINGS ARE SIGNIFICANT.
- CRITICAL QUALITY AND CYCLIC RELATED DISTRESS HAS BEEN CULLED FROM SERVICE ENGINES BY CRYOGENIC PROOF TEST.
- EXPERIENCE WITH THE F100 ENGINE FAN DESIGN, CRYOGENIC PROOF TEST AND REDESIGN HAS AGAIN HIGHLIGHTED THE IMPORTANCE OF UTILIZING DAMAGE TOLERANCE CRITERIA IN THE DESIGN OF FRACTURE CRITICAL PARTS UTILIZING HIGH STRENGTH MATERIALS.
- FOR THOSE OF US CLOSE TO THE F100 ENGINE, IMPLEMENTATION OF CRYOGENIC PROOF TEST IS THE RESULT OF A TREMENDOUS AMOUNT OF EFFORT AND COOPERATION BETWEEN THE U.S. AIR FORCE AND PRATT AND WHITNEY.

THANK YOU!

RETIREMENT FOR CAUSE OF THE F100 ENGINE

John A. Harris, Jr.
M. C. Van Wanderham
Pratt & Whitney
Advanced Engineering Operations
West Palm Beach, Florida

ABSTRACT

Retirement for Cause (RFC) is a life cycle management procedure for gas turbine engine components, such as fan, compressor and turbine disks. The procedure enables full use of the safe life inherent in each component, as opposed to arbitrary retirement from service of all components at a calculated low cycle fatigue life. Historically, these components have been retired at the accumulated time (or cycles) where the first fatigue crack in 1000 identical components, all used in an identical manner, could be expected to occur. By definition then, 99.9% of these components were being retired prematurely, while they still may have had useful life remaining. The Retirement for Cause approach is based on fracture mechanics and nondestructive evaluation, and is evaluated economically. The U.S. Air Force recognized the potential of this approach for maintenance/life cycle cost savings and began development programs in the late 1970's and early 1980's to reduce the RFC concept to practice. Those programs have been successfully completed. This paper discusses the development and integration of the methodology, its implementation for 23 USAF F100 engine components by the San Antonio Air Logistics Center, and its economic and other benefits.

Retirement for Cause Methodology

The methods used for predicting the life of gas turbine engine rotor components have historically resulted in a conservative estimate of useful life. Most rotor components are life-limited by low cycle fatigue (LCF) generally expressed in terms of mission equivalency cycles or engine operational hours. When some predetermined life limit was reached, used components were retired from service and replaced with new components.

The fatigue process for a typical rotor component such as a disk can be visualized as illustrated in Figure 1. Total fatigue life consists of a crack initiation phase followed by growth and linkup of microcracks. The resulting microcrack(s) would then propagate subcritically until the combination of service load (stress) and crack size exceeded the material fracture toughness. Catastrophic failure would ultimately result had not the component been retired from service. To preclude such failures, disks have typically been retired at the time when 1 in 1000 would be predicted to have initiated a short (.030 inch) fatigue crack. By definition then, 99.9% of the disks are retired prematurely. This results from the fact that all fatigue data have inherent scatter. When this data scatter is coupled with other uncertainties in any design system, (e.g., stress analysis error, mission/usage variability, fabrication tolerance, temperature uncertainty) a final deterministic life prediction is made for an occurrence rate of 1 in 1000. When plotted on a life distribution curve as in Figure 2, this corresponds to approximately a -3 sigma lower bound. It is at this life that all fatigue limited disks are removed from service. This procedure has successfully prevented catastrophic

in-service failures. However, in retiring 1000 disks because one may crack, the remaining crack initiation life of the 999 theoretically good disks (shaded area in distribution curve of Figure 2.) is not utilized. It has been calculated that many of the 999 retired disks have considerable useful remaining life (Figure 3).

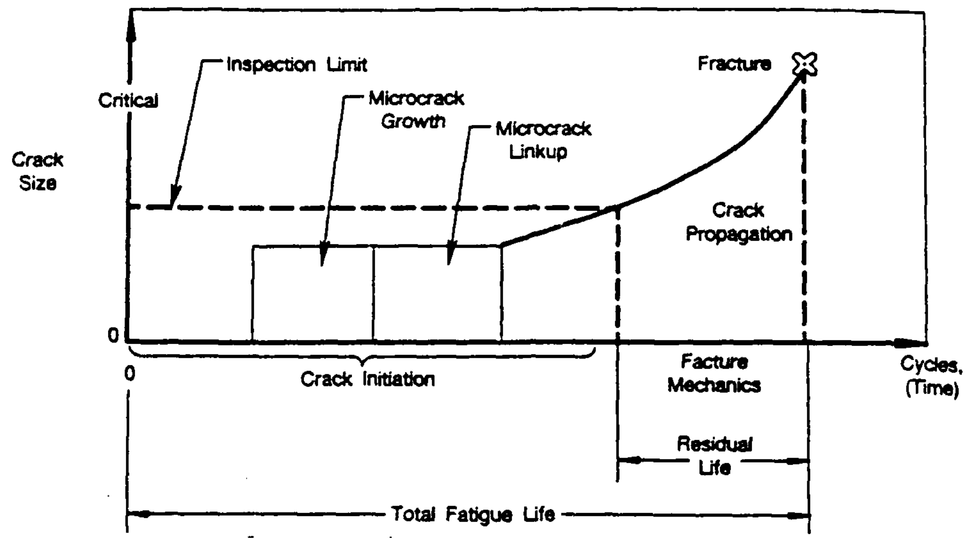

Figure 1. Total Fatigue Life Can-Be Segmented Into Stages of Crack Development, Subcritical Growth and Final Fracture

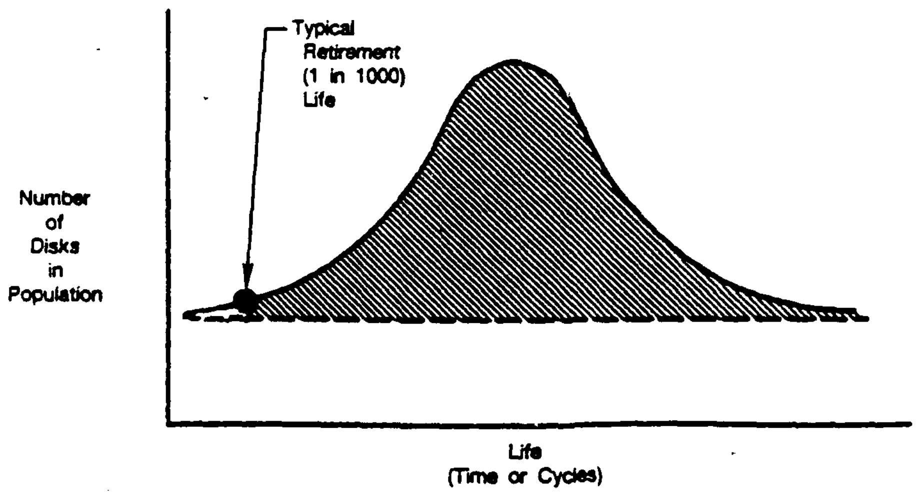

Figure 2. Historical Life Limit Methodologies Have Precluded Use of All Available Life in a Population of Disks

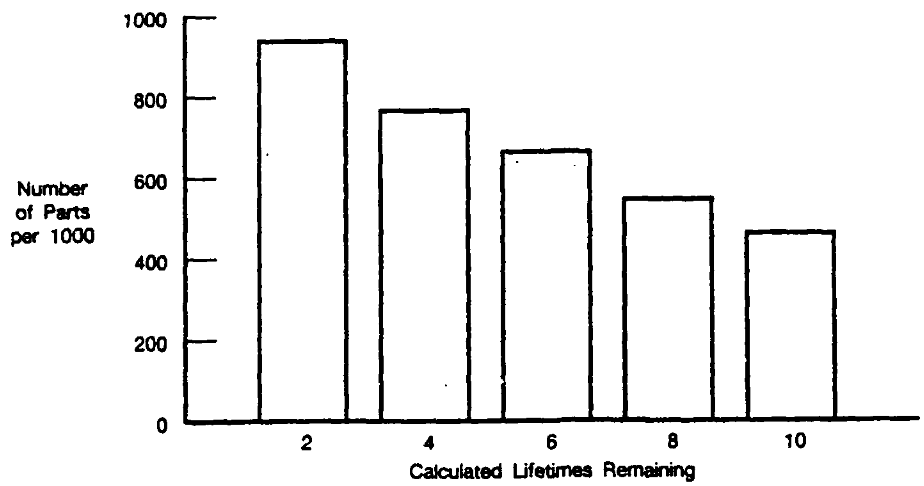

Figure 3. Useable Life in Excess of the Typical Components 1 in 1000 Retirement Life Can Be Significant for a Component Population

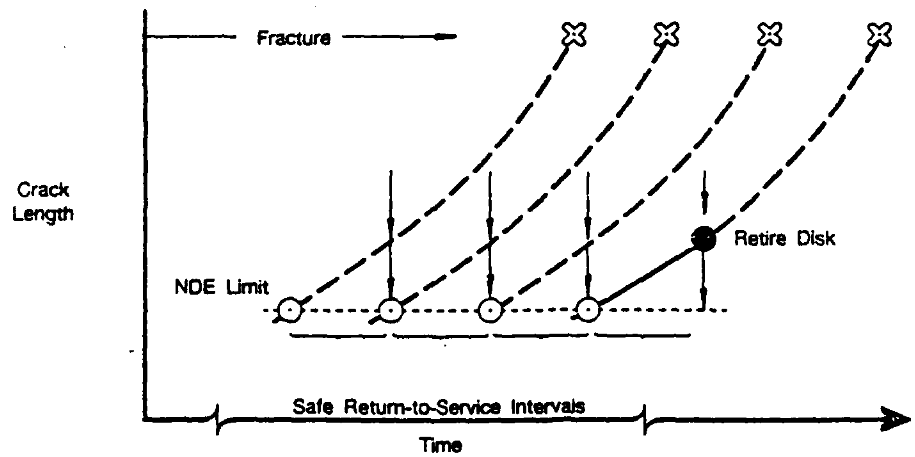

Figure 4. The Retirement for Cause Procedure Involves Inspection and Return-to-Service Until a Quantifiable Defect Is Found, Resulting in Retirement.

The ability to safely utilize the remaining life in that population of 999 retired disks had been limited by the understanding of the fatigue and fracture process. Technology advances have improved that understanding, resulting in the ability to eliminate or define the uncertainties in life prediction, thus enabling the Retirement for Cause approach. Under the RFC philosophy, each of these retired disks could be inspected and, if sound, returned to service. The return-to-service interval is determined by a fracture mechanics calculation of

remaining propagation life from a defect just small enough to have been missed during inspection. If the propagation life from this "missed" defect exceeds the economically feasible return-to-service interval, the disk is reused. This procedure is repeated, as shown in Figure 4, until the disk has incurred detectable, measurable damage, at which time it is retired for that reason(or cause).

Referring to Figure 4, it can be seen that the RFC concept is based on fracture mechanics and nondestructive evaluation. Nondestructive evaluation is used to ascertain the presence or absence of defects in critical locations on a component. Fracture mechanics is used to predict the crack propagation life at every critical location from a defect size just below the NDE limit of reliable crack detection. Given that the technology exists to accurately do these two things, a third factor impacts the decision making process: is RFC economically beneficial?

Economic benefits of RFC are a function of the return to service interval-crack propagation life relationship. If the return to service interval is short, relative to crack propagation life, high costs may be incurred due to frequent return of modules or engines for depot inspection/overhaul. If the return to service interval is long relative to crack propagation life, high costs may be incurred due to in-service failure. The relationship between the return to service interval, and the crack propagation life, is defined as the propagation margin, and is illustrated in Figure 5.

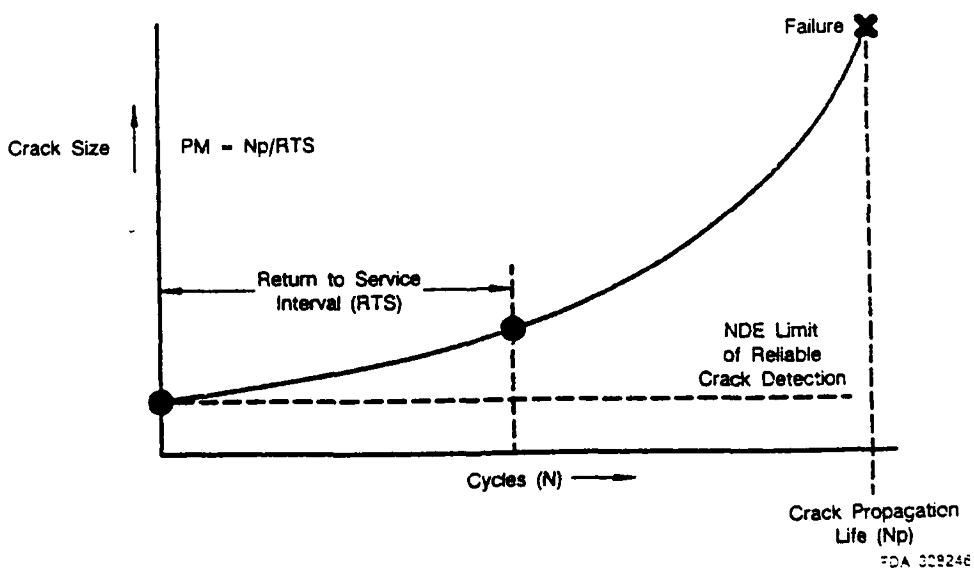

Figure 5. Propagation Margin (PM), Defines the Relationship Between Crack Propagation Life and Return-to-Service Interval. This Example Is for a PM=2.

Applying a propagation margin assures safety in utilizing the remaining initiation life in each component, recognizing that some uncertainties may still exist. This is done by determining the crack propagation life, Np, at every critical location on a component from a defect of a size barely small enough to be missed during inspection. The return to service interval, RTS, is then established by conducting life cycle cost analyses to determine the most economical propagation margin, PM, to apply to the shortest Np, thus RTS = Np/PM. In this context, propagation margin is akin to a safety factor. Life cycle cost versus propagation margin is plotted for each individual component and combined to determine the most economical interval to return an engine or module for inspection. An example is shown in Figure 6.

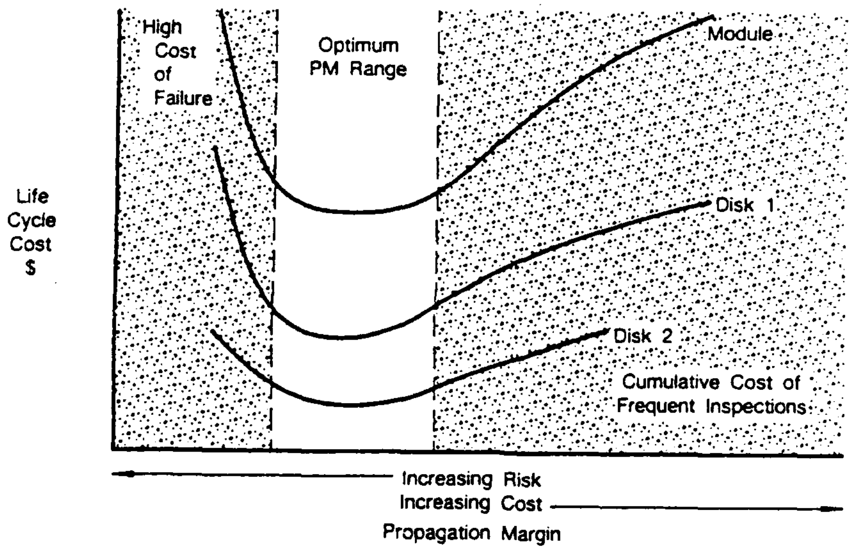

Figure 6. Propagation Margin Is Determined from an Economic Balance Between High Cost of Failure and Cumulative Cost of Frequent Inspection/Overhaul

The Retirement for Cause Procedure

The RFC flow chart, Figure 7, illustrates a simplified view of how this maintenance concept is utilized. When an engine or module is scheduled for maintenance, an economic analysis is performed on the engine or module, i.e., fan, compressor, high turbine, or low turbine, identified as a participant of the RFC maintenance program. If the module has already been in service for several inspection intervals, the probability of finding cracked parts may be great enough to make reinspection economically undesirable, and specific components of that module are retired without being inspected. This is determined by the economic analysis at decision point one and is one of three possible decisions. An unscheduled engine removal, UER, may bring a module out of service that is more economical to return to service for the remainder of its inspection interval than to inspect and release it for a new full interval, the second possible decision at point one. The remaining choice at

point one is to tear down the module and inspect the parts. During inspection, there again are three possibilities, (decision point two). If no defects are found, the part is returned to service. If the disk is found to be flawed, it is retired. The third choice is to investigate modification or repair of a flawed part. An economically repairable part may be repaired and returned to inspection, decision point three.

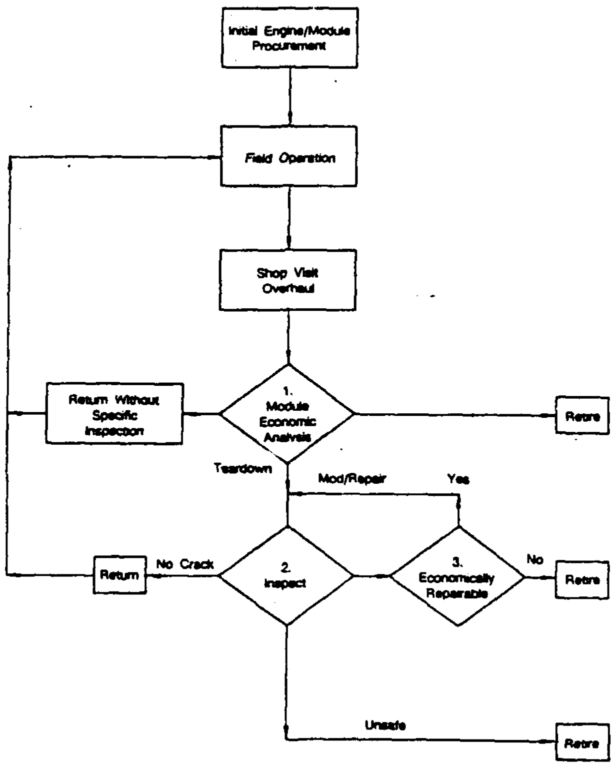

Figure 7. Retirement for Cause Procedure Flow Chart

Technical Development of RFC

As Figures 4, 5 and 6 have illustrated, the RFC methodology is based upon two broad technological areas: fracture mechanics and nondestructive evaluation, and evaluated economically.

It may seem incongruous that an effort to utilize the available crack initiation or LCF life inherent in a population of rotor components would emphasize fracture mechanics--crack propagation--knowledge as a requisite. Crack initiation phenomena are not completely understood, and the predictive ability is not precise. That is why, traditionally, lower bound limits are

used. The fracture mechanics--NDE approach of retirement for cause does not require knowledge of the exact crack initiation life of a component. The concern is if a crack initiates in a service interval, is it's growth predictable, with enough precision, to give assurance that the part would not fail prior to the completion of that interval? Should a crack have initiated, the inspection process at the end of the interval would then result in the retirement of that part long before failure could occur. The use of a propagation margin builds additional safety into the process. The understanding of how a crack will behave, if present, therefore, becomes the paramount concern in applying RFC.

The Materials and Acropropulsion units of the Air Force Wright Aeronautical Laboratories (AFWAL) have conducted in-house research and development activities in the RFC area since 1972. A joint study by the Metals Behavior Branch (AFWAL/MLLN), the Engine Assessment Branch (AFWAL/POTA), and the Directorate of Engineering, Aeronautical Systems Division, reference 1, was undertaken in 1975 to assess the state of the art of the technologies involved in RFC. This study addressed and utilized a TF33 3rd-stage turbine disk as a demonstration vehicle. As a result of this study, the technical requirements fell into four areas: stress analysis, crack growth analysis, nondestructive evaluation, and mechanical testing. Pratt & Whitney had also begun extensive research and development programs under corporate, IR&D, and government contract sponsorship in 1972 to identify and to develop the applied fracture mechanics and NDE technologies necessary to realize the RFC concept.

The culmination of these preliminary activities was a study conducted by P&W in 1979 and 1980 under Defense Advanced Research Projects Agency (DARPA) and AFWAL sponsorship entitled "Concept Definition: Retirement for Cause of F100 Rotor Components," reference 2. This program was the first to consolidate and focus these disciplines on a specific engine system and to quantify the benefits and risks involved. The methodology and results of the study program have been discussed at many workshops and symposiums (Reference 3-6).
Upon completion of that initial Concept Definition Study, AFWAL/Materials Laboratory established a major thrust in RFC with the goal of reducing the concept to practice with first system implementation to occur on the F100 engine at the San Antonio Air Logistics Center (SAALC). At the start of this major thrust, a number of fracture mechanics activities had been conducted giving high confidence in the ability to accurately predict crack growth. In particular, the Hyperbolic Sine (Sinh) procedure for modeling crack growth rate had been developed and validated. There did exist, however, some concerns and data gaps which conceivably could technically limit the application of RFC to specific engine components. In addition, the broad areas of economic assessment and logistics management had to be integrated with the RFC technical concept, to produce a viable, implementable system for managing life limited gas turbine engine components. The overall development process is shown schematically in Figure 8.

A number of the technology development activities were subsequently conducted to execute the development process. These activities addressed life assessment systems, nondestructive evaluation systems, and weapon system readiness concerns and the means of integrating and validating these disciplines for a coherent RFC methodology.

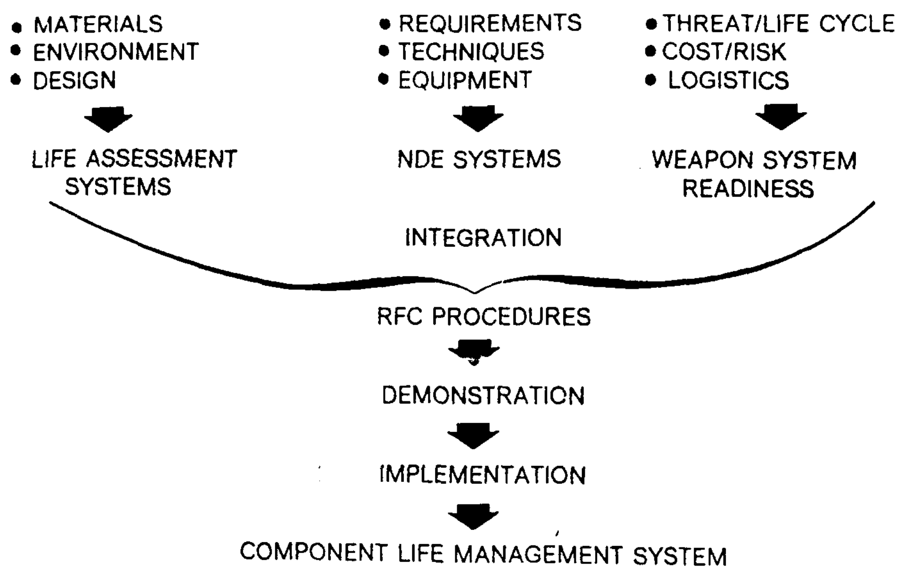

Figure 8. Technology Areas and Sequences Involved in Reducing Retirement for Cause to Practice

The life assessment systems activity developed fracture mechanics/crack growth tools for conditions present in gas turbine engines, such as thermal-mechanical cycling, multiple initiation sites, high frequency-low cycle fatigue interactions and others. As F100 engine components were the initial application of retirement for cause, the technology development efforts addressed the materials and conditions found in that engine.

A probabilistic life analysis technique had been previously defined and was developed. The information generated in the technology studies was used to refine this technique, or to provide inputs to the system. This probabilistic life analysis technique was then used in conjunction with thermal and stress analysis and nondestructive evaluation information to perform life prediction calculations, risk assessments, and sensitivity studies on specific F100 engine components selected for Retirement-for-Cause. Results of these analyses were, in turn, employed in comparisons of cost versus risk and the life cycle cost calculations to establish strategies for implementation of RFC, and to provide management information necessary for cost effective maintenance decisions.

Recognizing the concerns associated with the implementation of any new life-management concept, an extensive demonstration and validation effort was conducted. This activity demonstrated the tools previously developed by laboratory testing of specimens and subcomponents which verified the statistical aspects of the predictive models. In addition, spin-rig testing of turbine and compressor component assemblies verified the fracture mechanics tools. An Accelerated Mission Test (AMT) of a full F100 engine with purposely flawed components was conducted in the F100 Component Improvement Program, and the results were integrated into the RFC effort. This test, equivalent to more

than four years of service in the field, demonstrated fracture mechanics/life analysis tools under real engine operating conditions, and verified the laboratory specimen and component test results.

In addition to the laboratory and engine test results, information was obtained from F-15 and F-16 operating bases to confirm engine utilization, and from the San Antonio Air Logistic Center to assess field service performance of all F100 engine RFC Candidate Components. This information and data was used to verify and optimize the methodology.

While no development of NDE systems was conducted under this program, NDE support was provided during the methodology demonstration. Requirements for inspection of the candidate F100 RFC Components were defined, and were maintained current. This information was interacted with the Manufacturing Technology for RFC/NDE Systems program conducted by Systems Research Laboratories. In turn, information and results from that program were utilized in both the methodology demonstration and life assessment system development activities. Throughout the development process, similar coordination activities occurred with all of the effected Air Force and Government organizations.

Retirement for Cause and the Engine Structural Integrity Program

There is occasional confusion regarding the relationship of Retirement for Cause (RFC) and the Engine Structural Integrity Program (ENSIP). The Engine Structural Integrity Program is defined by Military Standard 1783 (USAF), and provides the basis for establishing the requirements, criteria, and methods for the design of gas turbine engines and/or components. Included is the requirement for damage tolerance in fracture critical parts, with fatigue crack initiation (LCF) life and crack propagation life criteria. In addition, certain nondestructive inspection criteria are specified.

Retirement for Cause is a component life management methodology. It may be applied to any life limited engine component, regardless of the criteria used in the design of that component. Retirement for Cause and ENSIP both draw from the same technology base, and both involve similar component analyses. In many instances they are complimentary: i.e. damage tolerance concepts of ENSIP can be utilized in RFC, and probabilistic analysis techniques developed for RFC can be utilized in ENSIP. In fact, much of the technology base of ENSIP has been demonstrated, and has been validated by the Air Force's RFC programs.

The major difference between the two programs is the point in time of application. ENSIP is applied in the initial design and development phase of an engine program. RFC is applied during the in-service, operational use, phase of an engine system. The use of the ENSIP philosophy for an engine design will greatly facilitate the use of RFC during its subsequent service life. As new engine systems entering the U.S. Air Force are being designed and developed under ENSIP criteria, RFC will emerge as the primary life limited component maintenance procedure.

In summary, use of RFC naturally accrues with an ENSIP designed engine; however, an ENSIP engine design is not required in order to apply Retirement for Cause.

The F100 Engine

The USAF F100 engine was chosen as the vehicle for the first implementation of the Retirement for Cause maintenance concept. This engine, in a number of models, powers the twin engined F-15 and the single engined F-16 fighter aircraft. It is an augmented turbofan engine in the 25,000 pound thrust class with a thrust to weight ratio approximately 8 to 1. The engine originally entered service in the early 1970's. There are in excess of 3,000 of these engines now in USAF service worldwide. The F100 is an axial flow, low-bypass, high-compression ratio, twin-spool engine with an annular combustor and common flow augmentor. It has a three-stage fan driven by a two-stage low pressure turbine (LPT), and a 10-stage high pressure compressor (HPC) driven by a two-stage, high-pressure turbine (HPT). The engine is shown in Figure 9. The engine consists of five major modules: fan, core (compressor, combustor, and compressor drive turbine), fan-drive (LPT) turbine, augmentor/exhaust nozzle and the accessory drive gearbox. Each module is interchangeable from engine-to-engine at the base and intermediate maintenance level. Each module has its own maintenance rhythm, and is returned to the San Antonio Air Logistic Center for overhaul/refurbishment independent from the other modules which constitute an engine.

The fan, core and fan drive turbine modules contain the rotor components - disks and rim spacers/air seals - considered for RFC.

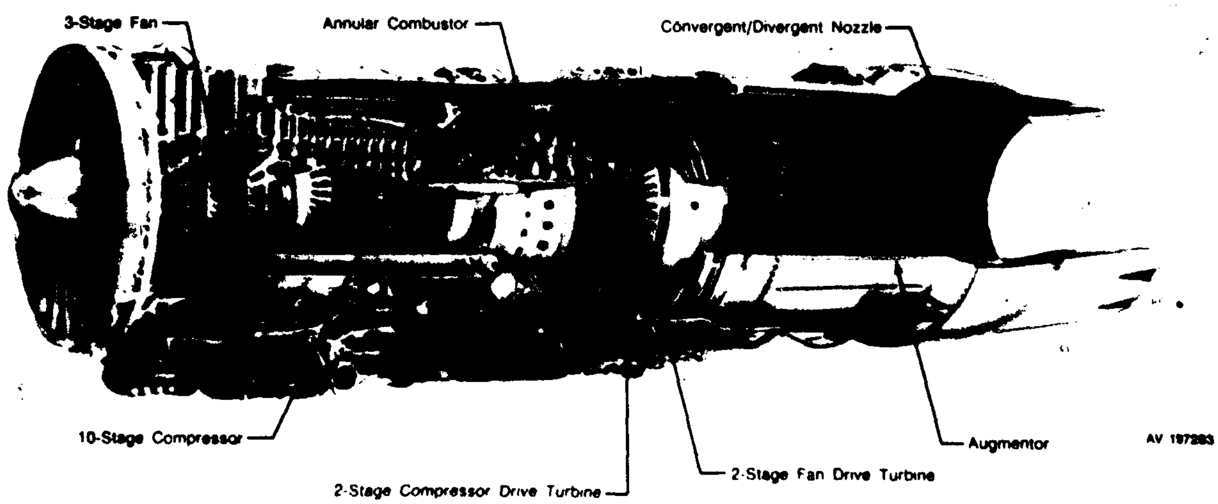

Figure 9. The F100 Turbofan Engine Which Powers the F 15 and F 16 Aircraft

Figure 9. The F100 Turbofan Engine Which Powers the F-15 and F-16 Aircraft

F100 Retirement for Cause Components

There are two primary criteria used to select F100, or any other engine, rotor components for RFC implementation. These criteria must be sequentially applied:

- Component low cycle fatigue life at any location is less than the anticipated system life;
- Component crack propagation life at all locations must be greater than the return to service interval with an appropriate Propagation Margin,

The critical locations considered for each of the two criteria may not be the same: that is why the criteria are applied sequentially. The LCF limiting locations may not be those with the most limiting crack propagation life. Failure to meet both criteria removes a component from consideration. For example, a component with a LCF life greater than the engine system life would not be retired in the life of the system, therefore, RFC is not applicable. Conversely for a component with a LCF life less than the engine system life, but with a low propagation margin, RFC is also not applicable, assuming no change in return to service interval, and the component would be retired and replaced at its LCF limit.

There are two additional factors that should also be considered in selecting candidate components.If the cost of conducting the necessary component inspections exceeds the cost of a new replacement component, it is not economically feasible to apply RFC. If replacement components are not available (-at any cost-) in the required time period, RFC may be necessary in order to maintain force readiness. These factors were also accounted for in selecting F100 RFC components.

At the present time, 23 fan, high pressure compressor and fan drive turbine disks and airseals/spacers are F100 RFC components. These components and their materials are listed in Table 1.

Coupled with fracture mechanics analyses, reliable production oriented nondestructive inspection techniques are required. The inspection requirements for each critical feature for each RFC component were established. Typical flaw inspection types are shown in Figure 10. The appropriate NDE techniques to meet those requirements are also listed in Table 1, and are of four types: proof test, eddy current, ultrasonic, and fluorescent penetrant.

Proof testing techniques at cryogenic temperatures are used for the three fan disks. The cryo-proof process for disks was developed in conjunction with the Aeronautical Systems Division and San Antonio Air Logistics Center. Under this process, a fan disk is spun to an overload speed while at cryogenic temperature. If a deleterious defect is present, the disk bursts: if not, the disk is certified for its next return to service interval.

All components receive standard fluorescent penetrant inspection (FPI) during the overhaul process to detect gross surface defects. Focused FPI is used for RFC inspections of some features where flaw size requirements are less stringent.

Table 1
F100 Engine Retirement For Cause Components

Module	Component	Material	NDE Technique*
Fan	1st Stage Disk and Hub	Ti 6-2-4-6	CRYO Proof
	2nd Stage Disk and Hub	Ti 6-2-4-6	CRYO Proof
	3rd Stage Disk	Ti 6-2-4-6	CRYO Proof
	2nd Stage Airseal	Ti 6-2-4-6	EC
Core-HPC	4th Stage Disk	Ti 6-2-4-6	EC
	5th Stage Disk	Ti 6-2-4-6	EC
	7th Stage Disk	WASPALOY	EC
	8th Stage Disk	WASPALOY	EC
	9th Stage Disk	In-100	EC/UT
	10th Stage Disk	WASPALOY	EC
	11th Stage Disk	In-100	EC
	12th Stage Disk	WASPALOY	EC
	13th Stage Disk	In-100	EC/UT
	13th Stage Rotor Spacer	In-100	EC/UT
	4th Stage Airseal	Ti 6-2-4-6	FPI
	5th Stage Airseal	Ti 6-2-4-6	FPI
	6th Stage Airseal	WASPALOY	FPI
	7th Stage Airseal	WASPALOY	FPI
	8th Stage Airseal	WASPALOY	EC
	9th Stage Airseal	WASPALOY	FPI
LPT	3rd Stage Turbine Disk	In-100	EC/UT
	4th Stage Turbine Disk	In-100	EC/UT
	4th Stage Turbine Airseal	WASPALOY	EC

* EC=Eddy Current feature inspection
UT=Ultrasonic zone inspection
FPI=Fluorescent penetrant inspection

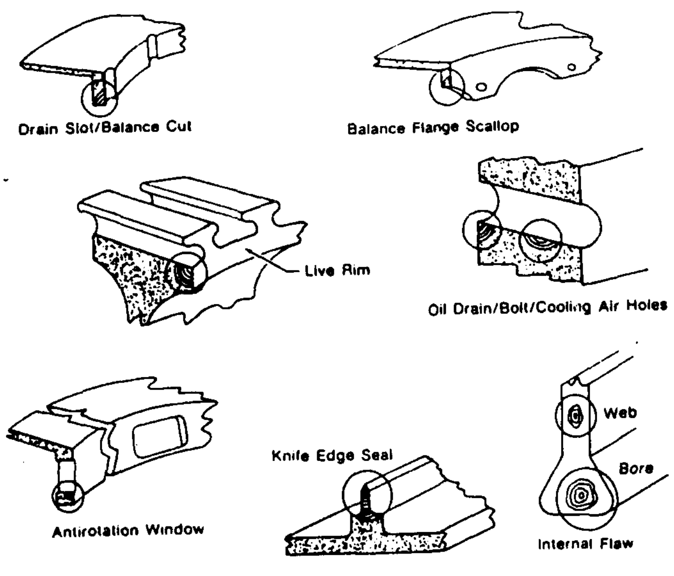

Figure 10. Typical Flaw Types to Be Detected in Fighter Engine Rotor Components

Eddy current and ultrasonic techniques are used where critical surface and/or subsurface (internal volumetric defect) inspections are required. Specific inspection requirements were established for each component location or zone and interacted with Systems Research Laboratories Inc. (SRL), the developer and producer of the automated eddy current/ultrasonic inspection system for RFC under USAF Contract F33615-81-C-5002. That system is now installed and operational at the San Antonio Air Logistic Center.

Benefits of RFC for the F100 Engine

The underlying fracture mechanics/life prediction and nondestructive evaluation technology basis of RFC has been demonstrated and validated. There are no technical reasons why RFC can not be used for engine maintenance. However, there must be a reason to use RFC. The primary premise for utilizing RFC is that it will significantly reduce the cost of ownership of an engine system. Life Cycle Cost (LCC) savings are the means of measuring the reduction in cost of ownership.

Life Cycle Cost analyses were conducted for two scenarios for the F100 fleet: baseline, with components retired and replaced with new components at their LCF limits; and RFC of the 23 components. The difference in the two analyses results is the LCC savings for RFC. In addition, as RFC contributes to an increase in the return-to-service of the core engine module, with resultant reduction in the number of core engine overhauls required, the cost savings for this reduced overhaul activity was also established.

Life Cycle Costs analyses are predicated upon various ground rules and assumptions concerning events in the future. These include aircraft/engine utilization rates, scheduled and unscheduled maintenance events, engine delivery schedules, engine/module retirement rate, labor requirements and many others. Ground rules and assumptions for this study were developed in conjunction with the Tactical Egines Program Office (ASD/YZ), Air Logistics Command Headquarters, Acquisition Logistics Division, San Antonio Air Logistics Center and AFWAL, and used with their concurrence. All costs were based on fiscal year 1986 dollars.

The LCC savings for the F100 fleet are $966.2 million due to parts cost avoidance for the nominal engine life. These savings are delineated by module in Table 2. In addition, $655.2 million is saved because of fewer scheduled core engine overhaul visits for the fleet over its projected life. This reduction results from the extension of the core engine return-to-service interval. RFC is responsible for a part of this savings. The cost of extending the interval would increase by a factor of approximately 1.7 to 1.8 if RFC was not used. Therefore, it is postulated that RFC is responsible for a proportionate amount of the savings due to the fewer core engine overhaul visits. With this premise, RFC can be credited with approximately $303 million of the overhaul visit cost avoidance.

The gross savings due to RFC must be debited by the investment costs required to enable its use. These cost accrue due to development, equipment and facility expenses to enable its application. A total of $52.5 million (FY 1986 dollars) including the NDE and cryo proof equipment and facilities, are considered RFC investment costs in this analysis and are deducted from the LCC savings.

Table 2
Life Cycle Cost (LCC) Savings For F100
Engine Component Retirement For Cause

Module	Number of Components	LCC Savings $ Million (1986)
Fan	4	263.7
Core-HPC	16	449.3
LPT	3	258.2
Total Parts Cost Avoidance		$ 966.2

With the above premises, the use of Retirement for Cause for the USAF F100 engine fleet produces a net life cycle cost savings in excess of 1.21 billion dollars.

Closure

Retirement for Cause of gas turbine engine components is technically valid and economically desirable. The benefits for the USAF F100 engine system are so significant that RFC was implemented for components of that engine in 1986 at the San Antonio Air Logistics Center. It is anticipated that RFC will become the standard procedure for maintenance management of life-limited components of all future USAF gas turbine engines, and potentially for any fatigue life-limited system.

In addition to the LCC savings, there are other, less tangible benefits. For the F100 engine alone, it is estimated that more than 3500 tons of strategic material use is avoided over the engine's life by not producing the quantities of spare parts that would have been required prior to RFC. In this regard, while not the primary objective, RFC evolves as one of the largest of the strategic material conservation efforts.

RFC has been demonstrated as technically viable and beneficial for the F100 engine. It is strongly emphasized, however, that in order to utilize RFC for other engines, thorough knowledge and understanding of the materials behavior, component operating conditions, engine use and logistics/maintenance systems involved are required to avoid undesirable consequences.

Acknowledgements

The Retirement for Cause concept as presented in this paper and as applied to the USAF F100 engine evolved from the work of many individuals in the academic, industrial and government communities. The authors acknowledge these contributions which have enabled RFC to move from an idea in 1972 to reality today for military gas turbine engines. Special acknowledgement is accorded to the Materials Laboratory of the Air Force Wright Aeronautical Laboratories for its management, sponsorship, and direction of the USAF RFC technology programs, and to the Defense Advanced Research Projects Agency for their sponsorship of the RFC technology base activities.

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A GENERAL DISCUSSION OF THE RETIREMENT FOR CAUSE INSPECTION SYSTEM

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ABSTRACT

Systems Research Laboratories, together with its many subcontractors, has developed the Retirement for Cause (RFC) system to inspect the F100 engine at the engine overhaul facility of Kelly AFB, San Antonio, Texas. This completely automated system, which utilizes both eddy current and ultrasonic inspection technologies, has been thoroughly tested before implementation at Kelly AFB. As of September 1, 1987, after ten and one half months in "production", the system has inspected approximately 1900 parts.

An overview of the RFC system and a presentation of the results of the reliability tests will be given. Some of the highlights of the inspection techniques and algorithms will also be outlined, in order to show how this automated, computer controlled system can achieve the reliable and sensitive inspections that have been demonstrated.

This work was conducted under USAF contract number F33615-81-C-5002.

INTRODUCTION

Systems Research Laboratories, together with its many subcontractors, has developed the Retirement for Cause (RFC) System to inspect the F100 engine at the engine overhaul facility of Kelly AFB, San Antonio, Texas. This completely automated system, which utilizes both eddy current and ultrasonic inspection technologies, has been thoroughly tested before implementation at Kelly AFB. As of October 15, 1987, after one year of "production", the system has inspected approximately 2300 parts.

An overview of the RFC System and a presentation of the results of the reliability tests will be given. Some of the highlights of the inspection techniques and algorithms will also be outlined in order to show how this automated, computer controlled system can achieve the reliable and sensitive inspections that have been demonstrated.

REVIEW

The Retirement for Cause System has been developed under contract with the Air Force. The project started in 1981. Since then, a prototype system has been developed and tested. Following this, a production system was built and then delivered. The first deliveries of inspection stations occurred late in 1985. Reliability tests have been conducted, and the system has been in production at Kelly AFB since October 1986.

The prime contractor has been Systems Research Laboratories of Dayton, Ohio. The contract monitor is Bruce Rasmussen. The subcontractors include major engine manufacturers (Pratt & Whitney Aircraft, General Electric, Garrett Turbine Engine, and Allison Gas Turbine), NDE research and equipment suppliers (Southwest Research Institute, Rockwell Science Center, AMES Laboratory, and Nortec), and mechanical module suppliers and designers (MM, CCDI, MMP, and MBS). Al Berens of the University of Dayton performed the statistical processing.

The result of this multi-million dollar, multiple subcontract effort is the generic system now in production at Kelly AFB. At present this system consists of four eddy current test stations and one ultrasonic test station, all interfaced to a mainframe computer. There is a system operator console which is used to monitor the status and progress of the inspection, as well as calling up data reports and graphics. The eddy current stations are used to find surface cracks, and the ultrasonic station is used to find subsurface defects.

The challenge has been to create a fast and reliable inspection searching for very small (0.005 x 0.010 in. surface, 0.020 in. dia. internal) fatigue cracks contained in complex geometry parts with equipment which must perform reliably in a factory environment. The type of geometries inspected include both basic geometries such as flat surfaces, corners, and internal volumes, and complex geometries such as the Live Rim, Scallops, Windows and Slots.

HIGHLIGHTS

The RFC System is completely automated. Some of the aspects of this automation that are particularly beneficial are part fixturing, probe pickup and return, dimensioning, calibration, and adaptive scanning techniques. In addition, the ultrasonic station adapts to the velocity changes of ultrasound due to water temperature variations. The accuracy and repeatability of some of these features is given in Table 1.

TABLE 1

Feature	Accuracy
Dimensioning	±0.002"
Bolt Hole Centering	<0.0006
Scallop Centering	±0.001
Phase Calibration	±2°
Gain Calibration	<1 dB

An important aspect of automation is algorithm portability. In this instance "algorithm" means that software which controls the inspection, also called the scan plan. For a scan plan to be portable, it must be able to execute properly on any of the stations without modification. The RFC System accomplishes this by keeping station dependent software resident within each inspection station computer. This coupled with adaptive scanning makes the scan plans truly portable, enabling the scan plans to be written from the blue prints of the engine parts.

Adaptive scanning techniques are those methods used to adapt the inspection to a given part, allowing for part-to-part and machine-to-machine variations. These techniques include hole centering, the use of compliant and airbearing probes, and dimensioning.

Dimensioning is the technique used to verify the part, its orientation, and its dimensions. It is also used to adjust the mechanical positions in the scan plan to that of the particular part being inspected, so that the integrity of the inspection is maintained.

Calibration of the inspection is also automatic. The goal behind calibration is to set up the system in a consistent way from day to day. It adjusts for differences in machines, instruments, and probes.

The inspection of the engine parts is done in two phases. The first phase is a rapid scan. Any suspect flaw regions are then re-inspected in the second phase using slower, more complicated signal processing techniques. In this two-phase approach, high throughput rates can be achieved while still setting the thresholds near the noise, and reducing the number of false calls. Some of the signal analysis techniques used in the RFC System are pattern recognition, digital filtering, correlation fourier analysis, and spatial and time averaging. The slower of these techniques are typically incorporated in the second phase of an inspection.

All of these system features and techniques work together to produce a reliable, fast, sensitive inspection system.

SYSTEM TESTING

The RFC System has been extensively tested. These tests began in October 1985. This acceptance/reliability test was just one phase of a technical reliability effort. The reliability working group developed the test procedure and test specimens. This group was made up of the major engine manufacturers, NDE research houses, and the University of Dayton[1]. The test specimen contained fatigue cracks in representative geometries. The test was blind, implying that neither the system nor SRL personnel had inspected the actual samples prior to the test.

The first phase of the reliability test was conducted at SRL in Dayton. The results of that test determined the ship status. The second phase of the reliability test was then conducted at Kelly AFB after shipment to confirm the capability shown before shipment.

Each phase of the test consisted of three parts: a test of the system on real engine parts, a test on reliability specimens, and a system operational reliability/dependability test. This last test evaluated the system with regard to uptime and downtime and subsystem failure. To date, the RFC System has been tested the equivalent of 16 consecutive 40-hour weeks.

The parameters tested in the NDE reliability portion of the tests are shown in Table 2. Also shown in the table are short explanations of the parameters.

TABLE 2

Parameter	Explanation
Basic Capability	Detection Capability
Repeatability	Multiple Inspection Effects
Reproducibility	Effects of Changes (Instrument...)
Variability	Effects of Human Parameters
Reliability	Composite of Above

Phase I test results are shown in Figure 1 and Table 3. Figure 1 shows the eddy current probability of detection of the fatigue cracks in the specimens showing the effects to this curve for several parameters and changes in the parameters[2]. As can be seen, the results show a good detection capability, good repeatability with little effect due to probe changes, load changes, or even operator changes. Table 3 gives the uptime for the three stations tested. This data shows very good system dependability.

TABLE 3. PERCENT UPTIME

Module	Hours	Mechanical	Software	Production
EC #1	126	86	99	82
EC #2	139	99	99	95
UT #1	147	100	98	96
Total	412	95	99	91

Phase II tests began in the summer of 1986. In addition to the tests conducted in Phase I, the system was tested under a full production load. The tests confirmed the system dependability, detection reliability, and the ability to handle a full production load. They also established the system throughput rates that can be used to estimate loading of the machines in a factory environment. Figure 2 shows the eddy current probability of detection data for the rivet hole specimens for Phase II. Table 4 shows the eddy current detection capability for all of the specimens tested in Phase II. The depth given is the 90% probability of detection point at the 95% confidence level.

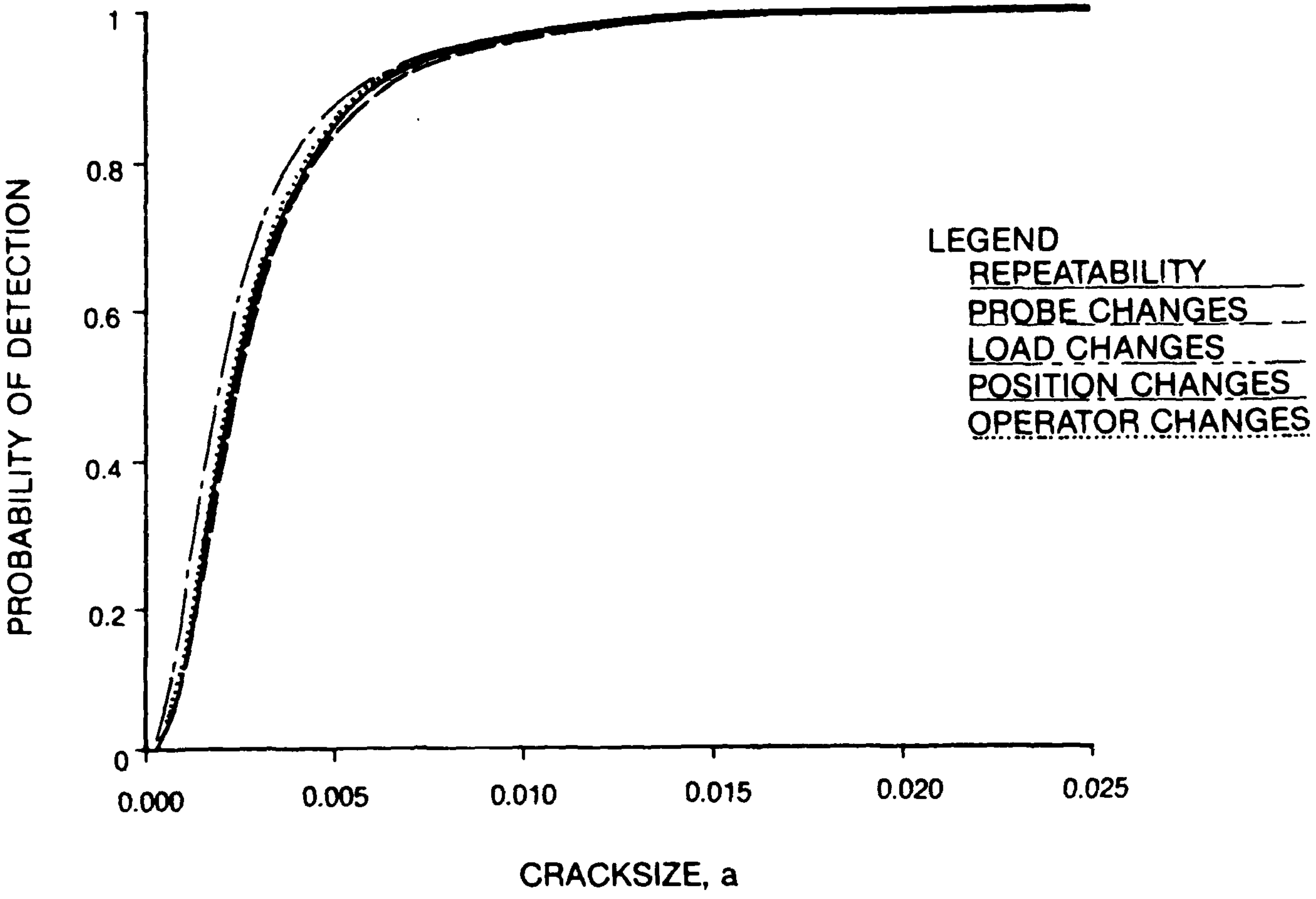

Figure 1. Phase I test results for the Rivet hole specimen.

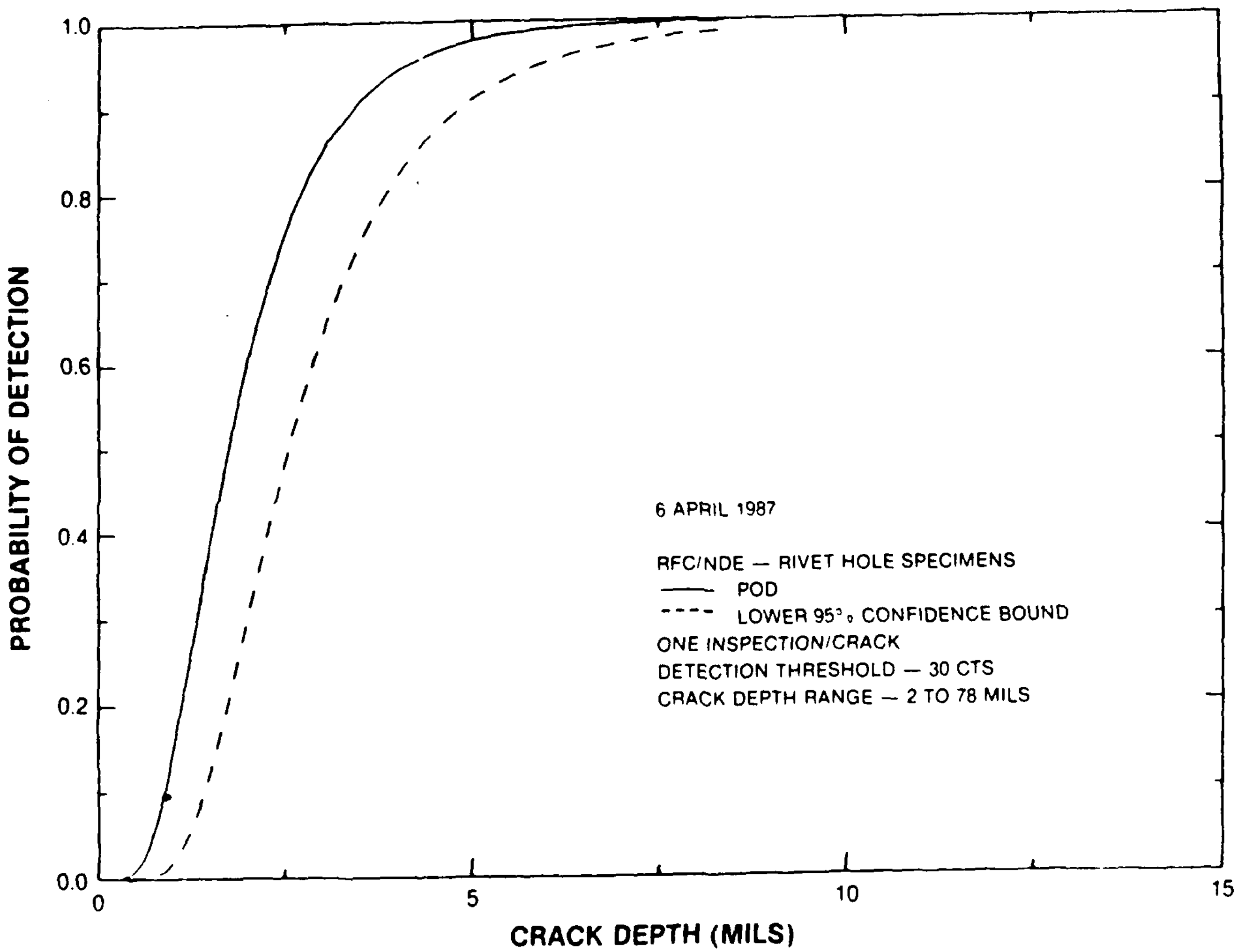

Figure 2. Phase II test results for the Rivet hole specimen.

TABLE 4. RFC CAPABILITY

Specimen	Depth (mils)
Rivet Holes	4.8
Bolt Holes	7.2
Web/Bore - IN100	5.1
Web/Bore - Waspaloy	7.4
Web/Bore - Ti	9.5
Scallops	5.0
Snap Fillet	38.0*

As can be seen from the depth in Table 4, not all of the geometries tested have the desired probability of detection (POD). These areas, together with other difficult geometries, such as antirotation windows and tangs, are still being worked on in order to develop a reliable technique. Therefore, of the parts presently in production certain geometries are not being inspected using the RFC system.

One of the results of the reliability tests has been the generation of theoretical POD verses threshold tables. Based on these tables, certain geometries, at the request of the Air Force, had their thresholds raised. These specimens were subsequently tested with these raised thresholds, and the theory confirmed. Finally, these raised thresholds were incorporated in the engine part inspection software.

* This result was found to be partially due to missed flaws that were outside of the inspection region. Taking those samples out of the population, and other improvements has reduced this number considerably.

In summary, the system has shown its capability to inspect these engine parts for flaws down to 0.005 x 0.010 inches (surface) and 0.026 in. dia. (internal) with a high probability of detection. Phase II data correlated strongly with Phase I. Finally, the system has shown excellent repeatability, and very small variability due to changes in probe, load, orientation and operator.

PRODUCTION

On October 15, 1986, the RFC system went into production on 11 compressor section engine parts. As mentioned before, not all geometries on these parts have been implemented. These geometries are awaiting for the development efforts to improve their reliability. And yet, the system has outperformed its expectation for the geometries that are inspected. For the first year in production, there has been an average of 30 cores/month inspected. And for the geometries and parts that are being inspected, the estimated inspection time/core was 38 hours, while the actual time/core is 28 hours.

The overall statistics are as follows: 2324 engine parts have been inspected in the first year. 680 were accepted on the first inspection. A total of 1955 were accepted. 148 were rejected. Those that have not been accepted nor rejected were in the process of being inspected, or reworked, etc. at the time of compilation of these statistics.

CONCLUSION

The RFC system has proven its capabilities both on tests at SRL and Kelly AFB, as well as in production. It is proving to be a reliable, cost effective means of extending the life of these costly engine components. This successful effort will save the Government much money and strategic materials in the years to come.

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SESSION IV: MATERIALS/TRACKING

COMPUTERIZED CORROSION FORECASTING MODEL FOR C-5 AIRCRAFT

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Fred H. Meyer
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ABSTRACT

A predictive corrosion model which will enable optimum corrosion maintenance scheduling for C-5 aircraft is being developed under Air Force Contract F33615-85-C-5058. The VAX-11 computer program is based on the kinetics of corrosion of aircraft alloys and upon the environmental conditions existing at Air Force bases. When completed, it will provide a fully integrated method for predicting crack growth, corrosion damage, and coating degradation for C-5 aircraft operating in a variety of environments.

The corrosion model is being validated by comparing predicted crack lengths with actual crack lengths in the test article used in the C-5A Modified Wing Structural Test Evaluation and by comparing predicted corrosion damage with corrosion control manhours expended on selected areas of the C-5B aircraft. In the cracks analyzed to date, the predicted crack lengths are very close to the actual lengths.

With only minor modification, the predictive corrosion modeling program may be used for the C-141, C-130, B-52 or any other aircraft fleet which already has a crack monitoring program in operation.

OPERATION

This is a report on a program Lockheed is conducting for the Air Force Aeronautical Laboratories, Air Force Systems Command.

Aircraft corrosion damage and paint degradation are accelerated by salt water, sunshine and acid rain. The present method of scheduling corrosion related maintenance is based on calendar time or flying hour intervals dictated by statistical probability of fatigue damage to structure or wear of engine parts. It does not take into consideration the wide variation in environmental conditions at Air Force Bases. Consequently, aircraft stationed in marine environments are not inspected often enough and others, based in dry areas, perhaps more frequently than required.

The objective of this program is to develop a corrosion prediction model which can be used to optimize:

1. Field and Depot Level inspection programs for existing aircraft.

2. Selection of aircraft for the Analytical Condition Inspection program, and

3. Initial inspection programs for new aircraft entering the Air Force inventory.

During the past two years a predictive corrosion model has been developed which is based on the kinetics of corrosion reactions of aircraft alloys and upon the environmental factors which exist at Air Force bases. The initial program was designed for C-5 aircraft but, with only minor modification, can be used for any aircraft which already has a crack monitoring program.

This program to incorporate corrosion rate data and prediction technology into inspection and maintenance scheduling consists of the following tasks:

Task 1 - Review and evaluate current Air Force maintenance programs and recent work on aircraft corrosion mechanisms and fracture mechanics.

Task 2 - Develop corrosion rate equations for aircraft corrosion processes and degradation rate equations for aircraft coating systems and incorporate them into a corrosion prediction model.

Task 3 - Convert the equations and models into a Vax-11 FORTRAN program to establish field and depot level inspection programs for aircraft already in operation, and inspection programs for new aircraft.

Task 4 - Validate the computerized corrosion forecasting models and maintenance scheduling decision logic by comparing the predictions of the model with actual corrosion histories of C-5 or C-141 aircraft.

Task 5 - Evaluate the efficiency of a logic which integrates the corrosion forecasting model with the structural integrity programs now in use by the Air Force.

PROGRESS

The key to this entire project is relating the kinetics of corrosion of aircraft alloys to the environments in which aircraft operate. The literature search disclosed much of the data necessary to construct the computer model. The environmental condition which exist at Air Force bases were obtained from a report by Dr. Robert Summitt of Michigan State University.(1) This report, "An Environmental Corrosion Severity Classification System", lists the atmospheric contaminants, intensity of sunlight, rainfall, distance to sea, dewpoint, and temperature data for most of the Air Force bases.

Fatigue cracking and stress corrosion data for aircraft alloys were found in the Damage Tolerance Design Handbook MCIC-HB-01, Part II (2).

The computer program is based on the following types of corrosion and coating failure:

1. Fatigue Corrosion
2. General Corrosion
3. Coating Degradation

FATIGUE CORROSION

Initial efforts were devoted toward developing a method for predicting and

tracking fatigue corrosion because it is the type of corrosion which is most apt to shorten aircraft life.

Because of the many variables involved, tracking crack growth in an air frame is a very complicated procedure. A crack monitoring program is already in use for the C-5 aircraft. It utilizes parametric input from crack growth tests, the Forman crack growth equation, data for the specific alloys involved, stress spectra relating to the type of mission, correction factors for the specific geometry of the crack being monitored, and adjustments for retardation effects caused by higher than average stresses. This program utilizes NASTRAN's Finite Element Analysis methods for calculating stress levels at selected points of the C-5 airframe.(3)

NASTRAN is a computer program designed to solve mathematical models for problems in continuum mechanics. It embodies a lumped element approach, wherein the distributed physical properties of a structure are represented by a model consisting of a finite number of idealized elements that are interconnected at a finite number of grid points. Loads are applied at these grid points and solutions to complex stress and displacement problems can be obtained. A typical NASTRAN model consists of membrane elements, rod elements and fastener elements as illustrated in Figure 1.

The C-5 crack monitoring program assumes a flaw of 0.05 inches at critical points on the aircraft at the time of manufacture, and calculates the length to which a crack would grow as the result of the missions the aircraft has flown. The program is based on crack growth data for aircraft alloys obtained at 100% relative humidity conditions.

Corrosivity Factors

Instead of repeating all this programming for the predictive corrosion model, it was decided to use the crack lengths predicted by the C-5 Crack Monitoring program and correct them for variations in the environments in which specific aircraft operate. This is being done by multiplying the crack lengths based on 100% relative humidity data by a time weighted "Corrosivity Factor" which is based on the severity of the environment in which an aircraft operates.

The basic Forman crack growth equation used in the C-5 program is

$$da/dN = \frac{C(\text{delta } K)}{(1-R)Kc - \text{delta } K}$$

where

da/dN = crack growth per cycle

delta K = difference between the maximum and minimum values of stress concentration

Kc = the critical stress intensity for fracture

C = a material constant

R = ratio of minimum to maximum load

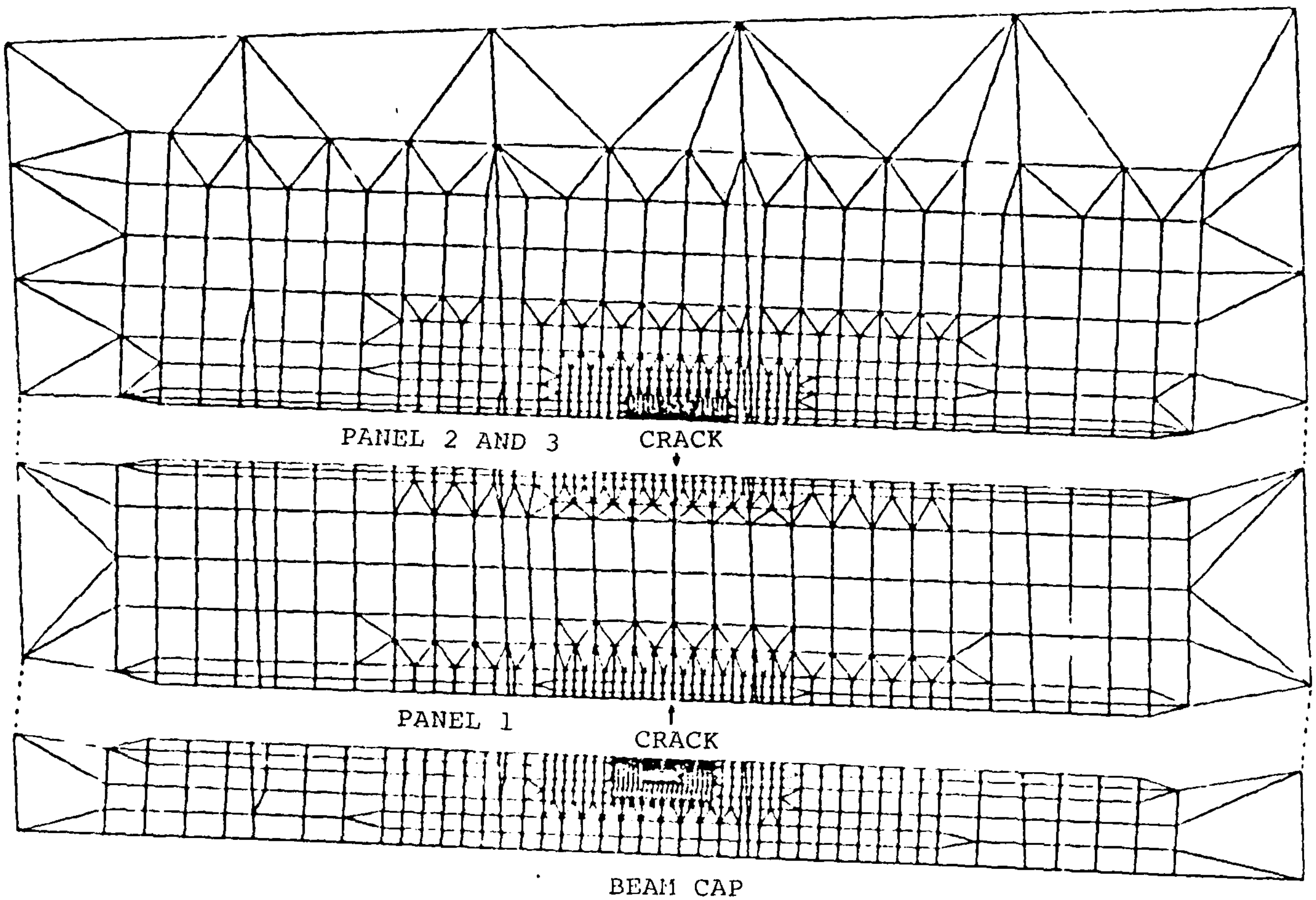


FIGURE 1. EXAMPLE OF FINITE ELEMENT MODEL

The Corrosivity Factor, CF, is defined as

$$\frac{\text{da/dN in Actual Environment}}{\text{da/dN in 100\% Humidity Environment}}$$

The Corrosivity Factors were determined by plotting da/dN vs delta K data for specific aircraft alloys in dry air, distilled water, and in 3 1/2% NaCl solution on the same plot as illustrated in Figure 2. Then, assuming an average stress intensity of 10 KSI in, the values of da/dN for each environment were read off and converted to corrosivity factors using the above formula. Since 3 1/2% NaCl is a more corrosive environment than 100% humidity, the factor for a salt water environment would be greater than 1. Conversely, for a dry environment, the factor is less than 1.

Using Dr. Summitt's data for Air Force bases (1), each base was assigned a set of corrosivity factors corresponding to its environmental conditions. Factors were calculated for 7075-T6 and 7075-T73 aluminum, and for 4340 and 300M steel. These were included in the computer programming in such a manner that, when a specific Air Force base is designated, the appropriate factor is automatically used in the calculation.

Tracking Points

The C-5 crack monitoring program now in use at Oklahoma City ALC, Tinker Air Force base tracks 46 theoretical cracks (4). Figure 3 shows the location of seven of these points which have been selected for use in the initial predictive corrosion modeling computer program.

1. Nos 629 and 425 - spanwise splices on the lower wing surface
2. Nos 525 and 325 - spanwise splices on the upper wing surface
3. No 761 - skin on the upper fuselage
4. No 818 - panel splice on the vertical stabilizer
5. No 852 - spanwise splice on the upper surface of the horizontal stabilizer

GENERAL CORROSION

Damage functions for metals in contaminated environments frequently follow the general model

$$M = At^b$$

where M is the metal loss by corrosion, t is exposure time, and A and b are empirical constants determined by the environmental conditions, the metal involved and the type of corrosion product on the metal.(5)

The exponent b theoretically takes on the value of approximately 1/2 when corrosion is limited by the diffusion rate of the reactive species through a semi-permeable film of reaction products. This would be the case for most aluminum alloys. When the corrosion products are flocculant or soluble and offer no protection, as is generally true for steel, linear corrosion kinetics are observed and b = 1.

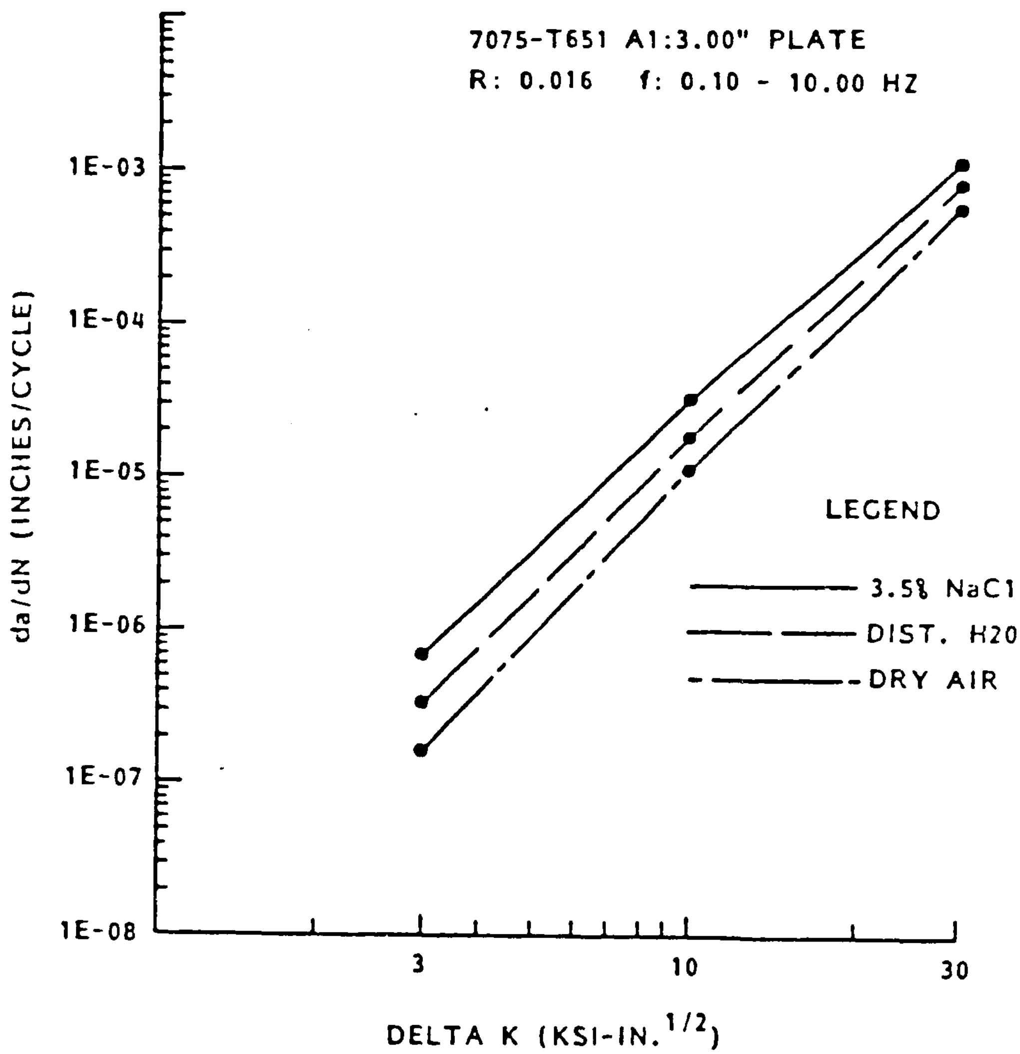


FIGURE 2. da/dN vs DELTA K PLOT FOR 7075-T651 ALUMINUM

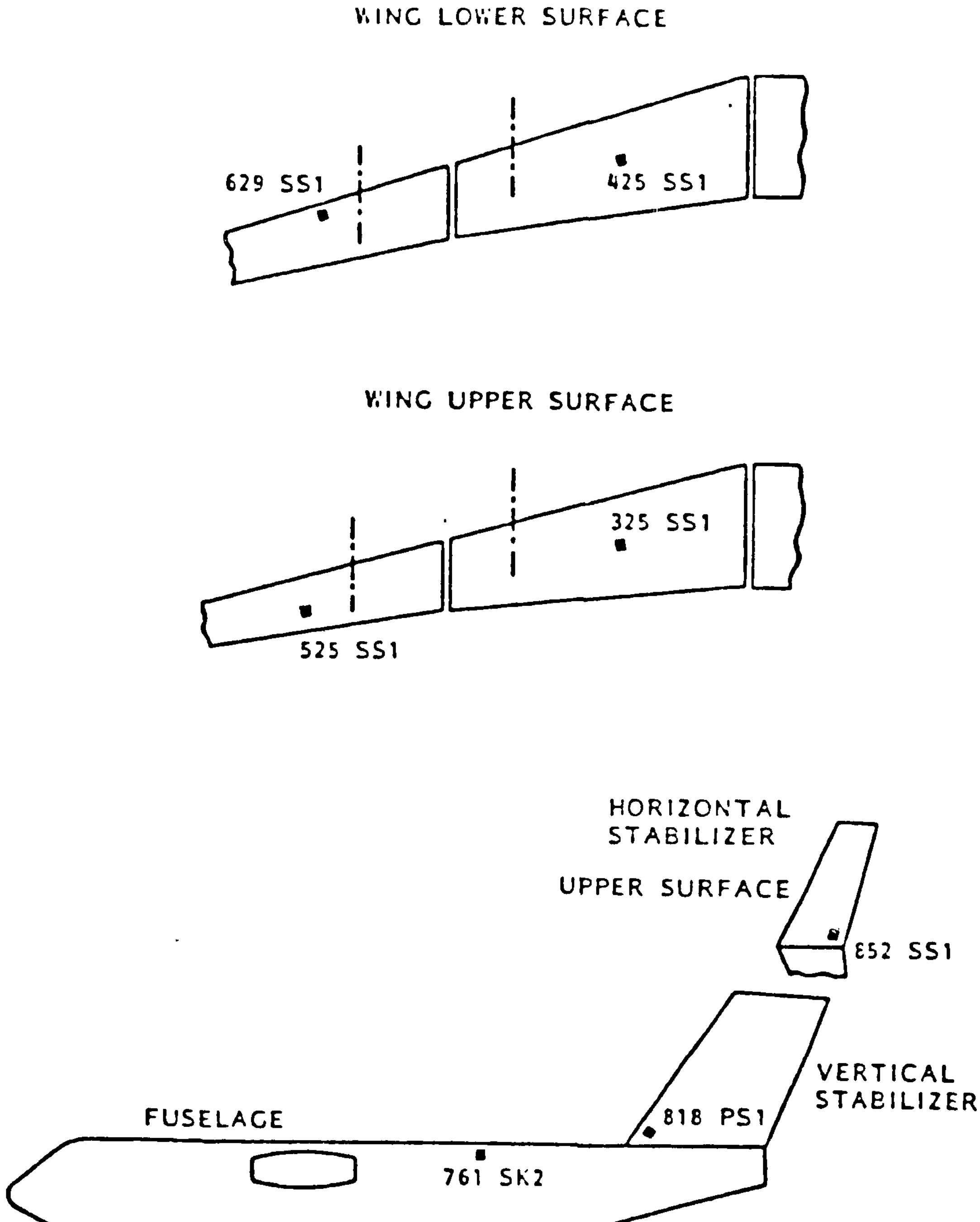


FIGURE 3. LOCATION OF CRACK MONITORING POINTS ON C-5A AIRCRAFT

The literature search revealed only scattered data for the corrosi . of aircraft alloys in the range of environments encountered by aircraft. It was, therefore, necessary to conduct corrosion tests with some of the more widely used alloys in solutions with compositions simulating those of condensate and rainfall encountered by aircraft under service conditions. Using linear regression mathematical techniques, the laboratory test results and the data from the literature were used to determine the constants A and b for specific alloys and environmental conditions.

These constants, summarized in Figures 4a and 4b, are being used in the predictive corrosion computer program. When a specific Air Force base is called out, the program automatically uses the constants and equation which correspond to the environmental conditions at that base.

COATING DEGRADATION

The external surfaces of most Air Force aircraft are completely painted. Except in the case of mechanical damage and initial defects, the time required for fuselage and wing structure to corrode is the coating degradation time plus the corrosion time. In his study of environmental conditions at Air Force bases, Summitt (1) analyzes the factors involved. His basic coating degradation algorithm is presented in Figure 5.

The environmental factors which cause the breakdown of coating systems are ultraviolet radiation, ozone, and sulfur dioxide. By establishing threshold values for the intensity of ultraviolet and for concentrations of ozone and sulfur dioxide, this algorithm enables the rating of bases for their effect on paint systems. An "A" rating represents high values of UV and atmospheric contaminants. A "B" rating represents intermediate values and a "C" rating, low values. This algorithm, combined with Lockheed data on the service life of various coating systems, provides a good basis for determining coating life.

In the predictive corrosion modeling project, the time to initial breakdown of the coating system is more important than the time to completely repaint an aircraft. By the time a paint system has degraded to the point where repainting is desirable, an extensive amount of corrosion damage may have occurred. It is strongly recommended that more emphasis be placed on the touchup and repainting of worn or damaged areas. For this type of paint renewal maintenance, A, B, and C in the algorithm will represent 12, 24, and 36 months. A complete repainting operation should take place at every third paint renewal interval.

COMPUTER PROGRAM

The computer program provides a fully integrated method of predicting crack growth, corrosion damage, or coating degradation for C-5 aircraft in a variety

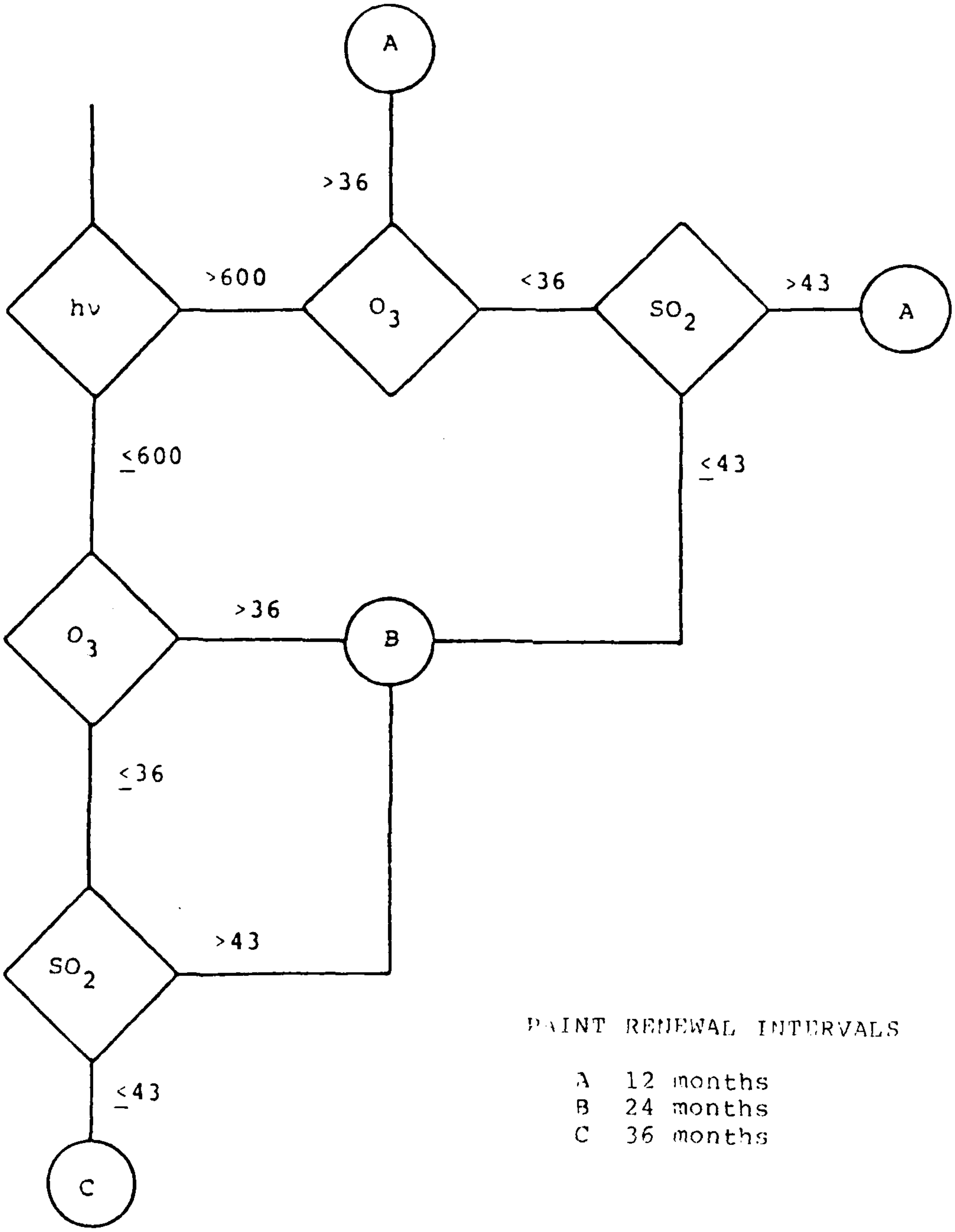
Alloy	Corrosion Index							
	Mild		Moderate		Severe		Very Severe	
	A	B	A	B	A	B	A	B
7075-T6 Al	3.0E-5	.46	2.95E-5	.59	2.9E-5	.72	1.78E-3	.12
2024-T3 (CLAD)	3.6E-6	.70	4.9E-6	.77	6.3E-6	.85	1.48E-5	.70
7079-T6	1.9E-6	.89	2.05E-6	.94	2.2E-6	1.00	5.4E-9	2.00
7075-T73	3.0E-5	.46	3.6E-5	.50	9.0E-4	.50	9.3E-4	.50

FIGURE 4a. ENVIRONMENTAL CONSTANTS FOR CORROSION EQUATIONS

Alloy	Corrosion Index							
	Mild		Moderate		Severe		Very Severe	
	A	B	A	B	A	B	A	B
AZ31B-H24	4.0E-4	.77	2.8E-4	.87	1.6E-4	.97	1.2E-4	1.30
2024-T3	5.0E-4	.30	5.3E-3	.11	1.43E-2	-.05	1.1E-3	.30
4340 Steel	3.5E-11	2.52	2.6E-8	1.40	4.11E-5	1.00	6.9E-5	1.00
300M Steel	3.5E-11	2.52	6.3E-9	2.00	7.3E-5	1.00	5.7E-4	1.00

FIGURE 4b. ENVIRONMENTAL CONSTANTS FOR CORROSION EQUATIONS

Lockheed



PAINT RENEWAL INTERVALS

A 12 months
B 24 months
C 36 months

FIGURE 5. AIRCRAFT REPAINT INTERVAL ALGORITHM (1)
Units for radiation, hν, are Langleys; for ozone and SO2, ug/cu.m.

of environments. The flow diagrams for the VAX-11 FORTRAN computer program are illustrated in Figures 6 and 7. The program contains three modules - the first calculates the remaining effective life of the coating system, the second module predicts the length of any fatigue crack which may be present, and the third calculates the remaining effective life of the paint system. When completed, the program will convert the data obtained into optimum time to next inspection and will select specific scheduled maintenance times for doing the corrosion repair or paint renewal.

In its present form the program is to be used in conjunction with the C-5 and C-141 crack monitoring programs and useage tapes which, for each aircraft, give a record of the bases of operation, the flight dates, flight durations, and the total mission hours.

To run the program we first enter the type of aircraft and the tail number. Next, we specify the type of corrosion being investigated - coating degradation, fatigue corrosion, or general corrosion. If we are interested in checking possible crack growth in a specific part of the aircraft, we specify the location of the crack, the alloy involved, and the theoretical crack length at the last check.

We then input the Air Force bases where the aircraft has been since the last check, and the ground and flight time at each base, and the crack length predicted by the C-5 crack monitoring program. For any given aircraft, the historical information is extracted from the C-5 log tapes.

When the program is run, the output tells us the total crack length, and the number of flight hours remaining until the crack reaches half its critical length.

If the coating degradation module had been used, the output would have told us the remaining time until the next paint renewal or repaint operation.

The general corrosion module output gives the days remaining to corrode exposed metal to a depth of 3 mils. This depth of corrosion damage was selected because it can be readily detected and also easily repaired.

Since each of these modules will give a different time interval until the next optimum maintenance operation, it is now necessary to match the recommended inspection intervals with the maintenance operations already scheduled for the aircraft. This will be accomplished in a final module which will then specify the work task to be done at each of the scheduled inspection and maintenance times.

VALIDATION OF COMPUTER PROGRAM

The validation of the crack growth module is being accomplished by comparing analytical crack growth lengths, adjusted for environmental conditions during the test period, with actual crack growth in the test article used in the C-5A modified wing structural test evaluation (6). During this test a modified wing was subjected to multiple stress spectra representing the loading, takeoff, flight and landing of a C-5A aircraft. This test was initiated in June of 1981 and continued through December of 1982. Approximately 100 cracks were deliberately initiated by making sawcuts in the wing. The growth of the cracks was tracked throughout the test period.

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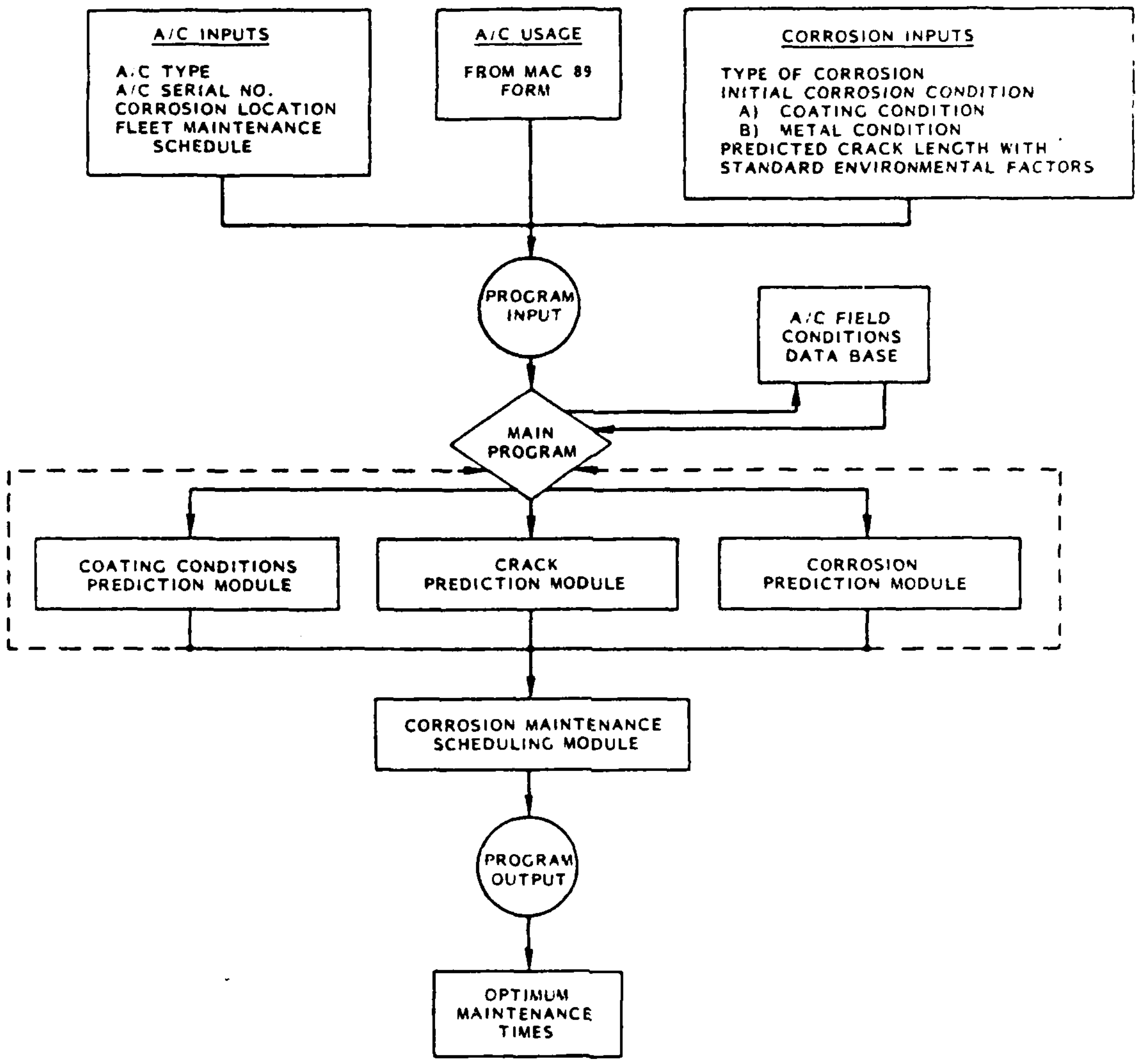

FIGURE 6. FLOW DIAGRAM FOR FORTRAN COMPUTER PROGRAM.

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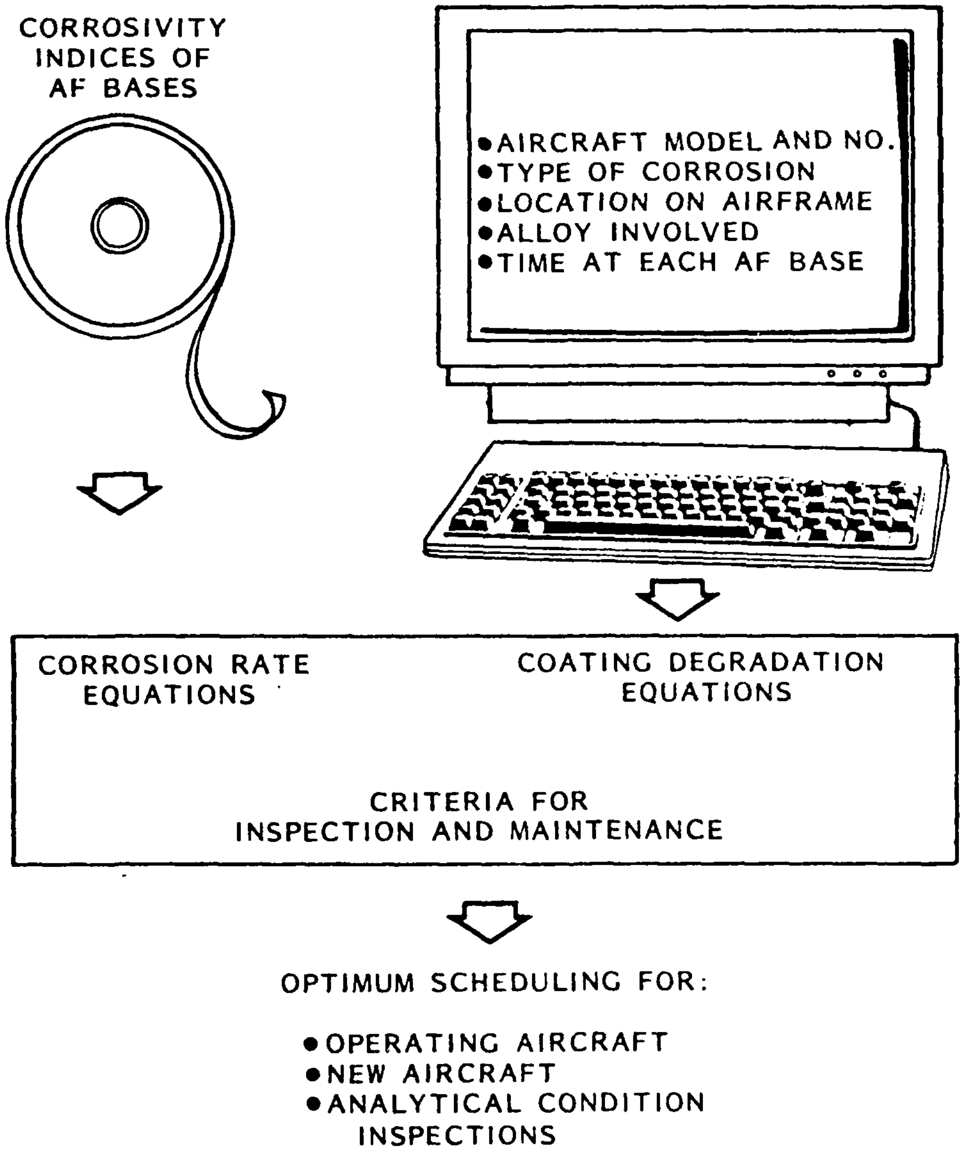

FIGURE 7. FLOW CHART FOR PREDICTIVE CORROSION COMPUTER PROGRAM.

Analytical crack lengths were calculated through the use of the C-5 crack monitoring program which utilizes NASTRAN's finite element analysis methods for computing stresses at selected points on the aircraft structure.

The crack lengths predicted by the C-5 crack monitoring program are based on laboratory fatigue crack growth data obtained at 100% relative humidity conditions. In the predictive corrosion model, this value is corrected for variations in actual humidity and for the presence of salt water in the environment. There was no salt water involved in the C-5 wing test so only the variations in humidity were considered.

Humidity data for the Atlanta area during the test period were obtained from the National Climatic Center at Asheville, North Carolina. Figure 8 lists the average temperature and average humidity for each month during the test period.

Crack growths generated by the predictive corrosion model were compared with actual measured crack growth for the following locations on the modified C-5A test wing:

1. 629-2A

   Flaw 629-2A, illustrated in Figure 9, was initiated in outer wing lower surface panel No. 6. This flaw is in a typical outer wing spanwise splice fastener hole in an area close to the runout of panel No. 6.

   The black squares show the actual growth of the crack during the test period. The upper solid line shows the growth predicted by the C-5 crack monitoring program. This line indicates a more rapid crack growth than actually occurred. However, when C-5 crack monitoring results were corrected for the environmental conditions which existed during the test period, the predicted crack length value almost coincided with the actual values.

2. IWBRS 174

   Flaw 174, illustrated in Figure 10, was initiated in inner wing lower surface panel No. 5. It was a corner flaw in an open drain hole. In this instance the crack lengths predicted by the C-5 crack monitoring program tracked the actual crack lengths very closely. When corrected for humidity conditions, the predicted were slightly less than the actual values.

3. 432S

   The next flaw, illustrated in Figure 11, was initiated in inner wing lower surface panel No. 1. It was cut in a typical spanwise splice fastener location in the splice between Panels No. 1 and 2.

   The plot of predicted crack lengths vs. actual crack lengths shows the uncorrected predictions to be higher than the actual growths. The points for the corrected lengths agree more closely with the actual lengths.

FIGURE 8.

AVERAGE HUMIDITIES AND TEMPERATURES DURING C-5 WING TEST

YEAR	MONTH	AVERAGE TEMPERATURE (degrees F)	AVERAGE HUMIDITY
1981	June	81.3	65%
	July	82.2	68%
	August	77.7	75%
	September	72.4	71%
	October	60.2	73%
	November	54.1	64%
	December	39.1	70%
1982	January	38.5	72%
	February	47.4	70%
	March	56.5	63%
	April	58.4	61%
	May	72.5	62%
	June	76.3	67%
	July	79.1	76%
	August	77.5	76%
	September	70.5	72%
	October	62.7	71%
	November	53.7	71%
	December	49.9	76%

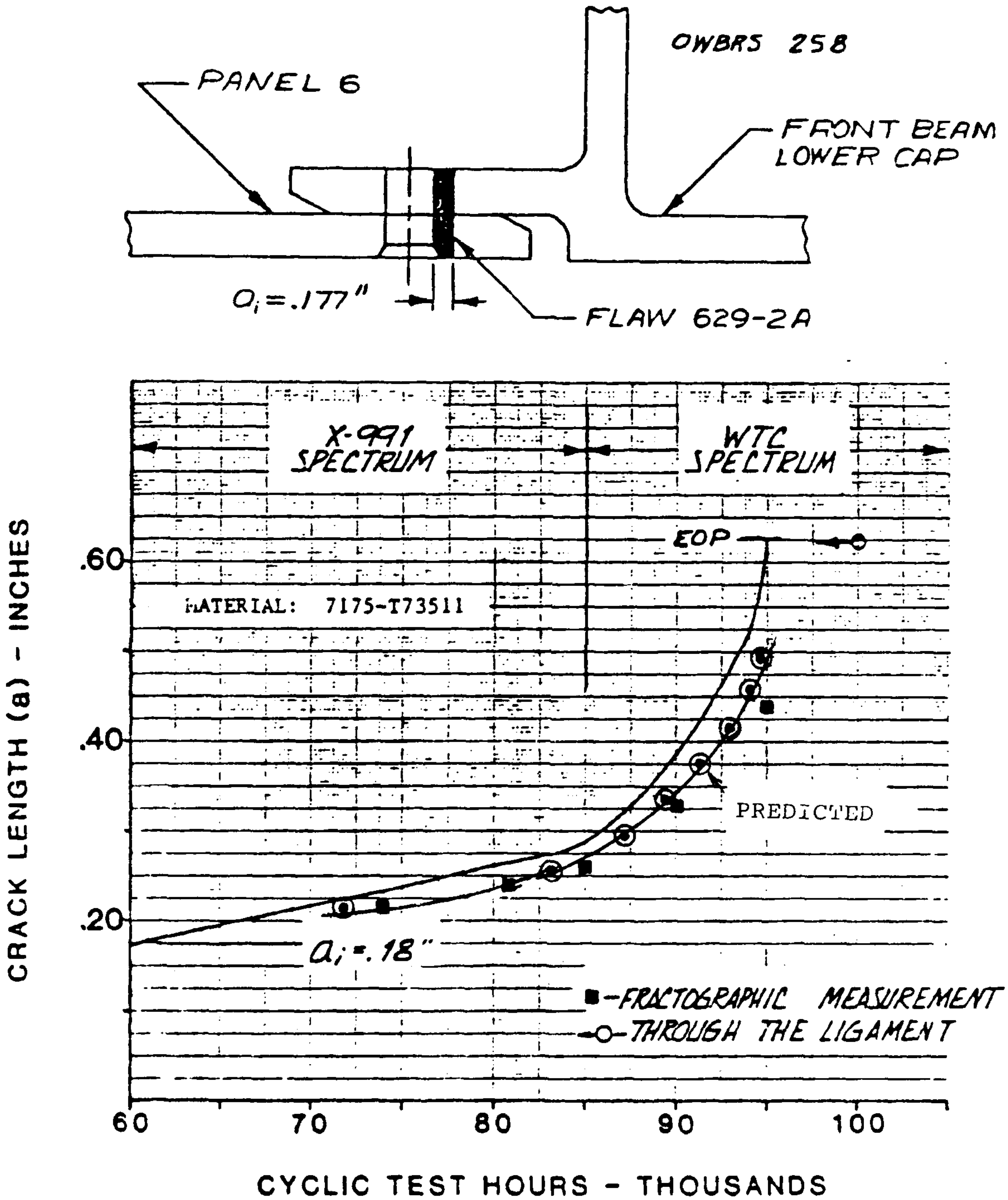

FIGURE 9. COMPARISON OF ANALYTICAL AND TEST DEMONSTRATED CRACK-GROWTH AT FLAW 629-2A.

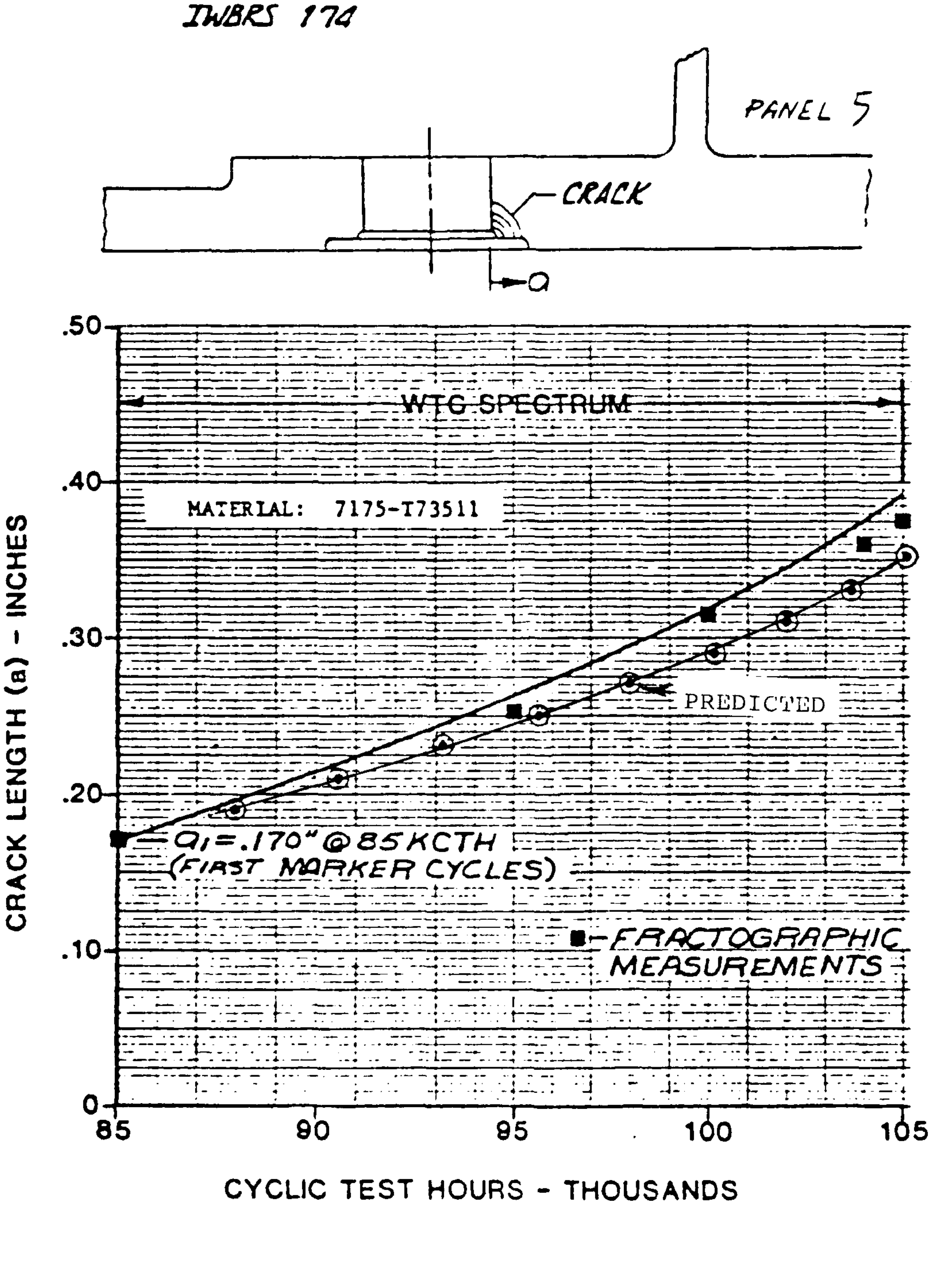

FIGURE 10. CRACKGROWTH COMPARISON - DRAIN HOLE.

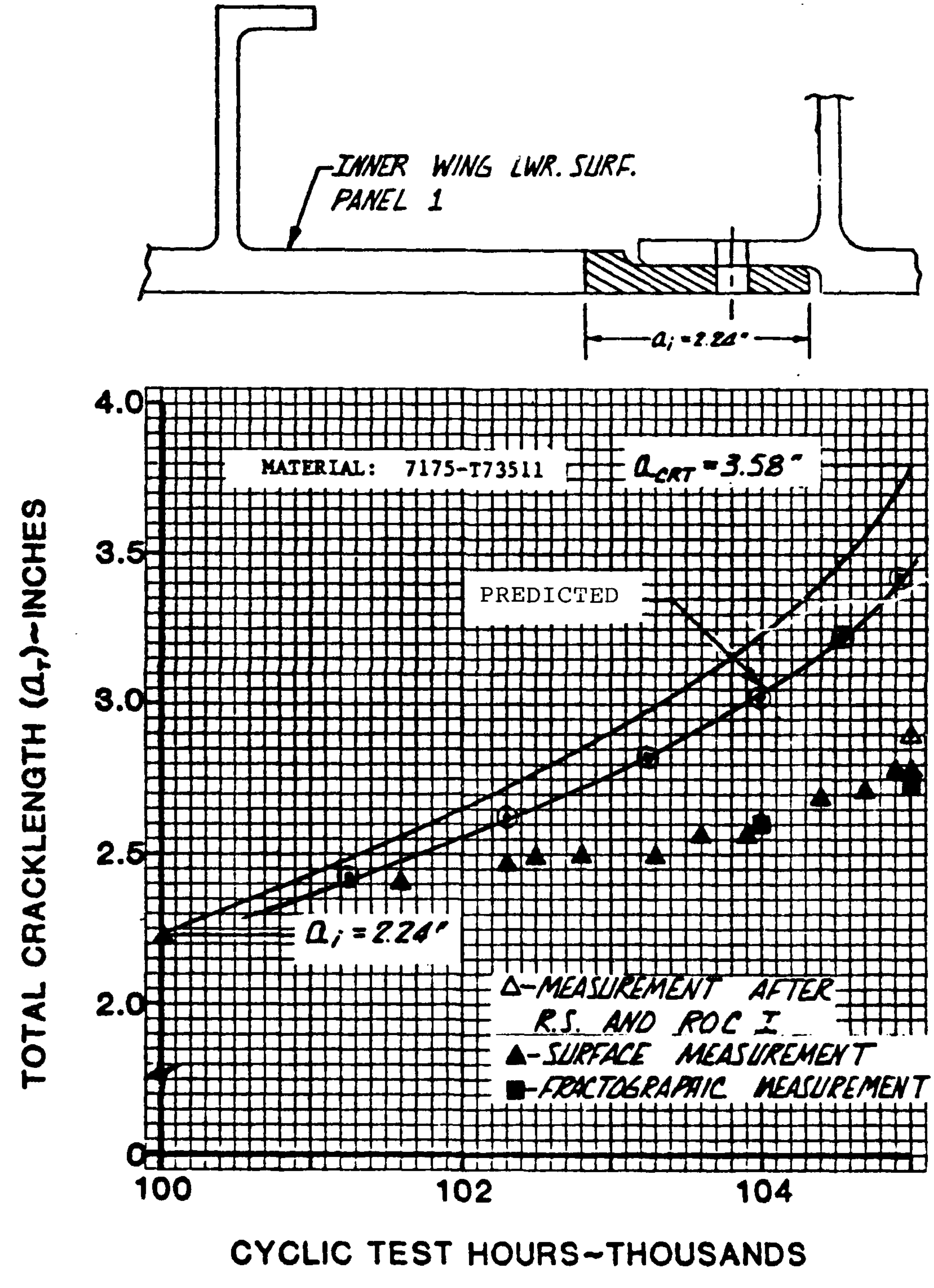

FIGURE 11. ANALYTICAL VS. TEST DEMONSTRATED CRACKGROWTH AT FLAW 432SS

4. 425CP

Figure 12 shows the location of Flaw 425CP. This was in inner wing lower surface panel No. 5 in the area where the panel is attached to the mid beam lower cap. Again, the solid line representing the C-5 crack monitoring predictions show greater crack lengths than actually occurred. The corrected values, indicated by the circles, are closer to the actual values.

5. 615SS

The location of Flaw 615SS was in the outer wing lower surface panel No. 4. As illustrated in Figure 13, the flaw was initiated at a spanwise splice fastener hole. The predicted crack lengths, corrected for environmental conditions, are very close to the actual crack lengths but on the conservative side.

DISCUSSION

In summary, a predictive corrosion modeling program has been developed which will give optimum inspection and maintenance scheduling for the major types of crack growth, corrosion damage, and coating degradation problems which occur on C-5 aircraft. Specific aircraft can be quickly checked for potential crack growth in critical areas, for probable corrosion damage to exposed structural alloys, or for the condition of the aircraft coating system.

The results of the crack growth validation demonstrate that the predictive corrosion modeling program, used in conjunction with the C-5 crack monitoring program, will enable more accurate predictions of fatigue crack growth than can be obtained from the C-5 crack monitoring program alone.

The validation of the general corrosion predictions is in progress. Corrosion predictions for selected areas on the C-5 wing and fuselage are being compared with plots of Corrosion Control Manhours vs. Time in Months for those locations. It is anticipated that there will be a significant increase in manhours when corrosion of exposed metal to a depth of 3 or more mils occurs.

The predictions of the paint degradation module will be compared with the paint touch up and repaint histories of selected C-5 aircraft.

The final task will be to determine the feasibility of modifying current maintenance activity control systems to include the predictive corrosion modeling program. The impact of making these changes will be estimated from the standpoint of cost of incorporation and cost increases or decreases of maintenance activity, as well as improvement in operational readiness of the aircraft.

CONCLUSIONS

1. A VAX-11 computer program, which will predict corrosion damage and fatigue cracking of aircraft alloys and the degradation of aircraft coating systems in a variety of environments, has been developed.

2. The predictive corrosion modeling program will enable optimum inspection and maintenance scheduling for the major types of crack growth, corrosion

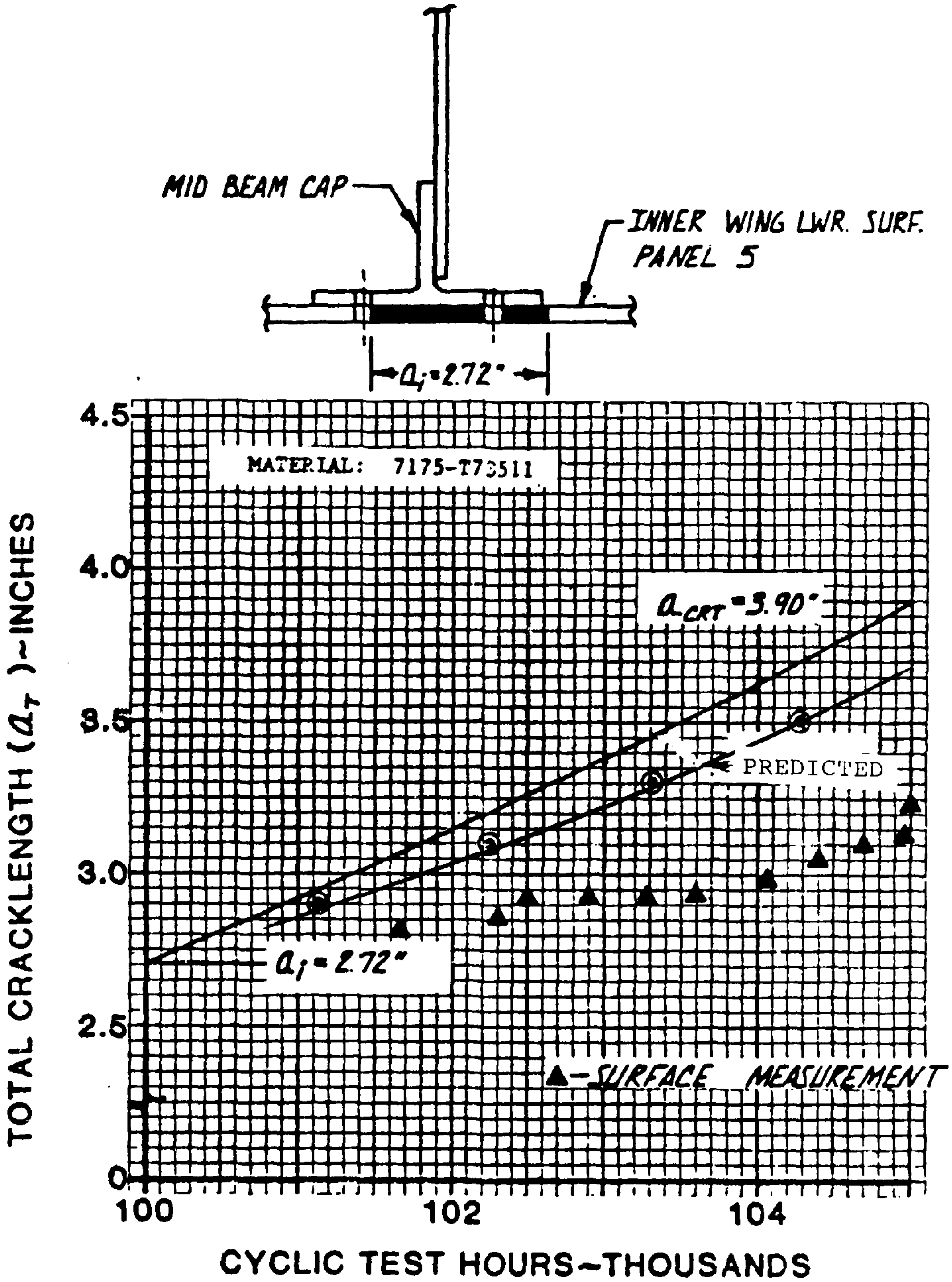

FIGURE 12. ANALYTICAL VS. TEST DEMONSTRATED CRACKGROWTH AT FLAW 425CP

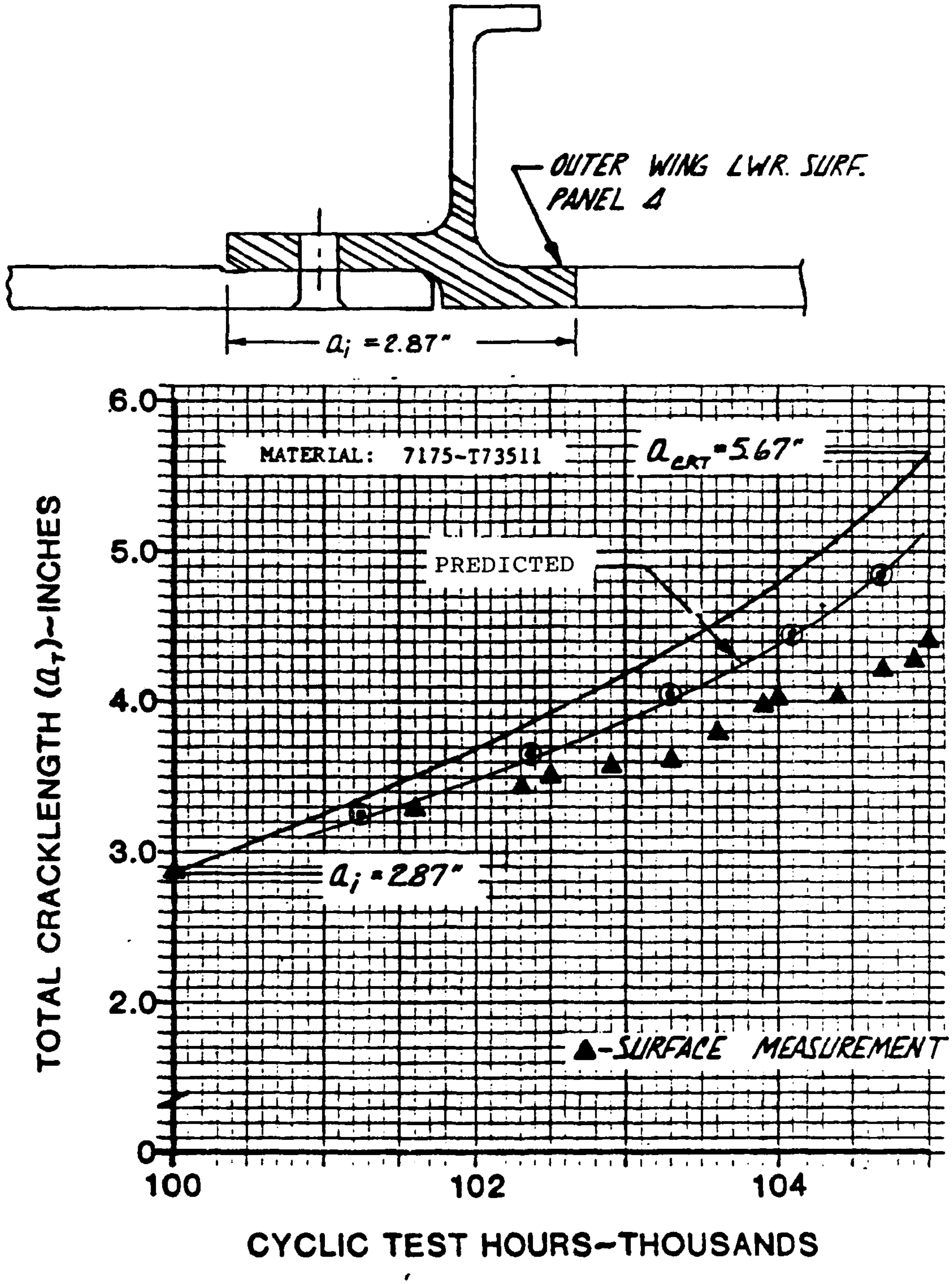

FIGURE 13. ANALYTICAL VS. TEST DEMONSTRATED CRACKGROWTH AT FLAW 615SS.

damage, and coating degradation problems which may occur on C-5A aircraft.

3. The program can be readily modified for use on the C-141, C-130, B-52 and other aircraft fleets which have crack monitoring programs.

4. The implementation and use of the predictive corrosion modeling program will minimize unnecessary inspections and will enable corrosion damage to be prevented or repaired at minimum cost.

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AN ALUMINUM QUALITY BREAKTHROUGH FOR AIRCRAFT STRUCTURAL RELIABILITY

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ABSTRACT

In 1983 Alcoa's Davenport Works initiated a statistically designed experiment to evaluate effects of metal processing on thick plate metal quality. An outgrowth of this program has been a breakthrough in quality and resultant property improvements that can be exploited for fatigue and fracture critical structures. This paper describes the statistical quality control effort, and gives evidence of the improved capabilities typical of recently produced high quality material.

Among conventional mechanical property tests, the smooth fatigue test is shown to be the most discriminating for initial metal quality. Data are shown correlating longer lifetimes to reduced microporosity size in the improved plate. Replicate fatigue tests enable definition of a "characteristic" initial flaw size distribution which can serve as a starting point for flaw growth analysis and life management. These findings are discussed relative to initial fatigue quality guarantees and compatibility with emerging U.S. Air Force durability analysis methodology. In summary, it is shown that the combination of more discriminating testing and a superior product offers considerable promise for reliability improvement in aircraft structural designs of the future.

Key Words: aluminum; statistical quality control; quality assurance testing; nondestructive inspection; mechanical properties; fatigue; fracture; aerospace; structural integrity; reliability; maintainability; life management

INTRODUCTION

Future aircraft designs will reflect more stringent reliability demands to contain mounting costs associated with maintenance and downtime (1-5). A survey of United States Air Force (USAF) logistic and maintenance centers revealed that most structural durability problems are the result of cracking (6). Consequently during design, assurance is sought that the structure will not crack excessively in service leading to functional impairment affecting the aircraft readiness (2-5,7-11). Durability design begins with quality of the starting material. Under fatigue loading, for example, surface scratches, inclusions or micropores can greatly accelerate the crack initiation process, and though cracks emanating from these origins are not an immediate safety hazard, they affect structural maintenance requirements.

Al-Zn-Cu-Mg alloy 7050 was developed by Alcoa to provide a superior combination of strength, stress corrosion resistance and fracture toughness, particularly in thick plate sections. Since the mid-1970s Alcoa has supplied millions of pounds of thick 7050-T7451 plate for fatigue and fracture critical aircraft structures, such as fuselage bulkheads and wing box applications. In 1983, Alcoa's Davenport Works implemented a statistical process control experiment with the objective of improving quality and engineering characteristics of 3 to 6-inch thick 7050-T7451 plate. While the current product was capable of meeting existing aircraft material specifications, improvements were sought to satisfy the higher integrity needs of future applications.

In carrying out this investigation, smooth coupon fatigue testing coupled with post-test fractography was used to characterize members of the microflaw population with the greatest likelihood of originating detectable cracks in service. This promising technique is appropriately sensitive and well suited to the needs of emerging USAF guidelines for reducing cracking problems in metallic aircraft structures (3-5,8,11,12).

Though the effort described in this paper focuses on aluminum alloy 7050 thick plate, the statistical quality control and fatigue test methodologies utilized are transportable to other high strength aluminum alloy systems. As a result, the 7050 alloy quality improvements demonstrated herein have also benefited other 2XXX and 7XXX aluminum plate alloy systems.

STATISTICAL QUALITY CONTROL EXPERIMENT

While not new conceptually (13-15), statistical quality control tools have not been incorporated extensively in problem solving methodology at the plant level. Once the commitment to improving quality had been established, the first task was to determine where to start. Process improvement opportunities on the shop floor are seemingly unlimited, so five process variables considered to be most critical were selected for initial investigation using a statistically designed experiment.(*)

A statistically designed experiment provides the means of separating the "vital few" process variables from the "trivial many" (16). For this experiment, a 2^{5-1} fractional factorial (five variable, two-level) design was selected, Figure 1. This design assures that all main effects and two-factor interactions are clear and not confounded with other main effects and two-factor interactions (17). A total of twenty runs were performed for the experiment--sixteen for the design and an additional four replicated at the midpoints to provide a measure of experimental error. About 300,000 pounds of metal was produced for the twenty run experiment. The experiment proved successful in that it identified the most significant process variables.

Following completion of the designed experiment, efforts were directed at bringing the variables into a state of statistical control. Figure 2 is a composite Shewhart run chart (15) showing the relative values of a significant process variable before and after incorporating the quality control effort. As shown, the mean level ($\bar{x}$) was lowered by 60 percent, and the variation as defined by the upper and lower control limits (UCL and LCL, respectively) was reduced 3.4 times. This activity was repeated many times over, and as quality improvements were realized, additional process variables were targeted for statistical control.

(*)Statistical quality control efforts focus on the current process, making improvements through reduction in variation. The focus of this paper is on the measurable improvements resulting from this effort, and mention of specific process variables is avoided in the interest of maintaining the proprietary advantage this commitment has achieved.

QUALITY IMPROVEMENTS

Process improvements are directly reflected by improvements in the product (15). While some improvements were realized following the breakthrough achieved from the initial effort, quality improved progressively over time as new variables were brought under control. As a result, some quality and engineering characteristic improvements were immediately apparent, while others became evident only after additional process improvements were made.

Ultrasonic Inspection

Ultrasonic indications in thick plate correlate to the degree of microporosity present. One of the first benefits realized from the quality improvement effort was the elimination of Class B (18) indications.(*)

In May of 1985 this improvement led to a new guarantee that Alcoa would no longer furnish plate with Class B indications, and that all plate for aerospace applications would meet Class A inspection limits. Even with the acceptance of the tighter Class A limits, ultrasonic inspection recovery continued to improve, exceeding 99 percent for the past three years. While ultrasonic inspection does not present a total picture of the micropore distribution it does portray the "worst cases" comprising the distribution tail. The quality improvement as determined through the Class frequency present in the distribution tails is shown graphically in Figure 3. In 1981, the average 7050-T7451 plate lot sampled contained 0.8 Class A indications and 4.8 Class AA indications. By 1985, the distribution tail had shifted to the point that the average lot sampled contained no Class A indications and only 0.02 Class AA indications. As such, it is evident that 7050-T7451 plate Class AA inspection can now be guaranteed upon request. More recently, Class AA capability has been demonstrated for all Alcoa plate alloys with controlled fracture toughness requirements for aerospace.

(*)Ultrasonic classes for single discontinuity response:

Class	Response	
AA	3/64 in.	Any discontinuity with an indication greater than the
A	5/64 in.	response from a reference flat-bottom hole of the size
B	8/64 in.	given (diameter) is not acceptable.

Reduction in the degree of centerline microporosity with time has been verified by an internal quality check. Since 1976, Alcoa has performed a separate ultrasonic scan for aerospace alloys in thick gauges (>3 inches), where rolling may be insufficient to heal microporosity. The scan consists of inspection for a continuous response at the mid-plane (location of greatest microporosity) using a 3/4-inch diameter 5 MHZ crystal set to 100 percent response of a Class A test block. An internal response limit of 15 percent was selected as a means of ensuring plate integrity. (This limit is much tighter than that of MIL-STD-2154 (18), which states that loss of back reflection exceeding 50 percent shall be cause for rejection.) Following the quality control efforts, Alcoa's internal inspection limit was lowered to 2 percent (threshold for detection) to reflect the improved product integrity.

Fluorescent Penetrant Inspection

The reduction of plate centerline microporosity was also verified by fluorescent penetrant response. Prior to incorporating the quality effort, banding of centerline penetrant indications was generally noted in thick plate. For 6-inch thick plate, the band would be centered on the mid-thickness (T/2) plane and approximately 1-inch wide. Following the quality control effort, the band was no longer evident, and the indications present were distributed throughout the thickness.

Mechanical Properties

The first production runs of 7050-T7451 thick plate, following quality control improvements implemented in 1984, showed a dramatic shift in both the mean and variation of short transverse elongation values. The mean increased from the historic level of 3.8 percent to 4.8 percent--a 26 percent increase, while the lower 95-99 percent limit increased from 2 percent to 3 percent--a 50 percent increase. Over the same time span, tensile and yield minimum strength values increased by 1 to 4 ksi, and minimum fracture toughness values increased 1 to 5 ksi $\sqrt{in}$. These improvements have been documented (19) and are being incorporated as revisions to AMS-4050D (20), MIL-HDBK-5 (21), as well as to customer specifications.

FATIGUE

Most early life fatigue failures originate from preexisting defects. Preservice inspection attempts to reduce both the magnitude and severity of these flaws. However, traditional nondestructive methods are suspect for the more demanding microdefect screening required by new aircraft and engine structural durability guidelines (3-5,8,11,12). Over the years much effort has been devoted to characterizing fatigue behavior using statistics of extreme values (22-25). A rationale behind these studies is that materials contain weakening flaws, and though the flaw size spectrum may be wide, the fatigue process seeks out the dominant flaw (weakest link). Therefore, controlling the distribution of all flaws is not as important as controlling size of the largest flaws (the extreme values). The smooth axial fatigue test is appropriately sensitive for quality screening because the failure process seeks out the weakest microstructural feature (26). The test is also simple and relatively inexpensive, thereby making it attractive for use in a production environment.

Figure 4 shows that smooth fatigue lifetimes of 7050 plate increase progressively with thickness reduction. Added rolling to thinner plate gages facilitates healing of preexisting micropores. Fractures associated with the 7050 plate data revealed that failure for the heavier plate gages, in all cases, originated from a micropore located at or just beneath the specimen surface; e.g., Figure 5. Consequently, the fatigue quality screening focused on samples from the T/2 mid-thickness location where micropore concentration is greatest in heavy gage plate (5.0-5.9 inch). Fatigue specimens were oriented in the long transverse test direction so that loading would be normal to the elongated direction of the micropores. After some preliminary testing, a 3.5 ksi minimum to 35 ksi maximum cyclic stress range was selected to produce failures in a reasonable time. Early in the investigation, broken specimens were examined in an attempt to correlate fatigue life and size of the microvoid at the failure origin. Figure 6 shows data established for commercial plate lots fabricated from 1984 to 1985. The micropore size corresponding to the plotted data is the maximum pore dimension measured from an SEM photograph of the specimen fracture (dimension "a" of Figure 5, for example). As expected, longer fatigue lifetimes tend to be associated with the smaller micropore origins, and Figures 6 and 7

show that 1985 process improvements successfully diminished occurrence of larger micropores responsible for early failures in the 1984 material. Fatigue testing is now a part of Alcoa's process monitoring strategy for thick 7050 plate, and the cumulative experience given in Figure 8 shows the quality improvement since implementation of this practice. Quality has improved to the point that the majority of specimens tested from 1986 to date survive an arbitrary 160 kilocycle truncation imposed to shorten test times for production material lot release.

Fatigue crack growth tests in accordance with ASTM E647 (27) were also conducted to determine the effect of microporosity degree on crack propagation behavior. Comparable growth rates (da/dN) were obtained from specimens removed at the T/2 location of various thickness 7050 plates, Figure 9, and from material at both the T/2 (high microporosity) and T/4 (low microporosity) locations of thick 7050 plate, Figure 10. It is concluded from these results that crack propagation rates are insensitive to microporosity degree when the size of the crack is much larger than the scale of the microstructure. In contrast, the preceding smooth specimen results of Figures 4, 6 and 8 imply that microporosity degree has significant influence on crack nucleation and early stage growth.

QUALITY IMPLICATIONS ON STRUCTURAL RELIABILITY AND LIFE MANAGEMENT

Total cost of ownership is becoming more important in selection and qualification decisions on aircraft materials and manufacturing processes (1). When averaged over the life of a part, structure or entire fleet, maintenance and downtime costs can become a driving force for change. Consequently, design and diagnostic life management strategies are needed to ensure longevity and safety without incurring excessive cracking problems over the design life period. Life assurance begins with controls on manufacturing, since life and consistency of performance are quality dependent. The conceptual drawing of Figure 11 illustrates that lifetime to grow a crack to size "a" can, on average, be extended and be reproduced more consistently by decreasing size of the largest preexisting flaws.

Aircraft structural durability requirements are concerned with reducing the probability of relatively small (0.0005-0.05 inch) flaws (of whatever origin)

growing to sizes resulting in functional impairment and high life-cycle costs. Analytical procedures for predicting fatigue crack exceedence probabilities as a function of time in service have been recently developed by the USAF (8) and verified on full scale structures (4,5). These procedures employ probabilistic fracture mechanics and correlate structural cracking to initial fatigue quality represented as an equivalent initial flaw size (EIFS) distribution (3-5,8). An equivalent initial flaw is a hypothetical crack assumed to exist prior to service. The EIFS distribution can be back calculated from smooth coupon specimen fatigue lives and the appropriate crack growth rate data as conceptually illustrated in Figure 12. An EIFS versus fatigue life curve calculated in this manner for the test conditions of Figure 6 is shown to fit the actual data reasonably well. The Figure 6 computation assumed a semi-elliptical surface crack of depth a and length 2c with stress intensity factor given by the solution of Raju and Newman (28). An aspect ratio (a/c) of 0.8 was chosen since it approximates the equilibrium shape partial-thickness crack of a uniformly loaded round tensile bar (28,29). It has been observed repeatedly in the literature (30-33) that small cracks grow faster than rates predicted by near-threshold data obtained from long crack specimens of the current standard ASTM practice (27). For simplicity and to compensate for the small crack effect, the EIFS curve in Figure 6 was calculated using 7050 growth rate data (R=0.1) corrected by linear extrapolation to low da/dN as illustrated in Figure 12. Refinements to further improve the computational accuracy of the EIFS model are presently being evaluated. The concepts incorporated into these enhancements are described elsewhere (33-35) and are outside the scope of this discussion.

The main point to be emphasized by the preceding example is that fracture mechanics interpretation of smooth fatigue data enables quantification of initial (weakest link) microdefect sizes in a manner that is totally consistent with new USAF durability analysis methodology (3-5,8). Once determined, the EIFS distribution can be viewed as an initial quality characteristic of the material or the manufacturing process. In contrast, life is dependent on a number of factors including load history, geometric details, and the material strength and toughness properties. Hence, the EIFS distribution can serve as a starting point for incorporating quality into computational design and/or

diagnostic tradeoffs based on required lifetimes, for a component, aircraft or entire fleet, Figure 13.

SUMMARY

Structural reliability and maintainability are becoming important tradeoff considerations in design of advanced metallic aircraft. To meet future needs, improved materials and quality assurances will be necessary to avoid excessive costs of maintenance associated with cracking problems in the field. Conventional lot release testing of mechanical properties is not sufficiently discriminating of initial metal quality in relation to reliability performance objectives. Smooth fatigue testing exhibits a level of discrimination to quantify material reliability in terms useful to design. The coupon test lifetime distribution can then be transposed to an equivalent initial flaw size distribution as a starting point for flaw growth analysis and life management. Thus in addition to use for warranty of metal quality and consistency on a lot-by-lot basis, the smooth fatigue test gives data enabling reliability assessment of well designed parts.

A quality breakthrough made on thick 7050-T7451 aluminum alloy plate was demonstrated with respect to various reliability criteria established at the material producer level. The statistical quality control methods adopted on a plant-wide basis at Alcoa's Davenport Works resulted in significant improvements in conventional quality indices and smooth fatigue specimen test results. The demonstrated combination of more discriminating testing and a superior quality product offers promising new options for incorporating reliability into aircraft structural designs of the future.

ACKNOWLEDGMENTS

This work represents over five years of accumulated effort by many Alcoa individuals, and any list of names is likely to miss valued contributors. In reality, it is the commitment to excellence of the organization for which they all work. Particular acknowledgment is owed to **R. W. Westerlund**, **K. P. Young**, **D. F. Skluzak**, **M. A. Green**, **D. W. Barber** and **D. S. Shryack** of Alcoa Davenport Works for technical support on various aspects of the plant experiments. To **G. Sowinski, Jr.**, **R. L. Brazill** and **W. T. Kaiser**, Alcoa Laboratories, for assistance in the fatigue experimentation and data analysis, and to **P. L. Mehr**, Aerospace Applications Engineer, for technical consultation.

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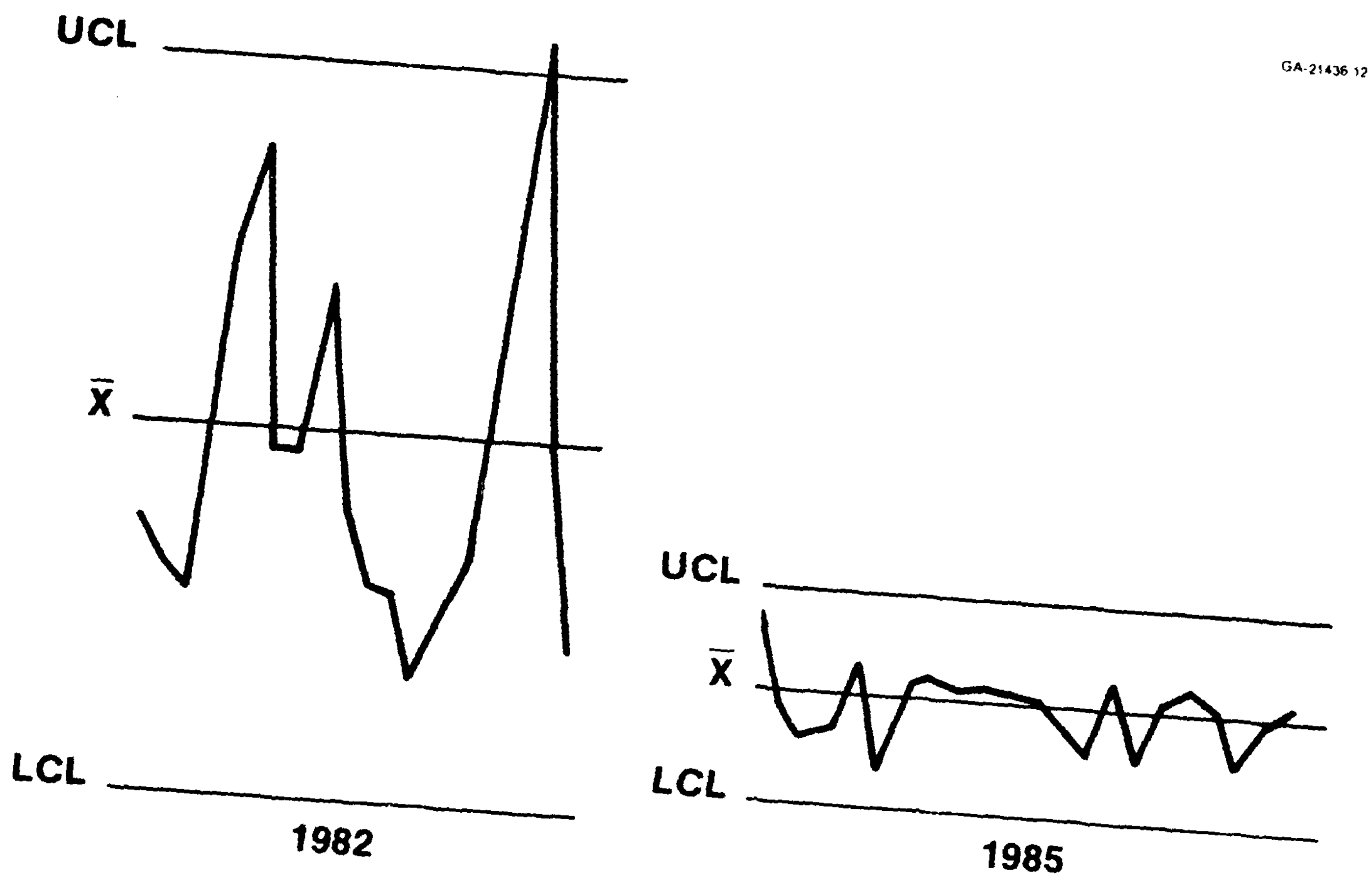
GA 21436.13

Design order	1	2	3	4	5
1	+	–	–	–	–
2	–	+	–	–	–
3	–	–	+	–	–
4	+	+	+	–	–
5	–	–	–	+	–
6	+	+	–	+	–
7	+	–	+	+	–
8	–	+	+	+	–
9	–	–	–	–	+
10	+	+	–	–	+
11	+	–	+	–	+
12	–	+	+	–	+
13	+	–	–	+	+
14	–	+	–	+	+
15	–	–	+	+	+
16	+	+	+	+	+
Replicates					
17	MP*	MP	MP	–	MP
18	MP	MP	MP	+	MP
19	MP	MP	MP	–	MP
20	MP	MP	MP	+	MP

*MP = Midpoint in range

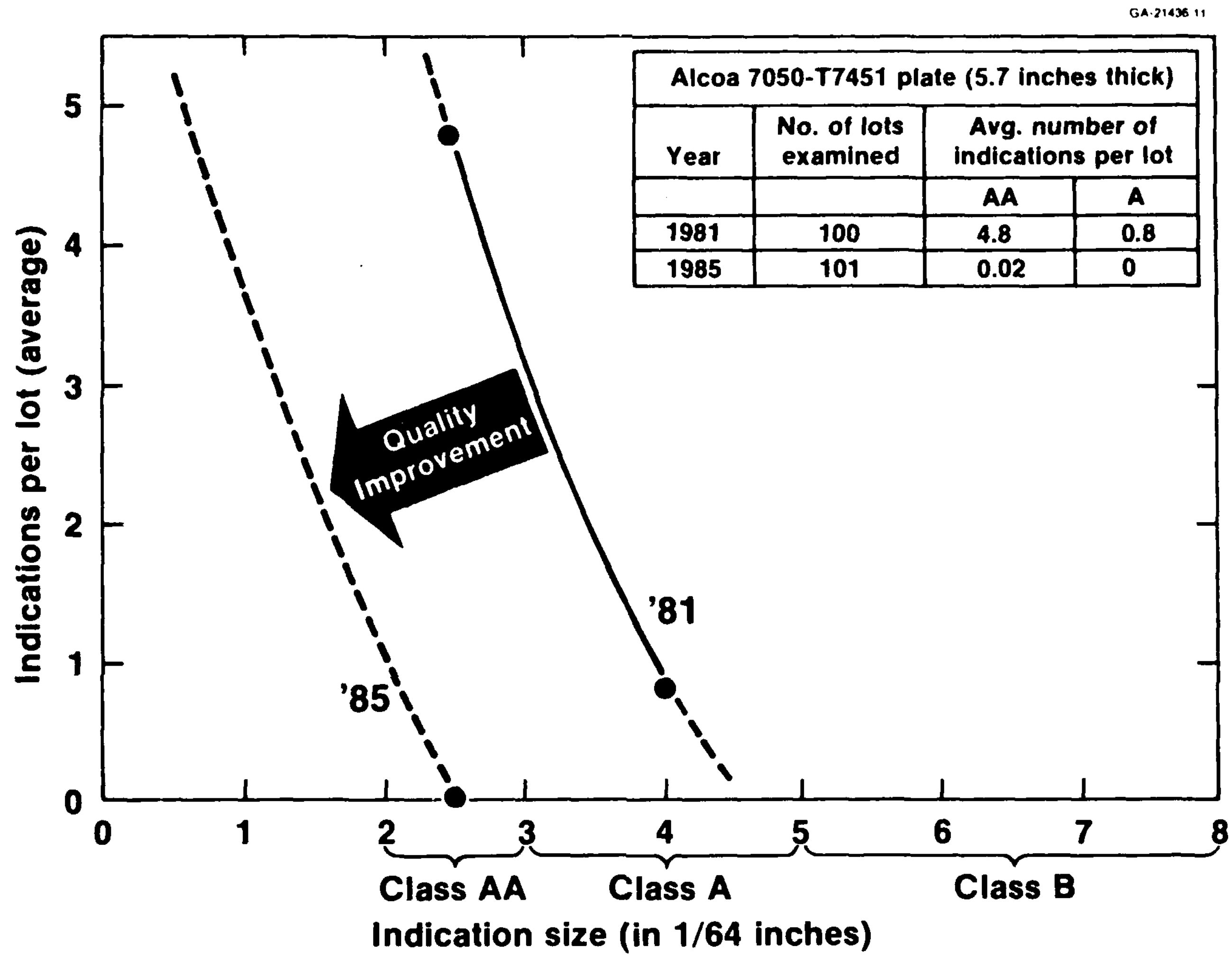
7050-T7451 Plate Improvement 2^{5-1} Fractional Factorial Design with Resolution V

Figure 1



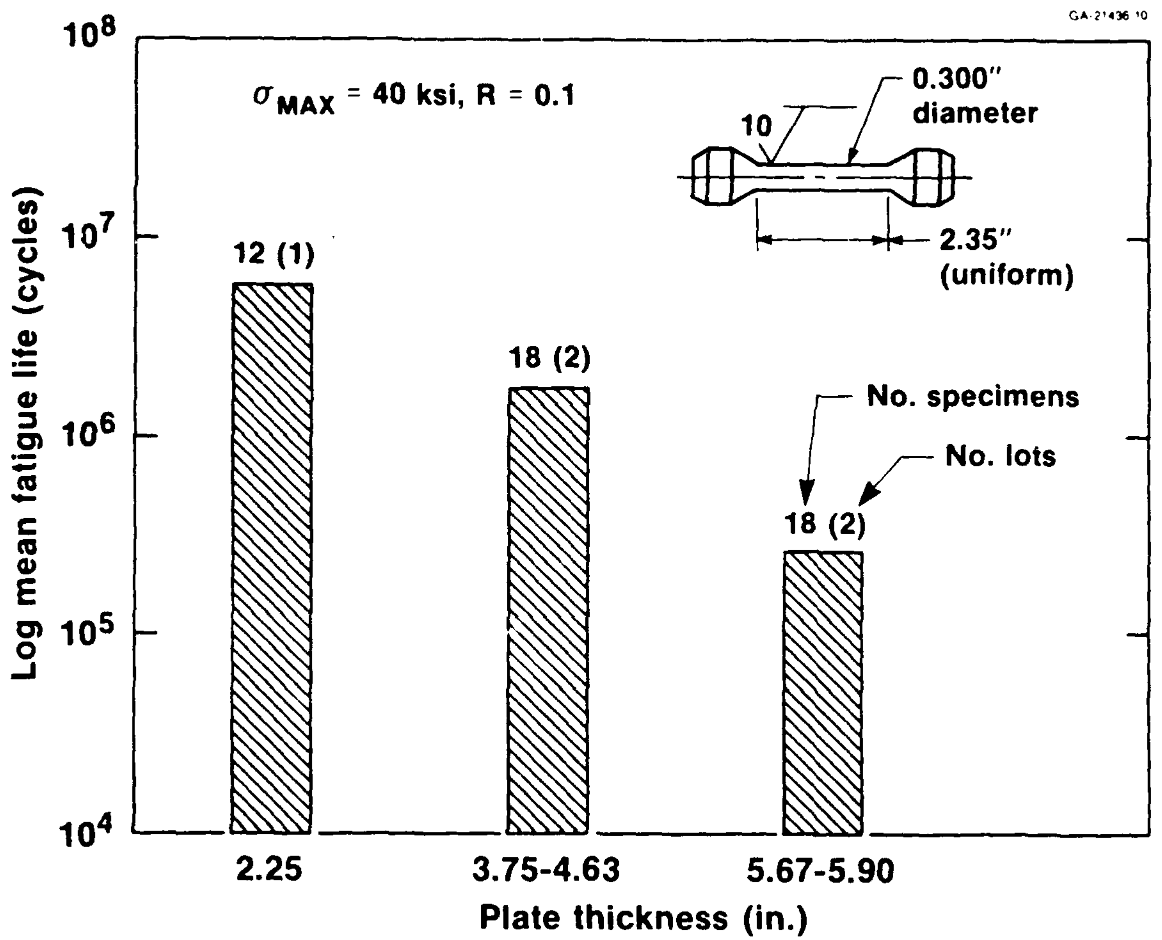
Run Chart of the Most Significant 7050-T7451 Process Variable

Figure 2



Alcoa 7050-T7451 plate (5.7 inches thick)			
Year	No. of lots examined	Avg. number of indications per lot	
		AA	A
1981	100	4.8	0.8
1985	101	0.02	0

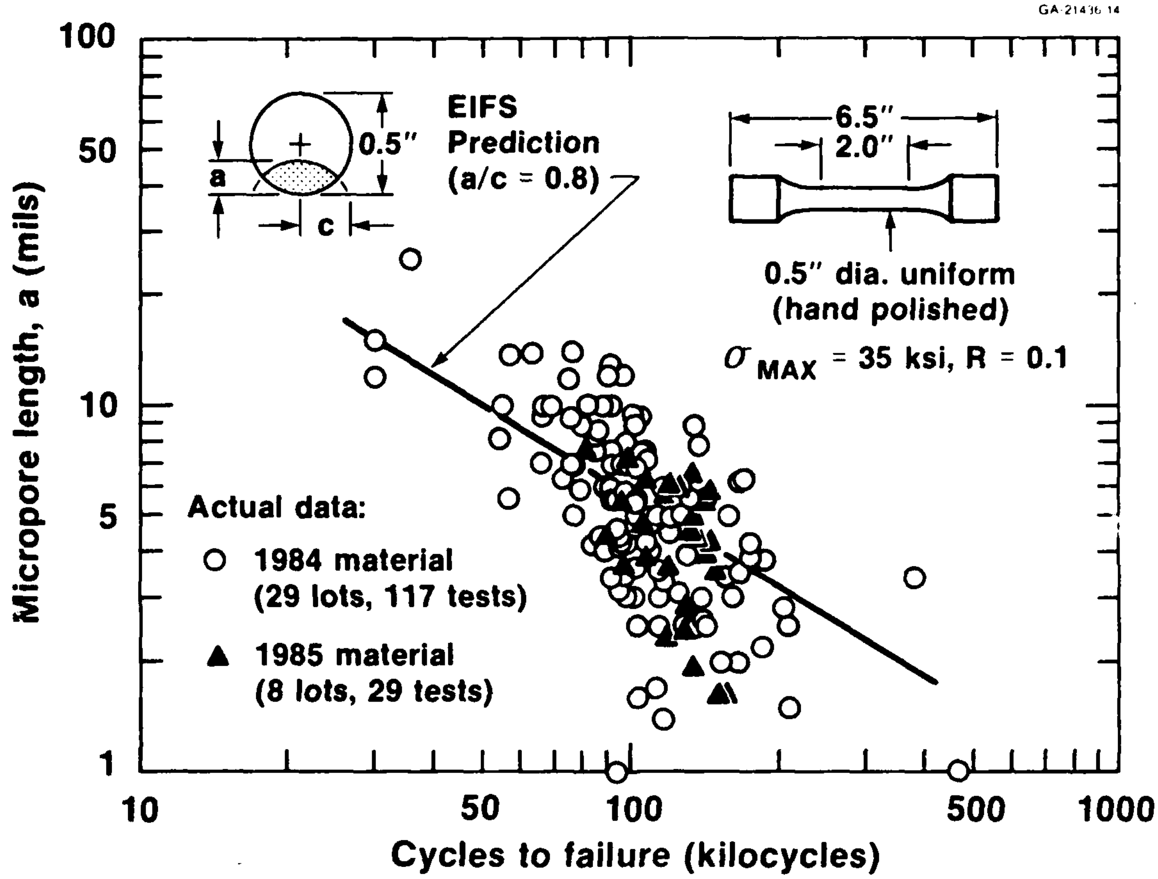
Thick 7050-T7451 Plate
Ultrasonic Quality Improvement
Figure 3



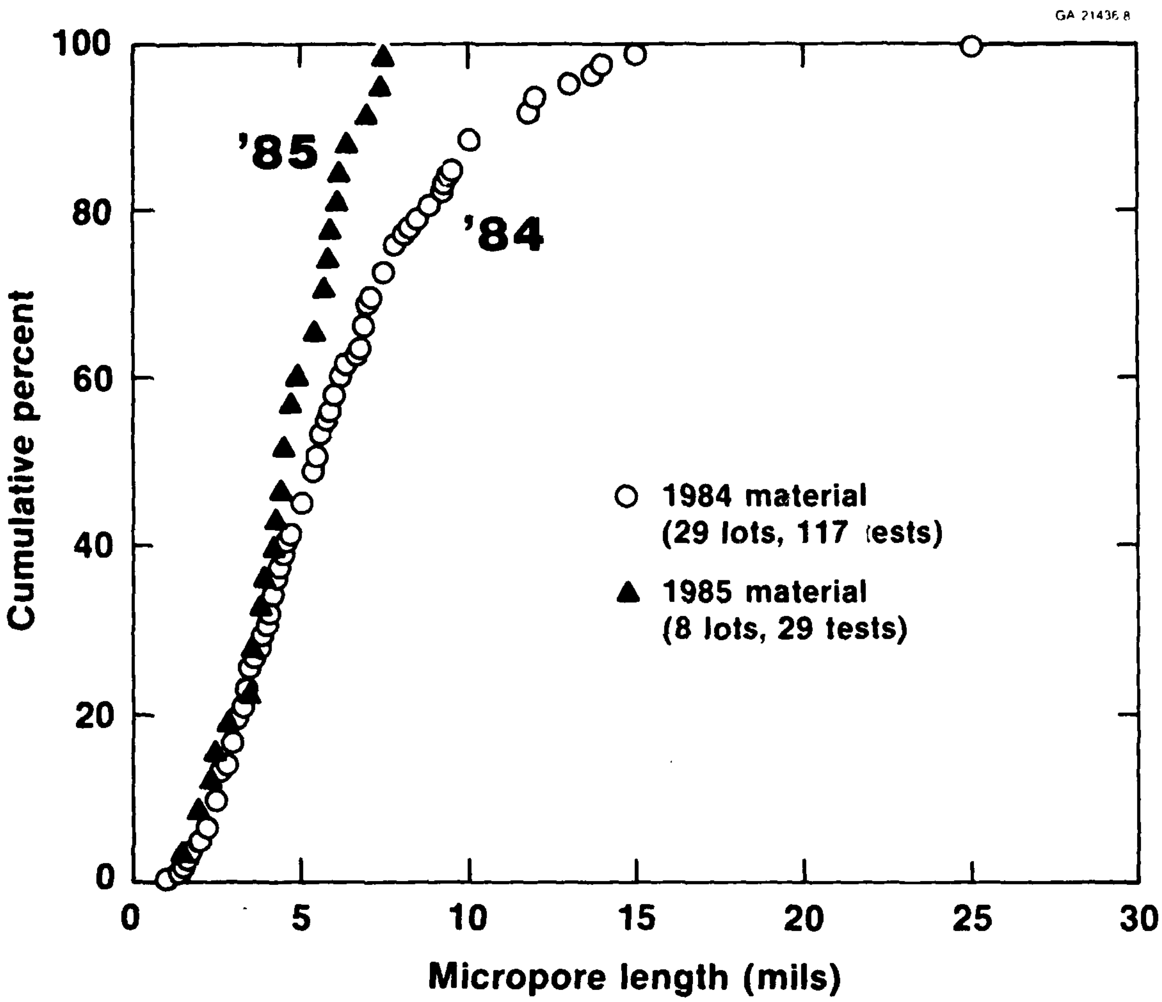
Smooth Axial Fatigue Performance at T/2 Test Location in 7050-T7451 Plate of Varying Thicknesses
(Long Transverse)
Figure 4

a = 0.0038 in.
N_f = 176,000 cycles
Specimen diameter = 0.5 in.
Long transverse test direction
σ_{MAX} = 35 ksi, R = 0.1

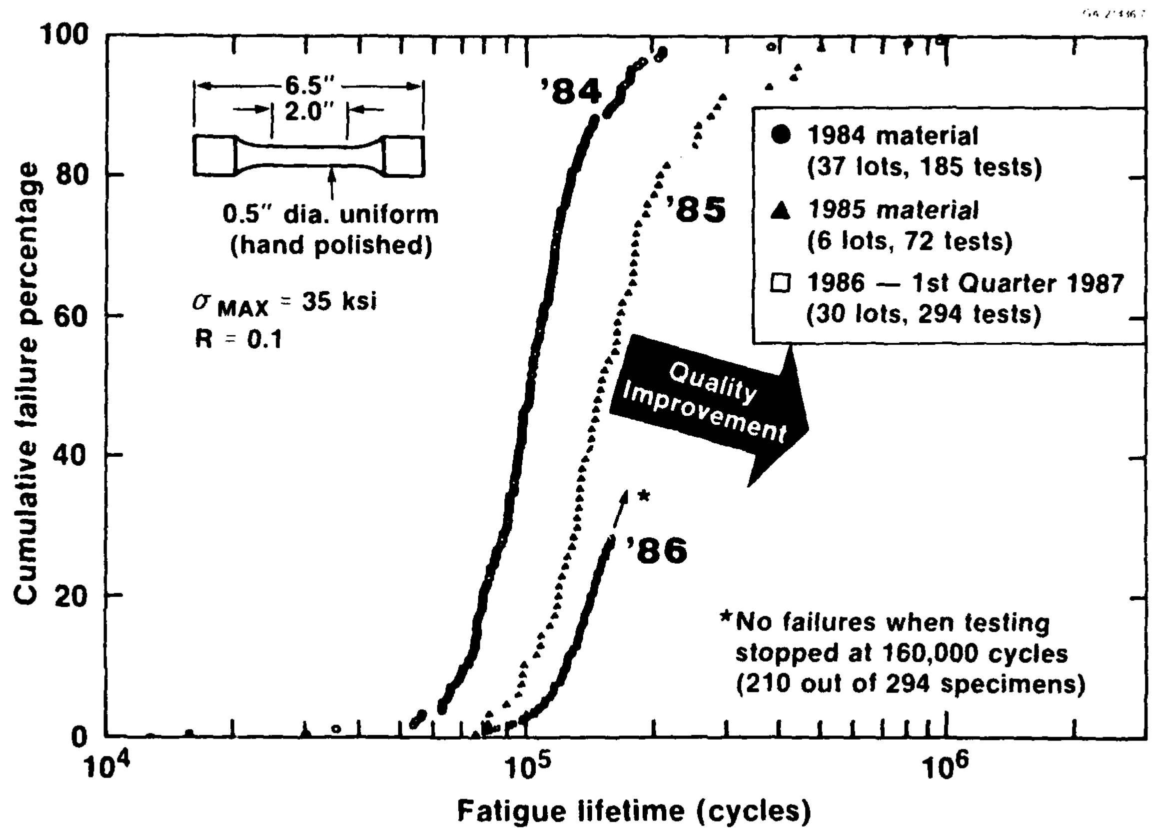
Micropore Failure Origin of Smooth Axial Fatigue Specimen
Removed from T/2 Test Location of 5.9 Inch Thick 7050-T7451 Plate (Specimen 676951-6)
Figure 5



Micropore Length vs. Cycles to Failure
Smooth Axial Fatigue Tests
7050-T7451 Thick Plate (5.7-5.9 in.)
(Long Transverse, T/2 Test Location)
Figure 6



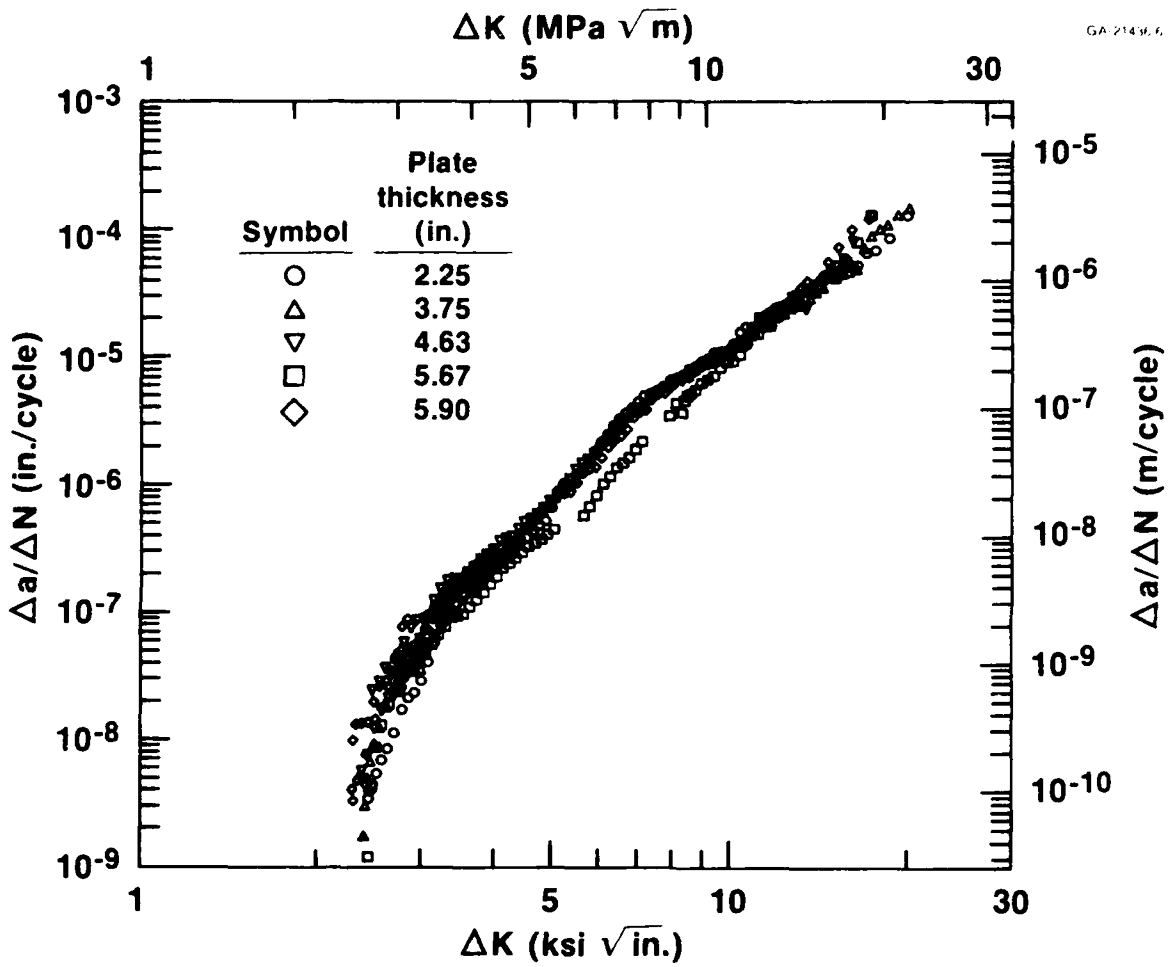
Cumulative Micropore Size Distribution
Smooth Specimen Fatigue Failure Origins
7050-T7451 Thick Plate (5.7-5.9 Inch)
Manufactured in 1984 and 1985
Figure 7



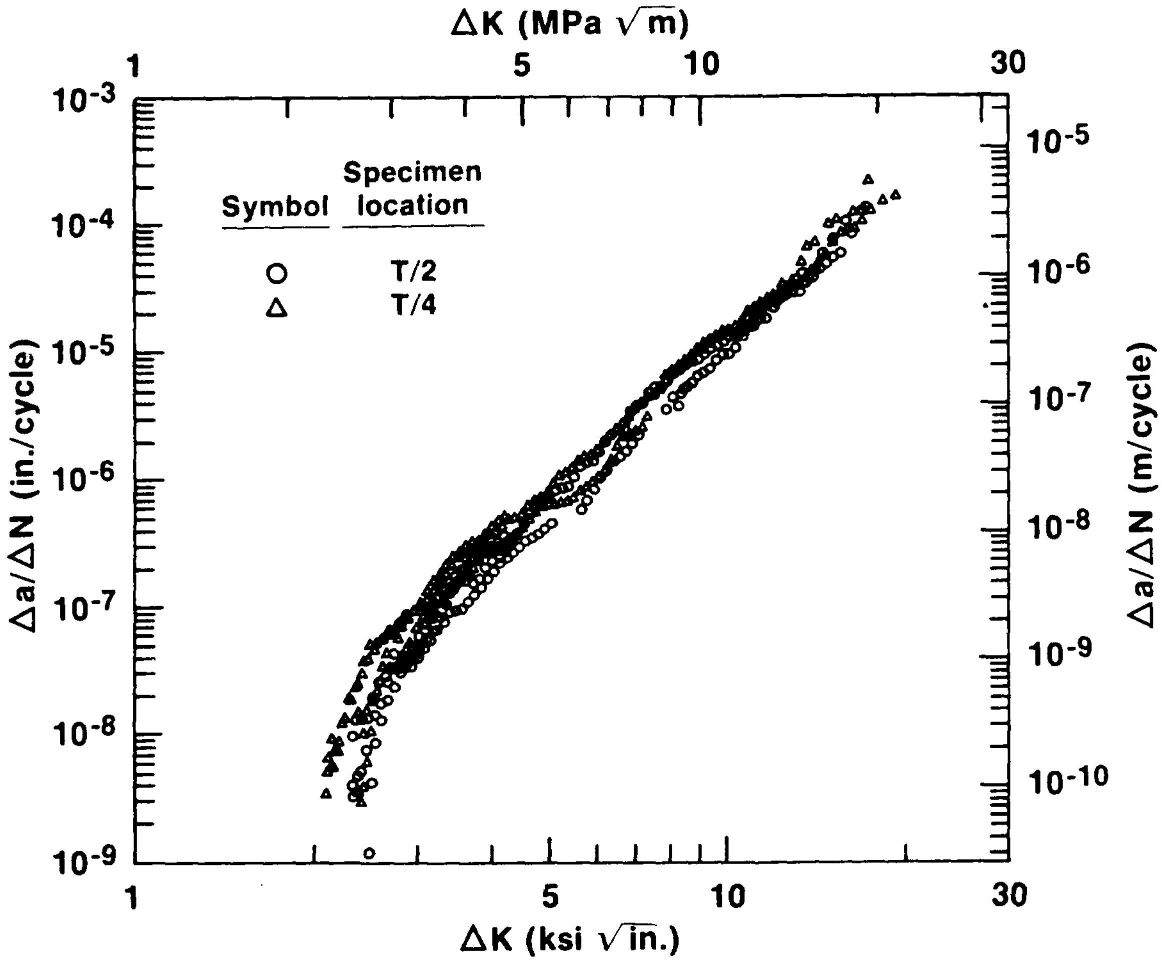
Cumulative Fatigue Failure Distributions for 7050-T7451 Thick Plate (5.7-5.9 Inch)

Produced Over 1984 to First Quarter 1987 Time Period (Long Transverse, T/2 Test Direction)

Figure 8



Fatigue Crack Growth Rates
Varying Thickness 7050-T7451 Plate
Long Transverse, T/2 Test Location
R = 0.33, Humid Air (R.H. > 90%)
Figure 9



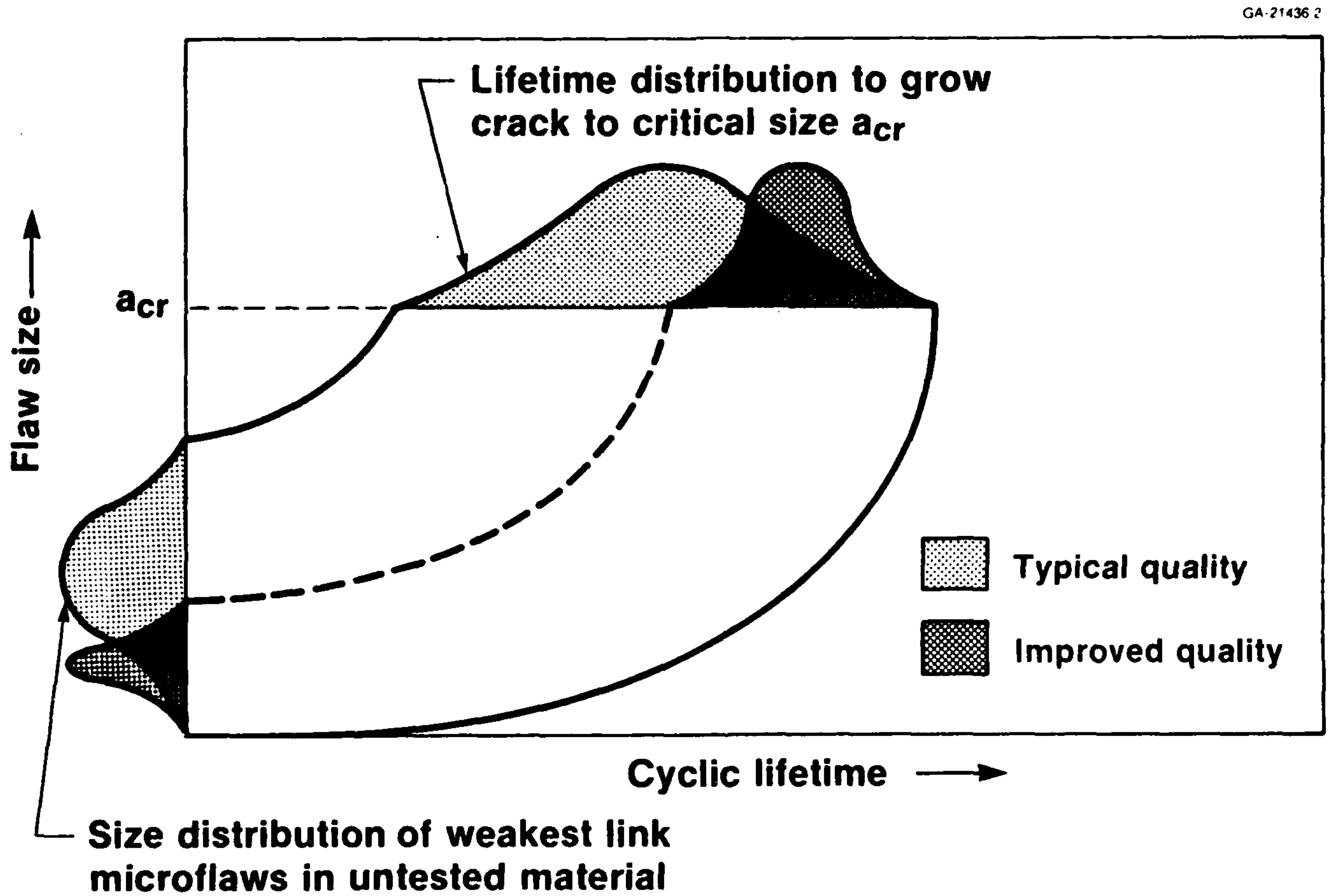
Fatigue Crack Growth Rates

7050-T7451 Plate (5.67 and 5.90 Inch Thick)

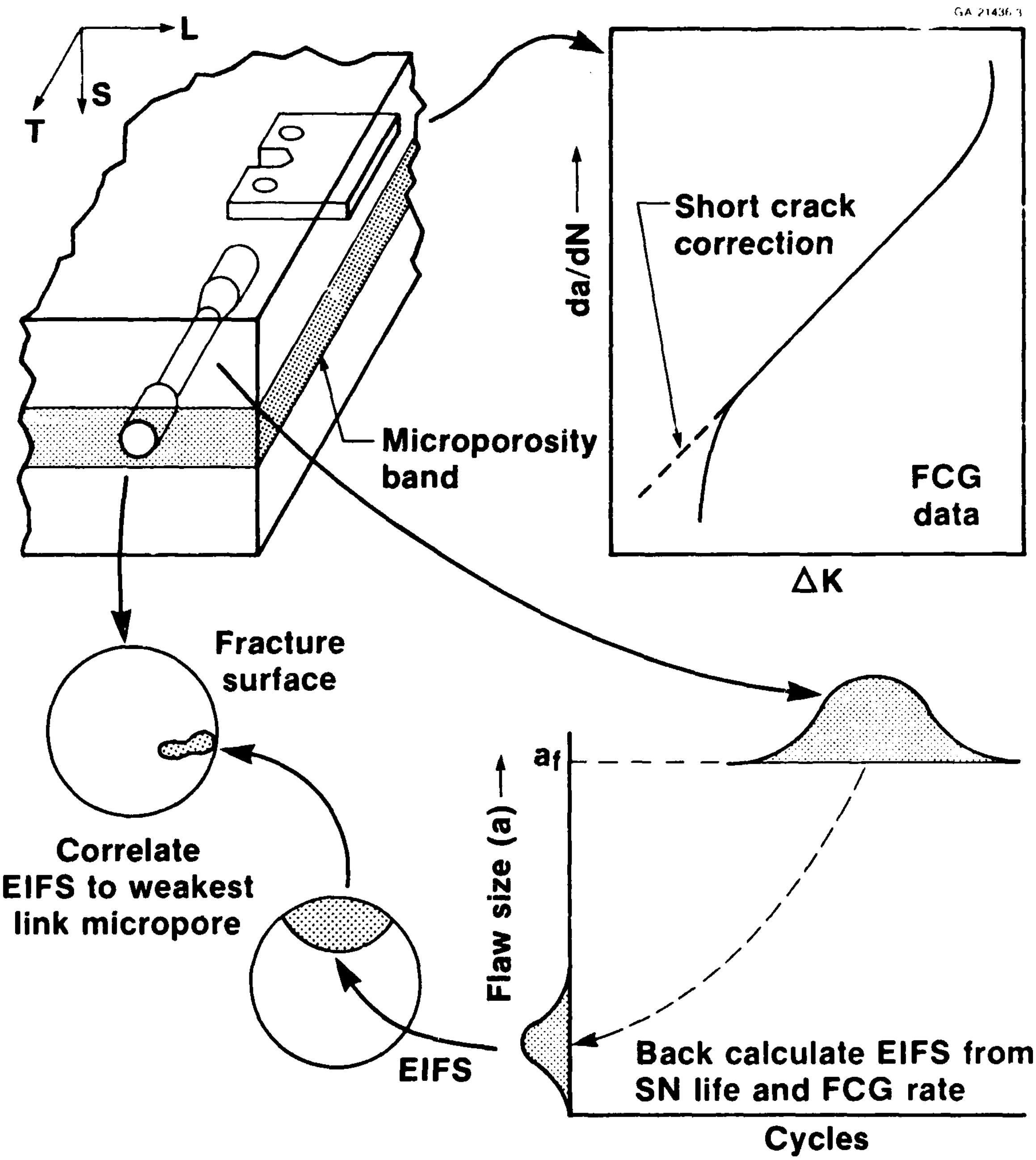
Long Transverse, T/2 and T/4 Test Locations

R = 0.33, Humid Air (R.H. > 90%)

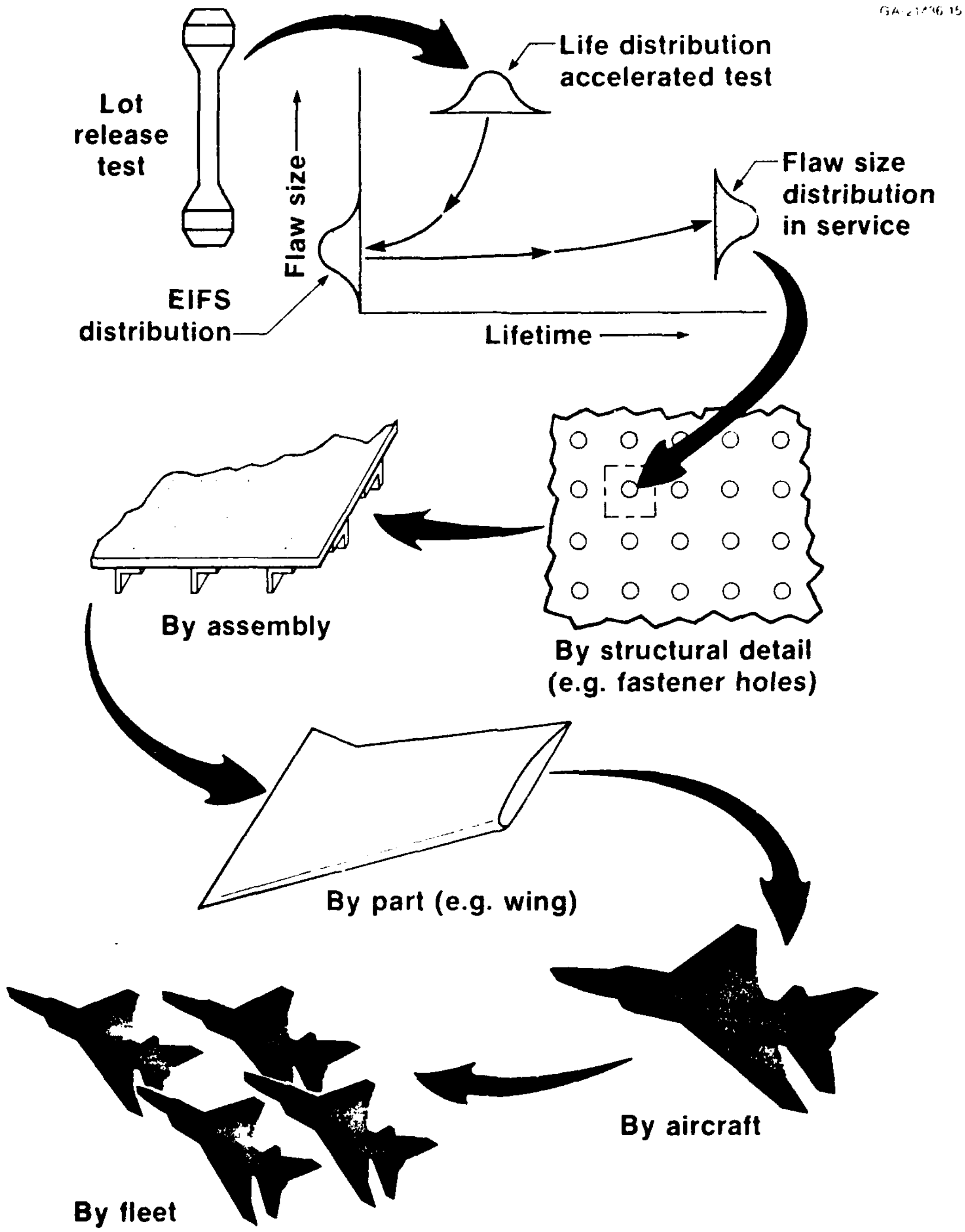
Figure 10



Conceptual Drawing Showing Quality Improvement
Translates to Longer Mean Fatigue Life and Reduced Variability
Figure 11



Fatigue Quality Screening Approach
Used for 7050-T7451 Thick Plate
Figure 12



The EIFS Distribution - Starting Point for Life Management at Various Structural Levels
Figure 13

SESSION V: HSIP/ANALYSES/METHODS

APPLICATION OF DAMAGE TOLERANCE
TO THE
H-53 HELICOPTER

George J. Schneider
Sikorsky Aircraft

Background

Sikorsky Aircraft has been under contract since late 1982 to investigate the application of damage tolerance to the H-53 Helicopter. This program has involved identification of critical structure, usage spectrum evaluation, loads evaluation, detail stress analyses, material and verification testing, and development of stress intensity solutions. The primary objectives of the program were to develop a computer program for use by Sikorsky and Warner Robins-ALC to perform crack growth analysis of helicopter structure, to evaluate crack growth in a select group of rotor and airframe structure, and to access the feasibility of damage tolerance force management for the H-53 Air Force fleet.

A presentation was made at the 1984 ASIP conference to provide an introduction to this program. Since then, Sikorsky has completed an initial crack propagation analysis of a select group of rotor and airframe structure. The purpose of this presentation is to review some of the technical issues involved in the crack growth analysis, to present some of the analysis results, and to review conclusions and lessons learned.

Viewgraph No. 1 H-53 Helicopter

The H-53 helicopter has a design gross weight of 42,000 lbs. The main rotor is 72 ft in diameter and has a normal operating speed of 185 rpm which results in a centrifugal force at each blade to rotor hub attachment of 84,000 lbs. The tail rotor is 16 ft in diameter and has a normal operating speed of 791 rpm which results in a centrifugal force at each blade to hub attachment of 35,000 lbs. The rpm of the rotors is relatively constant ranging between 98% to 108% of normal operating rpm, with overspeeds of 125% in autorotation maneuver.

The pilot controls the helicopter through collective and cyclic sticks, and a rudder control. The collective stick imparts a uniform pitch and thereby lift to all main rotor blades. Since the main rotor is a fully articulated system (i.e., hinged both horizontally and vertically), each blade will rotate at the blade to hub hinge through a flapwise angle determined by the vector sum of the centrifugal and lift forces. The blade load imparted to the rotor hub is then an out-of-plane load with both an in-plane and a vertical (or lift) component. The uniform lift created by the pilot's collective stick thus provides a vertical lift force (or thrust) to the helicopter. The cyclic stick provides a variable pitch to the main rotor blades and thus imparts a forward, sideward, or rearward thrust to the aircraft. The rudder controls the pitch of the tail rotor which in straight level flight reacts the main rotor torque. By varying the tail rotor pitch, the pilot can impart yaw motion to the helicopter.

Viewgraph No. 2 Presentation Summary

The topics discussed in this presentation include some of the technical issues in the damage tolerance evaluation, some initial crack propagation analysis results, a comparison of safe-life and crack propagation evaluations, and a review of conclusions and lessons learned.

Viewgraph No. 3 Usage Spectrum Variables

Listed are the important variables and their approximate number of variations which must be considered in defining the helicopter usage spectrum. Also listed are examples of possible flight regimes. As may be seen, the definition of a helicopter usage spectrum can be quite complex involving a large number of flight regimes and a large total number of variables (over 5000). In current safe-life (crack initiation) evaluations worst case regime severity, c.g., g.w., and altitude are normally used, thus simplifying the usage definition. This may not be entirely appropriate for damage tolerance evaluation, but additional investigations are required in this area. In the current flight data recorder being developed for the Air Force H-53 helicopter, it is planned to collect data on all the variables listed.

The issue of pilot technique is of considerable current interest. Flight test programs involving both Sikorsky test pilots as well as military pilots of various skill levels are being considered due to concerns that pilot skill and technique may significantly effect flight loads.

Viewgraph No. 4 Usage Definition

The conventional approach to defining helicopter usage spectrum is in terms of occurrences and/or percent time for each flight regime. In the H-53 damage tolerance assessment program an early decision was made to define a simulated real time usage spectrum out of concern for spectrum sequence effects on crack propagation. This real time usage spectrum was defined in terms of mission segments and flight conditions which is apparently a carry over from fixed wing practice. However, it is now believed that a random sequencing of helicopter loads is sufficiently accurate, and that it is more important to accurately represent the important variables in viewgraph 3. It is, therefore, planned to use the conventional approach to defining helicopter spectrum in H-53 flight data recorders and future damage tolerance evaluations since it is inherently simpler, and easier to incorporate the important flight variables.

Viewgraph No. 5 Main and Tail Rotor Structure

The primary components of the main and tail rotor head assemblies are illustrated. The structures selected for damage tolerance evaluation on this contract are underlined.

Viewgraph No. 6 Some Potential Main Rotor Crack Locations

Potential crack locations are illustrated in the cut away view of one arm of the main rotor head assembly. As may be seen, the crack locations are difficult to inspect and require disassembly of the rotor head. The crack locations evaluated in this contract are noted by a double asterisk(**). Potential crack locations on the tail rotor head are equally difficult to inspect.

Viewgraph No. 7   Airframe Structure Analyzed

The airframe structures selected for damage tolerance evaluation are illustrated. All structures, except the transmission support, were selected based on in-service fatigue crack experience. The transmission support structure was selected due to its primary function in transferring main rotor loads into the airframe.

Viewgraph No. 8   Measured Flight Loads Data

The fatigue loads on the structures selected for crack growth analysis are usually obtained from flight test measurements. During a flight test program, each flight regime is flown multiple times. During each regime occurrence (run) an analog loads data burst is recorded. These data bursts are illustrated on the left.

The processing of each data burst (run) involves an assessment of the dominant frequency (i.e., harmonic) in the analog signal and digitizing the signal to determine all load peaks and valleys at this frequency. The 95% or maximum vibratory load and its associated steady for each data burst are then determined. Each data burst (run) is then conservatively assumed to be represented by its 95% or maximum vibratory, which are plotted as illustrated in the center.

As shown on the left current safe life fatigue evaluation is based on the high envelop 95% or maximum vibratory. In other words, it is conservatively assumed that the high envelop vibratory occurs for the full time a specific regime occurs. In the H-53 damage tolerance crack growth analysis, the full range of maximum and 95% loads experienced in the flight test program were used. Funding is now being pursued with WR-ALC to evaluate the effect of using the actual flight test cycle count data.

The correct use of flight test loads data is still controversial. The choice of high envelop data for current safe life evaluation is due to the uncertainty in flight loads resulting from not well understood day-to-day, aircraft-to-aircraft, and pilot-to-pilot variations. Caution will, therefore, be necessary in using actual cycle count data.

Viewgraph No. 9   Typical Max/95% Main Rotor Push Rod Loads

Illustrated is a typical complete set of 95% and maximum vibratory loads and associated steady loads obtained from the H-53 1983 flight test program for a main rotor push rod. Most flight regimes are represented in this plot. Level flight loads are plotted as a function of airspeed (not shown). The full range of loads shown in this plot was used in the H-53 damage tolerance analyses, whereas only high envelop loads are normally used in safe life fatigue substantiation. The variation in load for a specific flight regime is the result of many factors including cg, gw, maneuver severity, and maneuver-to-maneuver scatter.

Viewgraph No. 10 H-53 Helicopter Load Frequencies

As discussed, the loads in helicopter structure are normally characterized by a dominate frequency. Rotating structure on the main and tail rotor heads normally are characterized by a once per revolution frequency of 3.1 Hz and 13.2 Hz respectively. Stationary structure on the main and tail rotor heads and local airframe support structure is normally characterized by a η per rotor revolution frequency (η = number of blades) of 18.5 Hz and 52.7 Hz, respectively. An η per main rotor revolution (18.5 Hz) is experienced by the stabilizer resulting from main rotor down wash, and some airframe tail structure is apparently sensitive to a 3 Hz first mode airframe frequency. As noted from this and previous viewgraphs, helicopter structure is subject to fairly high loads at high frequencies (3 to 53 Hz).

Viewgraph No. 11 Stress Analysis - Main Rotor Finite Element Models

Stress analyses of individual structures were performed to relate load to stresses a long potential crack paths. Various methods of stress analyses were used including boundary element and finite element analysis. Illustrated are the NASTRAN finite element models constructed for main rotor structure. In general, it was found that these models provided basic load path information, but required "rezoning" to obtain accurate stress distributions for small surface cracks (.010 inch deep).

Viewgraph No. 12 Stress Magnitudes

Steady and vibratory stress magnitudes are shown for one main rotor and two airframe structures to indicate the severity of stresses in helicopter structures. The frequency of the vibratory stress and the local stress concentration factor are also presented. As noted for the main rotor spindle, the steady stress is dominated by the stress produced by the centrifugal load. This is typical of main and tail rotor rotating structure.

Viewgraph No. 13 Material and Verification Tests

Material and verification tests were conducted as part of the H-53 damage tolerance contract. Material testing was performed with compact tension specimens and emphasized near threshold and spectrum effects. Verification testing was conducted on notched specimens with small EDM defects in the notch to investigate surface cracks growing in nonuniform stress fields under constant amplitude and spectrum loads. Analysis test correlation is in progress, but it is apparent that additional testing to evaluate small crack growth rates (cracks less than .020 inch depth), and threshold variability and retardation behavior are required.

Viewgraph No. 14 Crack Propagation Results for Rotor Retention Structure

The results of crack propagation analysis for 10 crack locations on main and tail rotor structure are presented. The bar graph indicates the cumulative percentage of the 10 crack locations which would provide mean crack propagation times greater than 200-500 flight hours for various inspectable crack sizes from .005 inch to .030 inch deep; i.e., if an inspection procedure could reliably detect .010 inch deep crack, 50% of the 10 crack locations (i.e. 5

locations) would result in mean crack propagation times greater than 200-500 flight hours. It should be recalled that these crack times are based on loads which may be conservative and possibly conservative usage spectrum. However, the results due indicated that reliable NDI of .010 to .020 inch deep cracks would probably be required to achieve reasonable inspection internals for a majority of rotor structure.

For the airframe structures evaluated on this contract, all crack propagation times were short except for the main rotor transmission support. The transmission support structure was the only structure evaluated which did not have a service history of cracks. It exhibited good damage tolerance capability and is considered typical of most airframe structure. The few problem areas (i.e. other structures evaluated) may require design modification to meet damage tolerance requirements. However, crack growth analyses with cycle counted loads rather than the max and 95% loads now being used is considered necessary for reliable decisions on airframe structure.

Viewgraph No. 15 Safe Life - Crack Propagation Evaluation

A comparison of recently published safe life replacement times for H-53 rotor structure and the H-53 rotor structure crack propagation times indicates no consistent trends, i.e., components with long safe life replacement times do not necessarily have long crack propagation times and vice-versa. The reason for this is not known, and it is somewhat difficult to evaluate due to the differences in safe life and crack propagation analysis illustrated in the flow diagram.

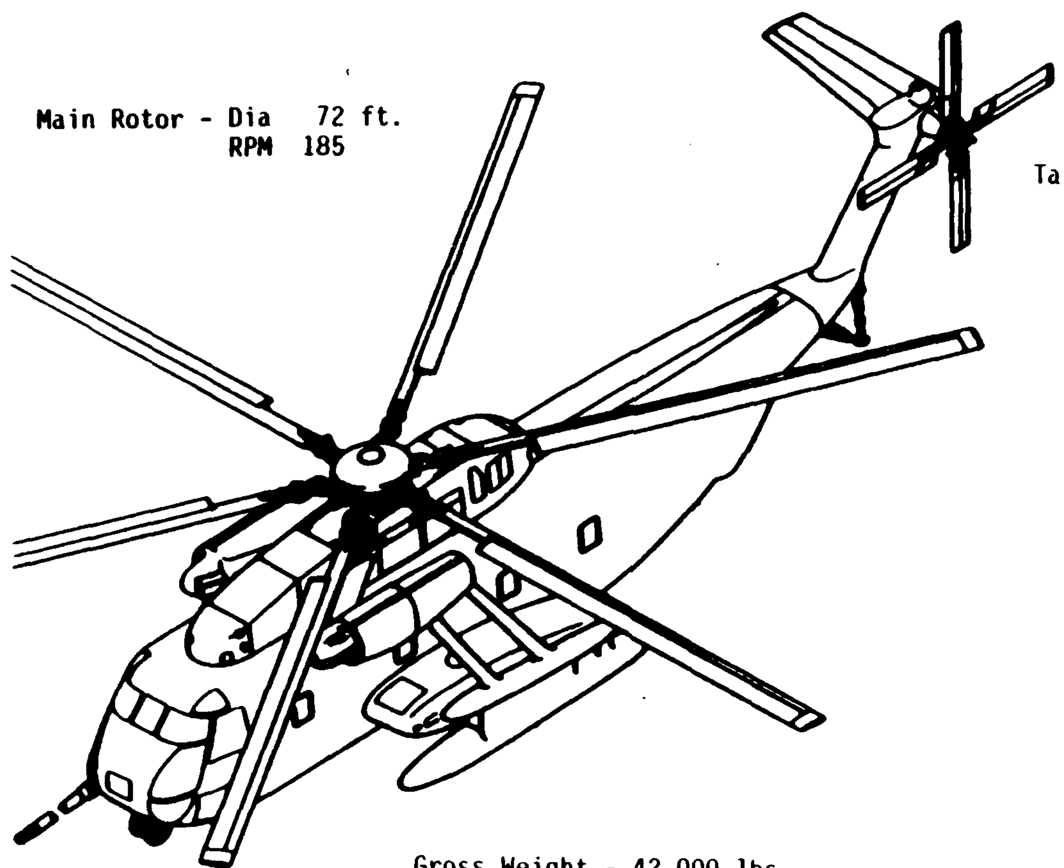
Viewgraph No. 16 Technical Conclusions

Technical conclusion and lessons learned are itemized and are fairly self-explanatory.

Viewgraph No. 17 General Conclusions

General conclusions are itemized and are fairly self-explanatory.

H-53 HELICOPTER



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PRESENTATION SUMMARY

DAMAGE TOLERANCE EVALUATION

- Usage Spectrum
- Structure and Crack Locations
- Flight Loads
- Stress Analysis
- Crack Propagation Test Data

RESULTS OF H-53 DAMAGE TOLERANCE ANALYSIS

SAFE LIFE - CRACK PROPAGATION COMPARISON

TECHNICAL CONCLUSIONS

GENERAL CONCLUSIONS

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USAGE SPECTRUM VARIABLES

Important Variables

Flight Regime	70
Regime Severity and Airspeed	6
C.G. and G.W.	4
Altitude	3
Rotor RPM	-
Pilot Technique and Proficiency	-
	5040

Flight Regimes

1. Rotor Engagement
2. Taxi
3. - left turn
4. - right turn
5. Takeoff - vertical
6. - running
7. Hover (IGE)
8. - left turn
9. - right turn
10. - longitudinal reversal

°
°
°

16. - rearward flight

30. Level Flight
31. - left turn
32. - right turn
33. - left turn recovery

°
°

42. - left sideslip
43. - right sideslip

°
°

55. Landing from Hover
56. Run-On Landing
57. Rotor Shutdown

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USAGE DEFINITION

Conventional Approach

Regime	Occur./100 Hrs	Percent Time
1. Rotor Start	40	- -
⋮		
7. Hover IGE		6.9
⋮	100	0.33
31. Level Flight Left Turn		2.60

H-53 DTA Real Time Usage Spectrum

Time (min)	Mission Type	Mission Segment	Flight Cond.	G.W. (lbs)	C.G. (in)	Speed (kts)	Altitude (ft)
⋮							
18.5	Training	Lev. Flt.	Steady	37547	359	128	1000
20.2	Training	Lev.Flt.	Lt. Turn	37484	359	124	1000
20.5	Training	Lev.Flt.	Steady	37413	359	137	1000
⋮							

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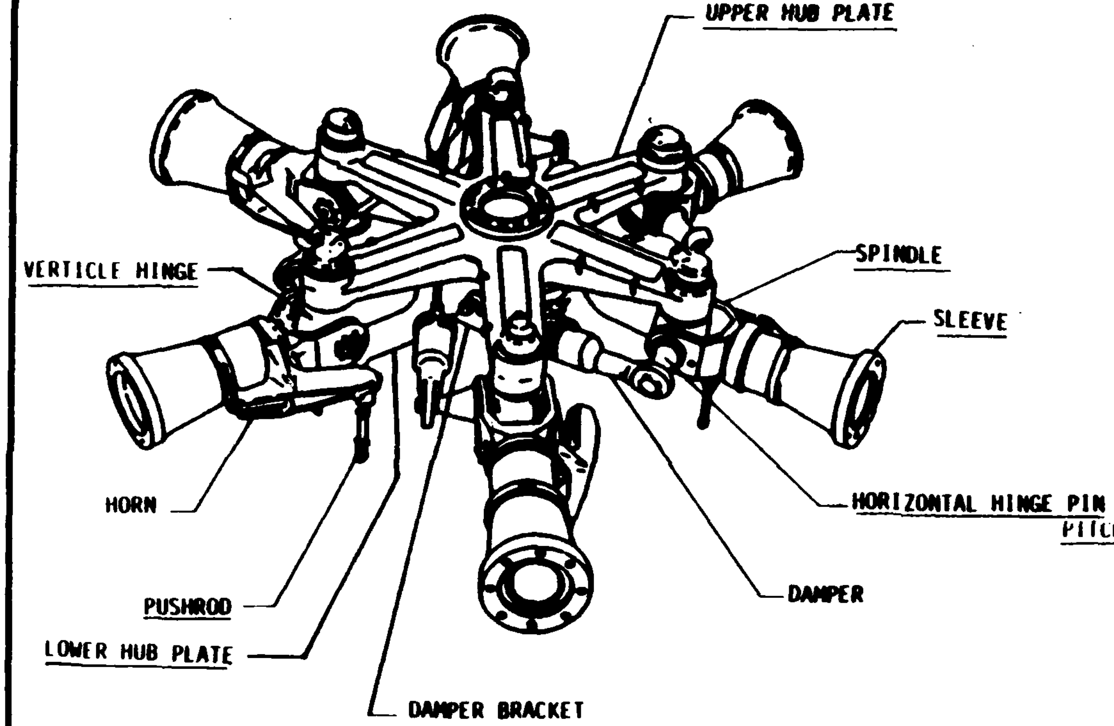
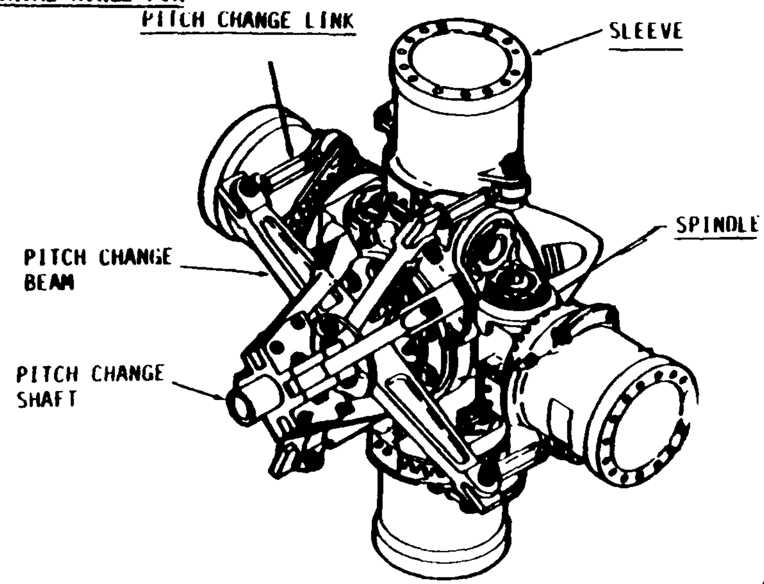
MAIN AND TAIL ROTOR STRUCTURE ANALYZED





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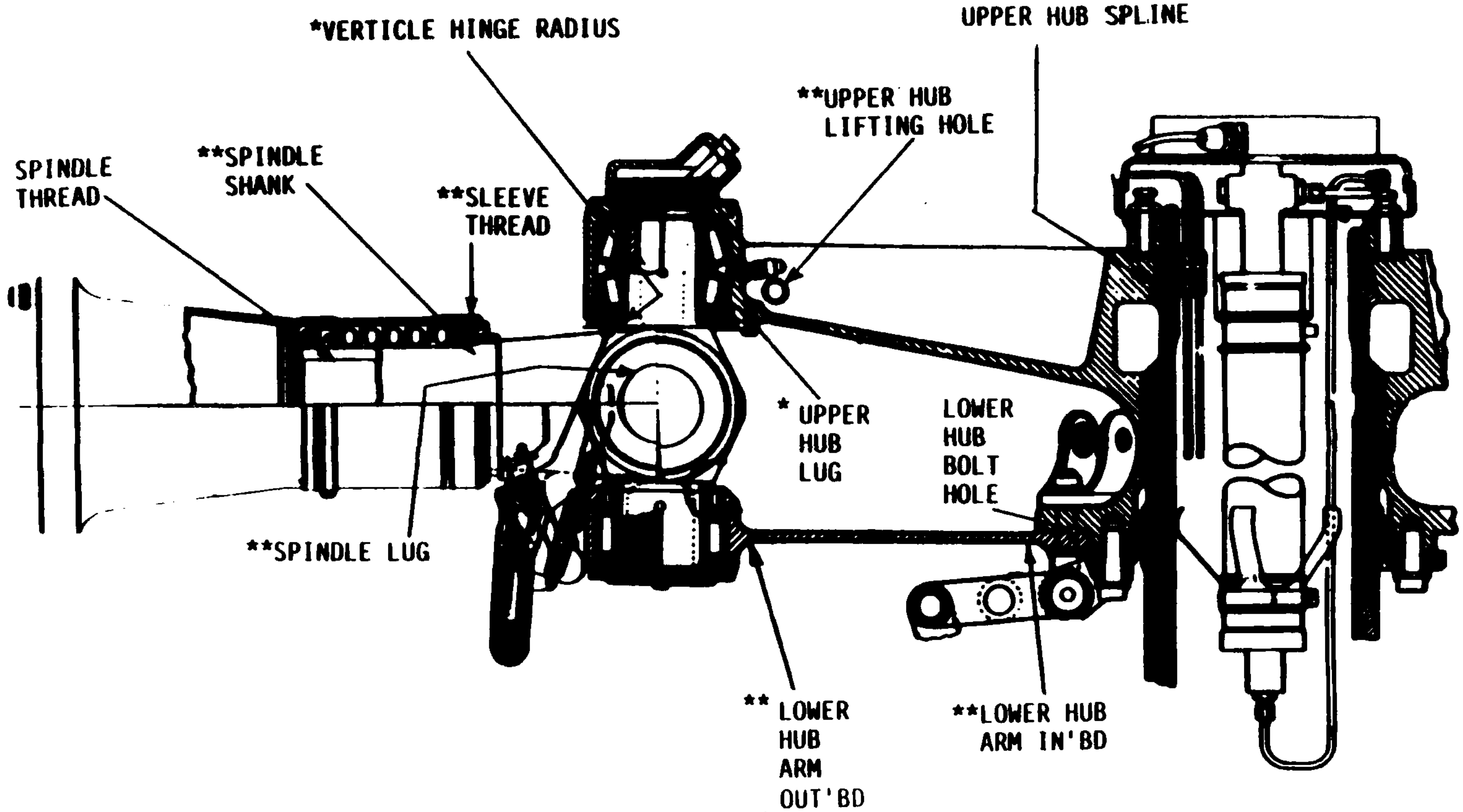
SOME POTENTIAL MAIN ROTOR CRACK LOCATIONS



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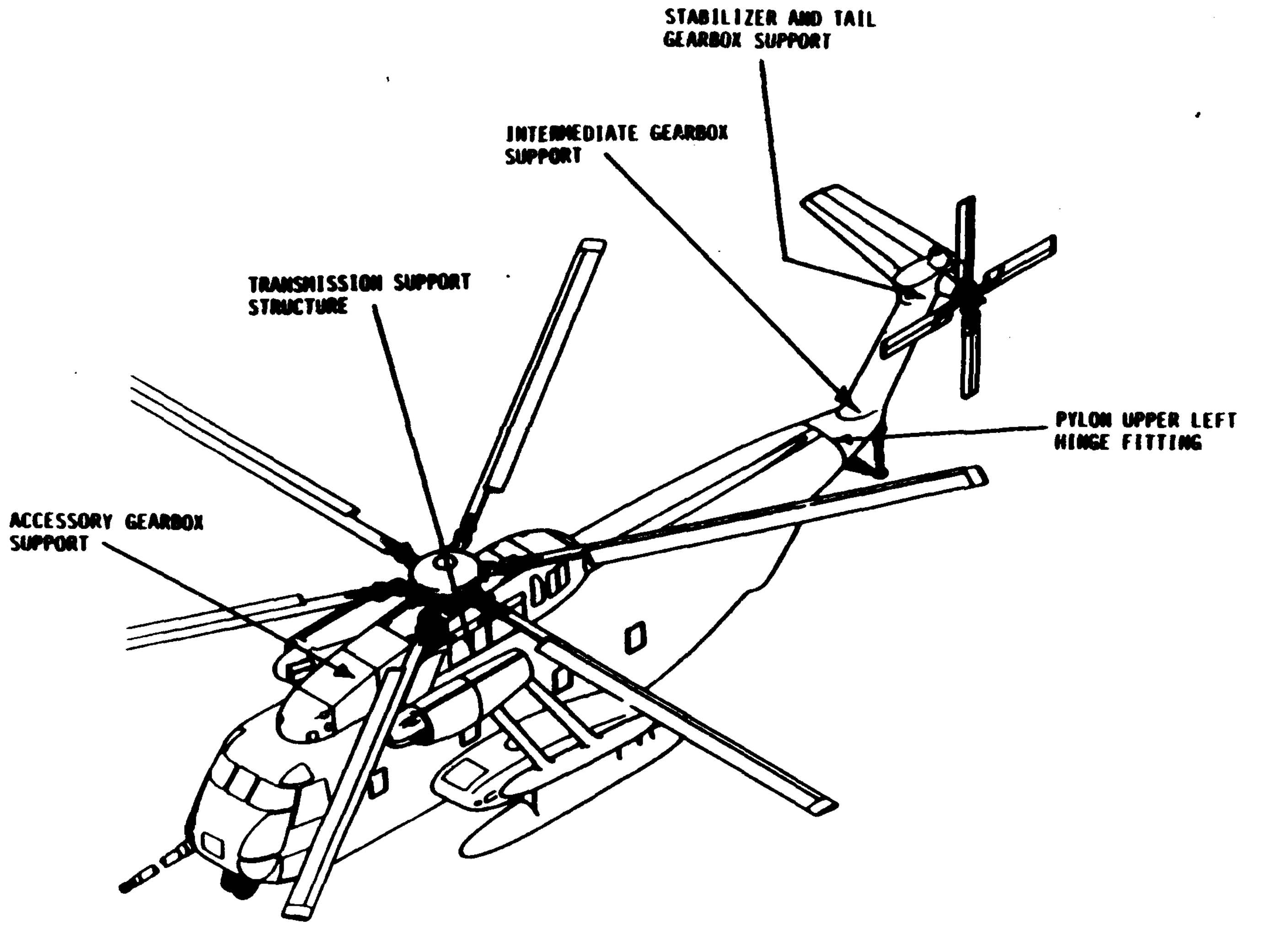
AIRFRAME STRUCTURE ANALYZED



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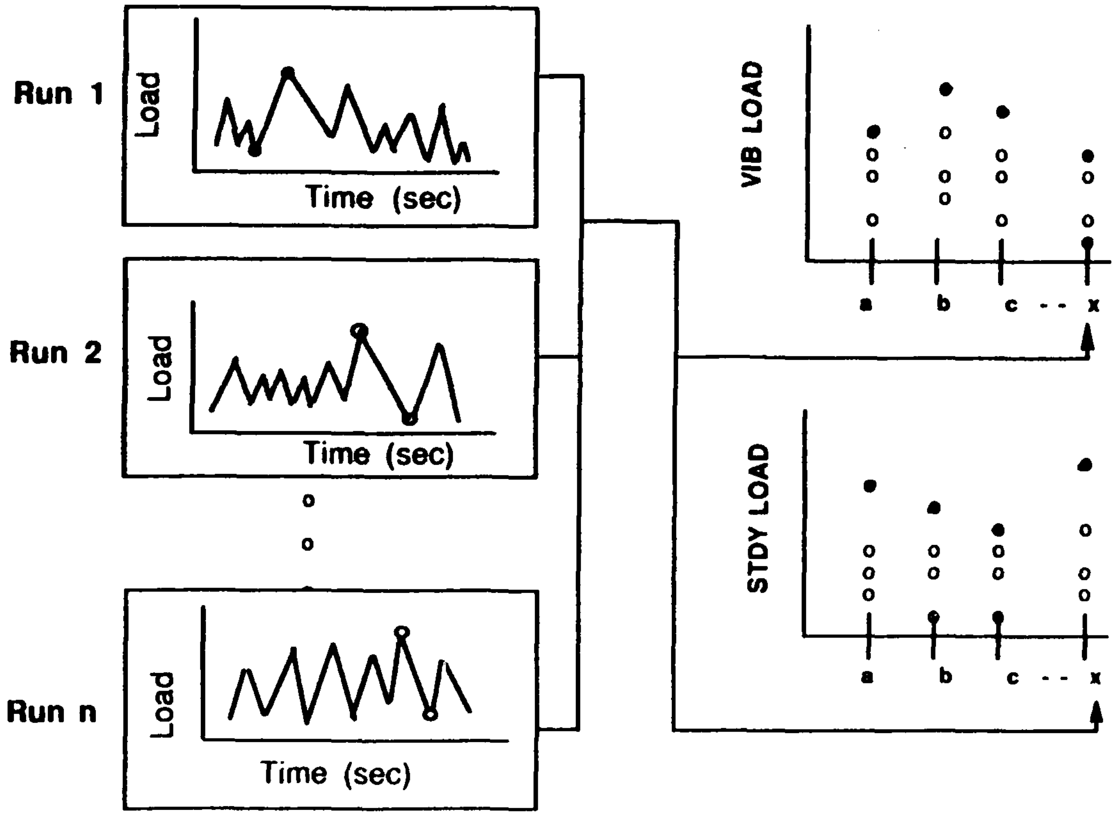
MEASURED FLIGHT LOADS

Flight Measured Data

Regime X Data Burst

Processed Flight Data

Steady State Flt Cond: 95% Vib
Transient Flt Cond: Max Vib



Flight Data Application

Conventional Safe Life

- High Envelope Load

Current DTA

- Full Distribution of 95%/Max. Vibratory

Future DTA

Cycle Count of Flight Regime

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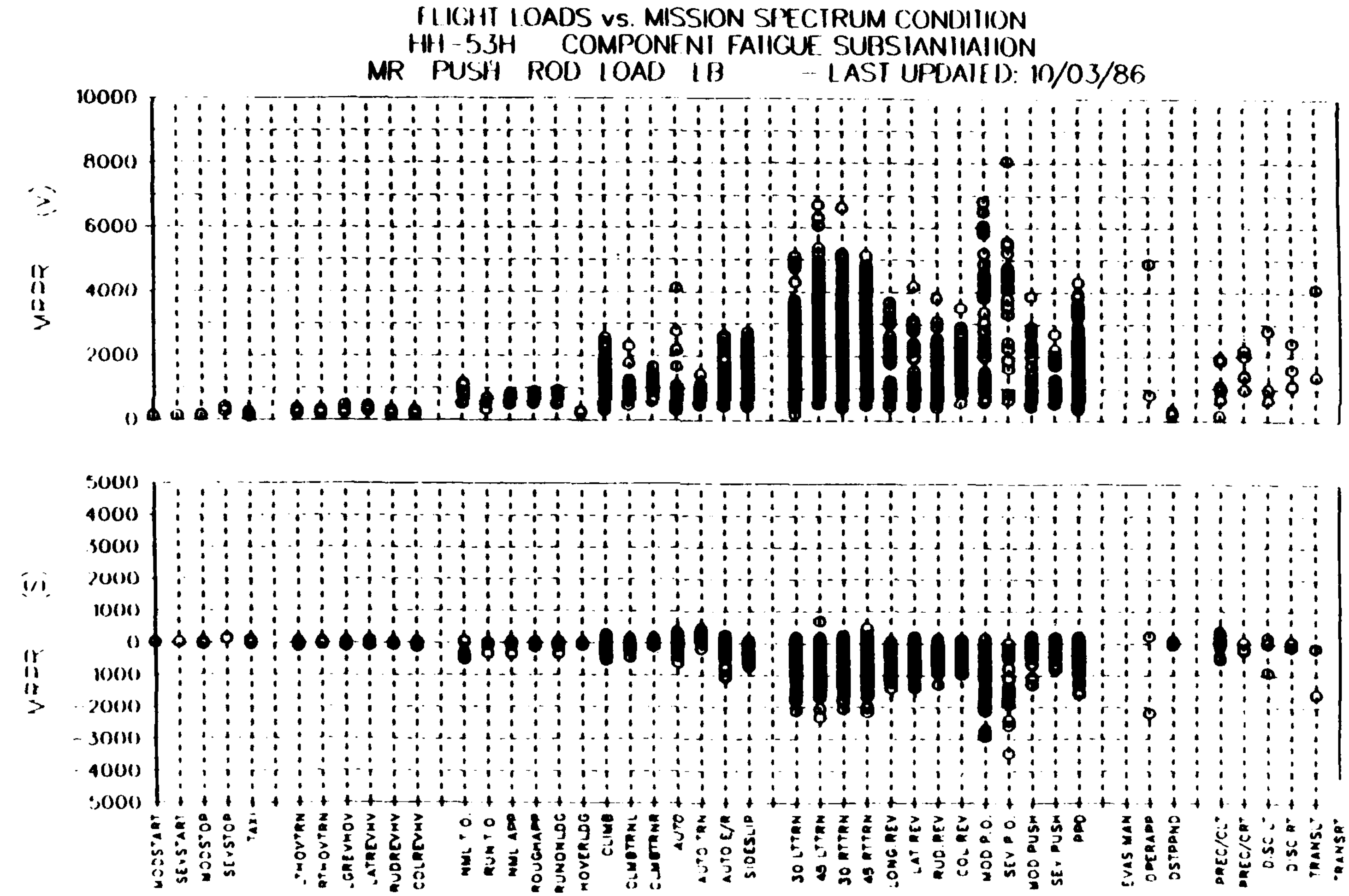
TYPICAL MAX/95% MAIN ROTOR PUSH ROD LOADS



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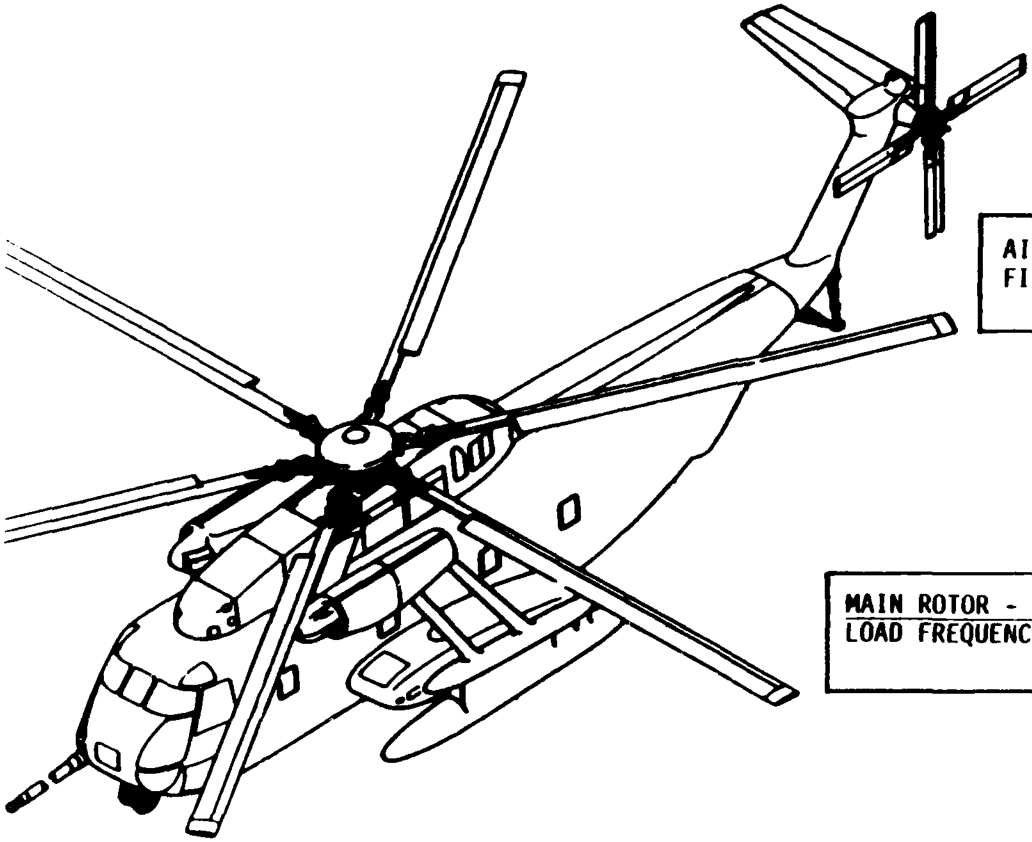
H-53 HELICOPTER LOAD FREQUENCIES



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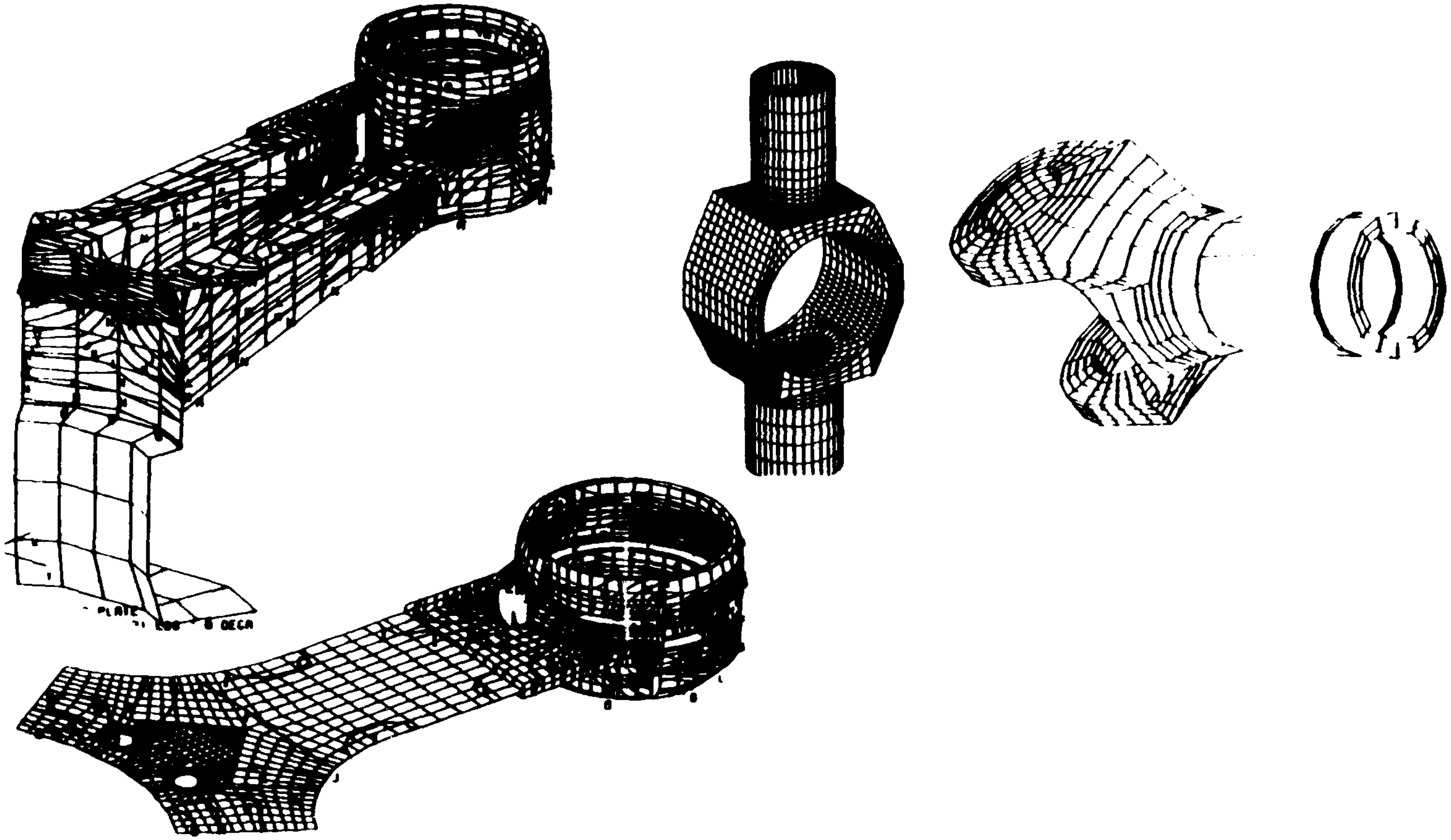
STRESS ANALYSIS-MAIN ROTOR STRUCTURE FINITE ELEMENT MODELS



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STRESS MAGNITUDES

	MR Spindle Lug (3.1 Hz) (K_t =2.42) σ_0(KSI)	Upper Left Pylon Hinge (3.1 Hz) (K_t =3.19) σ_0 (KSI)	MR Transmission Support (18.5 Hz) (K_t =3.2) σ_0 (KSI)
Centrifugal Load	30.4	—	—
Cruise and Moderate Man'ver	33.0+0.6 ⋮ 34.0+4.0	20.0+4.0 ⋮ 54.0+5.0	10.0+1.0 ⋮ 30.0+1.0
Severe Maneuver	27.0+5.5 ⋮ 35.0+7.5	55.0+13.0 ⋮ 72.0+10.0	— —
Ground Air Ground (GAG)	30.0+30.0	35.0+52.0	16.0+20.0

NOTE: σ_0 = crack origin stress

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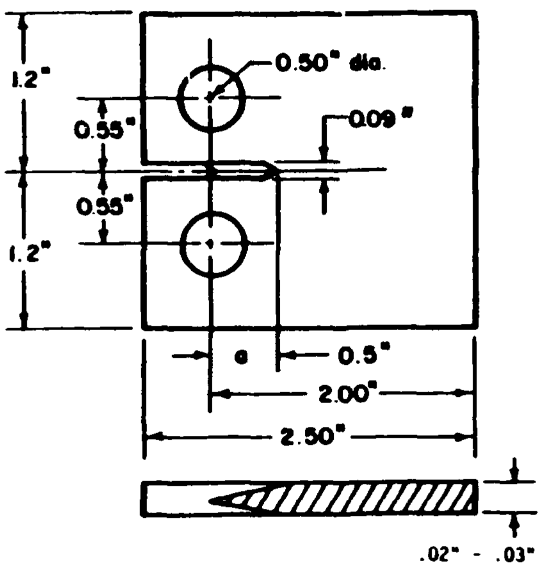
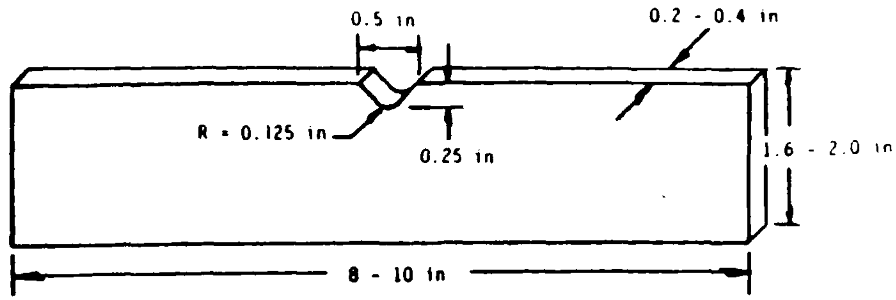
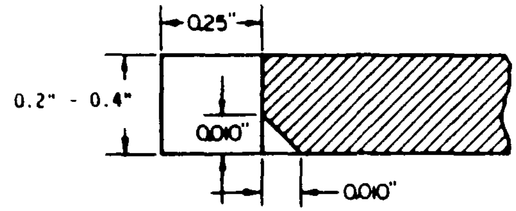
MATERIAL AND VERIFICATION TESTS





(a) Notched Specimen



(b) EDM Flaw Detail

Compact Tension Specimens

- **Constant Amp. Tests - Material Data**
- **Spectrum Tests - Spectrum Effects**

Notched Specimens

- **Const. Amp. Tests - Stress Intensity**
- **Spectrum Tests - Stress Intensity**
  - Spectrum Effects

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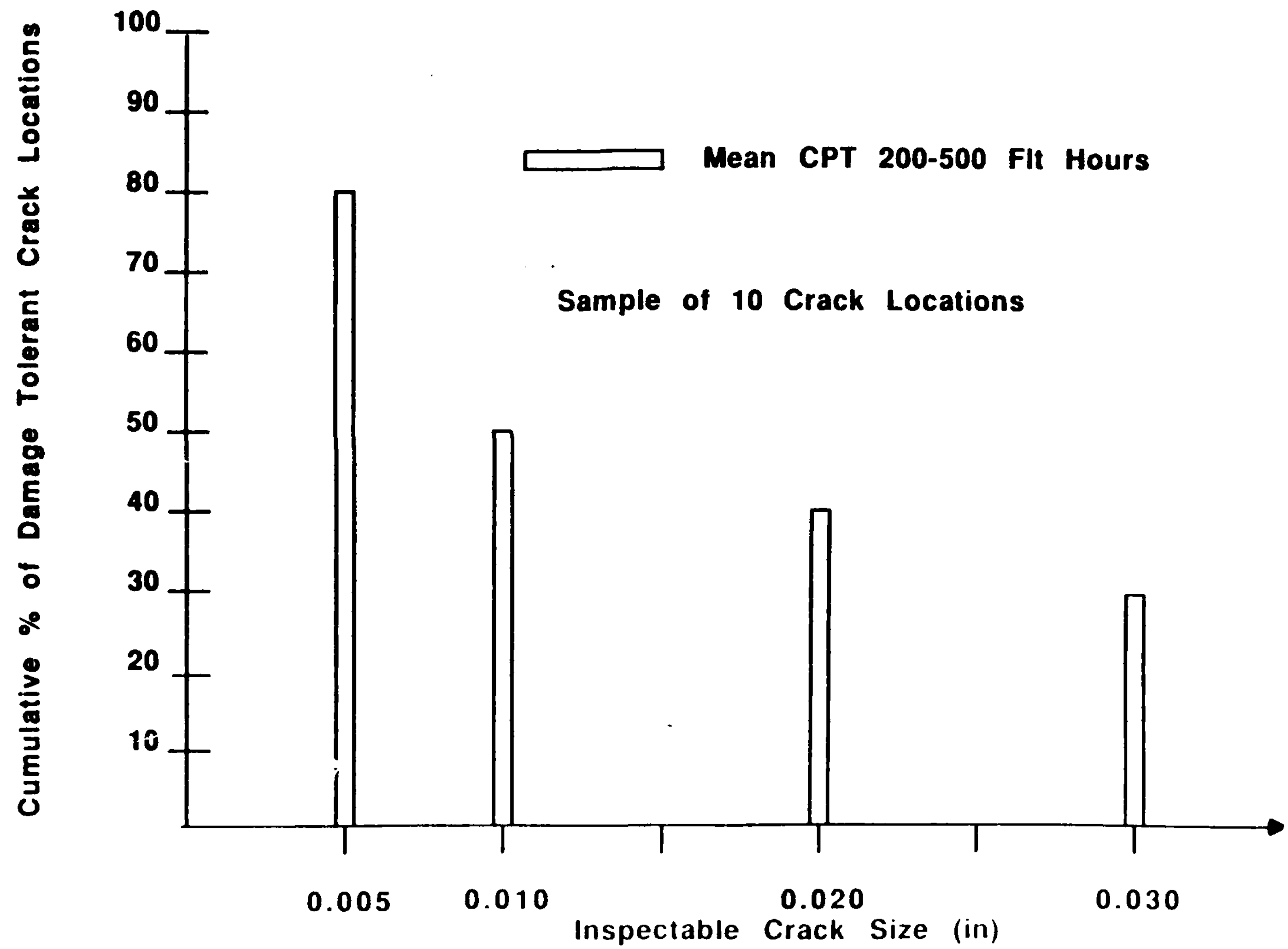
Crack Propagation Results for Rotor Retention Structure



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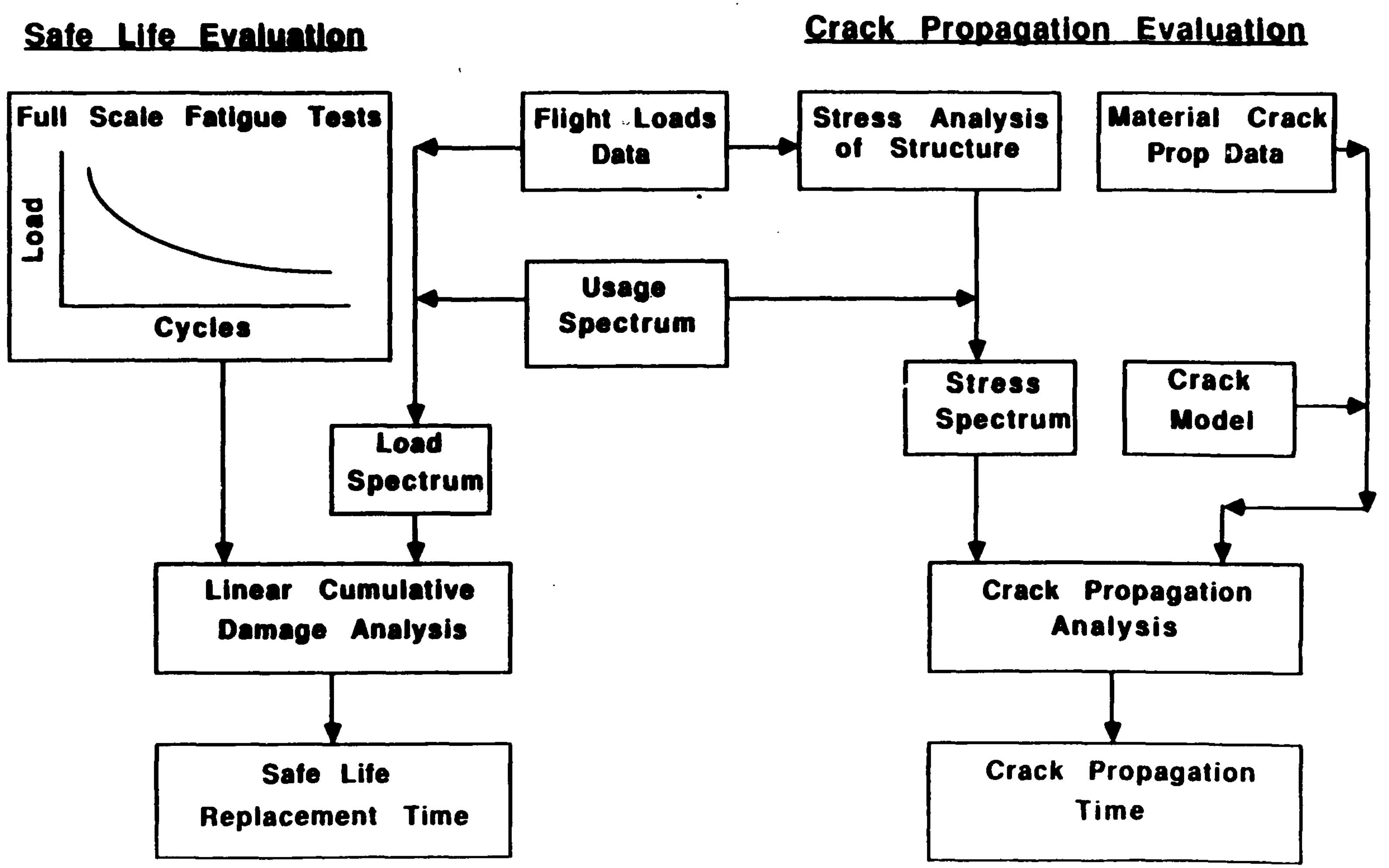
SAFE LIFE - CRACK PROPAGATION EVALUATION



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TECHNICAL CONCLUSIONS

- Usage Spectrum Is Best Represented In Conventional Form
  - planned in helicopter tracking contract and future DTA
- Actual Usage May Be Less Severe
  - helicopter tracking contract
- Cycle Counting Is Recommended to Avoid Conservatism in Crack Propagation
  - planned investigation
- Both Rotor and Airframe Loads Rely on Flight Measurement
- Loads Variability Is an Important Issue
  - maneuver repeatability
  - pilot technique and proficiency
  - needs investigation in flight testing
- Stress Analyses Should Be Verified
  - planned investigation
- Safe Life Designed Structure Can Be Sensitive to Crack Growth
- Additional Crack Propagation Data May Be Required
  - near threshold variability and spectrum effects
  - smaller initial cracks (0.010 - 0.020 inches)
  - planned investigation/test program

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GENERAL CONCLUSIONS

Helicopter Damage Tolerance Issues Have Been Identified

Airframe Structure

- generally damage tolerant
- hot spots
  - attention to high frequency stresses
  - possible design modification

Rotor Structure

may require

- improved technology
- design change, e.g. elastomeric rotor head
- NDI 0.01 to 0.03 inches

Results of This Program Indicate That We Should Continue to Pursue Damage Tolerance For Helicopter Structure

UNITED TECHNOLOGIES SIKORSKY AIRCRAFT

Individual Helicopter Tracking Program (IHTP)
For The
MH-53J Helicopter.

John G.B. Daniell
Project Engineer- Diagnostics, Sikorsky Aircraft.

Abstract

The cost and complexity of maintaining aircraft in the Air Force inventory escalates with time. New methods are required to increase the cost effectiveness of fleet aircraft maintenance while simultaneously decreasing down time and improving readiness rates.

A system is being developed which can collect details of usage of each helicopter in the fleet automatically. The resulting data base can then be used in conjunction with analytical processes to determine component inspection intervals using Damage Tolerance Assessment (DTA) techniques. This will replace the present assumption of one universal usage spectrum for all, regardless of the actual usage of individual aircraft.

The helicopter peculiar aspects of component life estimation and the present methodology will be reviewed. The new system will be described and the process involved will be outlined.

Background

The need to identify and record aircraft usage information has been well recognized for a number of years. Early efforts began many years ago with Vgh recorders, and crew questionnaires, to obtain fixed wing aircraft data and obtain some insight into fatigue life usage. Since fixed wing aircraft designs employ relatively damage tolerant structures with built in redundancy, an inspection frequency that bore an approximate relation to crack growth served to maintain a margin of safety. The subsequent introduction of microprocessors using digital software is revolutionizing operational recording systems for fixed wing aircraft, and for helicopters too. The Air Force has initiated a major program to upgrade the HH-53B/C helicopter fleet to a new configuration designated MH-53J. This will extend the service life of this aircraft well into the next century. Application of IHTP to this model will contribute to this extension, enhancing safety aspects, and helping control the cost of ownership.

However, applying IHTP to helicopters as opposed to fixed wing poses unique technical challenges because the two types of aircraft have very little in common, apart from the fact that they both fly. We will take a look at the requirements, some of the major differences, and the technical approach we are taking.

Figure 1

The Air Force has specific requirements for force management which include usage spectrum monitoring (USM), and Loads and Environmental Spectra Surveys (L/ESS). These requirements are primarily written for fixed wing aircraft, but can also be applied to rotary wing.

Figure 2

By their nature, rotary wing aircraft demand a different approach to usage monitoring, since what is important in fixed wing is often secondary in rotary wing, and vice versa. This figure highlights the principal differences. The predominance of periodic loads associated with helicopter flight , a very minor consideration in fixed wing operations, is one obvious difference. A typical load cycle for, say, a C-141 aircraft, is a complete flight, with random loads imposed on the structure by gusts. The effects of these gusts on the structure are dictated by the gust direction and velocity, the speed of the encounter, and other factors such as gross weight. If we consider one flight, we can divide it into segments - Take-off, Climb, Cruise, Descent, and Landing. Each flight also contributes one Ground - Air - Ground (GAG) cycle. Handled this way, we can keep reasonable track of structural life, and the damage tolerant nature of the design also works in our favor.

On the other hand, the "dynamic" components that make up helicopter rotor systems - main rotor shafts, rotor heads, control parts, are subjected to periodic loads during the the entire flight, and redundant design is not practical. Typical helicopter missions do not necessarily fit in to the Climb/Cruise/Descent pattern just described. A much more rigorous approach is therefore required, one that assesses the effect, not of a flight, but of each maneuver, since mechanically induced vibratory loading that changes with each maneuver is of dominant interest.

The nature of the flight test program for a helicopter is entirely different from fixed wing practice. It consists, for the most part, of structural load surveys.The results are correlated with structural analysis, and fatigue testing of parts in the mechanical test laboratory. Loads are measured in flight during every kind of maneuver likely to be encountered in service, and it is fair to say that most operational flights could be reconstructed from stringing together maneuvers flown in these surveys. This is, of course, the ultimate empirical approach, but it has served us well in the absence (up to now) of precise methods of analytically inferring vibratory loads in dynamic components.

Figure 3

All maneuvers performed by helicopters can be defined as flight regimes. Structural loads measured during flight test as described are related to regimes. Accordingly, if we can identify the regime, we can derive the corresponding loads.

Figure 4

This figure shows how analytical and ground test results are combined with flight test data, and are applied to a usage spectrum based on general and conservative considerations . For safety reasons this must be biased in favor of the most severe usage likely to be encountered but, in so doing, we penalize the individual helicopters that see a less severe usage. How can we improve the picture? One obvious improvement would be to measure the individual spectrum and compute the individual part life expended. Some safety factor on life calculated in this way must still be retained, but even so, a life much closer to reality, and in most cases an increased life, will be obtained. In a few cases a shorter than expected life may be found. This enhances safety, since the part would be replaced, and the service life reduced accordingly.

Figure 5

The advent of the microprocessor based flight data recorder will make it possible to collect individual aircraft data and accomplish real time processing. These data can be downloaded for second pass analysis later. This figure shows the different output that results when a recorder is substituted for the original usage spectrum, giving individual retirement times.

Figure 6

This illustration shows the equivalent process for Damage Tolerance Assessment (DTA). Since it is very likely that we will have both methodologies - Crack Propagation (DTA) and Crack Initiation (Safe Life) - actively in use together, the system must create a data base usable by either. We will now look at the arrangement of the proposed IHTP system.

Figure 7.

The input to the recorder consists of aircraft sensor data in analog form. These data are sampled and digitized before analysis by the regime recognition algorithm. This algorithm identifies the current regime once per second and causes a single memory location counter for that regime to be incremented. At the end of a flight, the array of counters represent a complete breakdown of that flight by the time spent in each regime. This technique enables data from any length of flight to be stored in a small array of counters. This technique simplifies the post flight processing to a level easily handled by an IBM or similar PC. In fact, the Air Force has selected the **Zenith Z 248** for Squadron use and IHTP data management would be only one of the tasks it could perform. Another task associated with this program is the manual input of inspections performed, and the results, whether or not they were cued by IHTP data.

Data from each Zenith will be transmitted, either voluntarily or on command, to the **Aircraft Retrieval System (AIRS)** VAX 11/780 mainframe computer situated at Warner

Robins. The entire MH-53J data base will be resident in this location. It will be accessible to the DTA Lab. VAX 11/785, which will execute the DTA Runstream Program. The resulting inspection interval and/or fatigue life data will then be transmitted back to AIRS, and back again to the Zenith units, as required. The ASIP Manager, Sikorsky Engineering Personnel, and other users of the data will be able to access the AIRS data by means of terminals.

Figure 8

This presents an idea of how data will be handled by the system. About 95 percent of the fleet will have usage spectrum monitors (USM), and 5 percent the L/ESS monitor.

Data from the USM will be periodically downloaded to the ground processor at the Squadrons and thence to the AIRS data base.

L/ESS data, after downloading, is processed into histogram format allowing it to be compared with the Sikorsky flight test data library. Should the L/ESS data not correlate acceptably, separate flight testing would be required to investigate, and obtain new data to update the library. However, if L/ESS and library data are in acceptable agreement, the data base for that particular flight pattern or mission is validated. In either case, from that point on, USM data takes over. The validation process is a combination of automatic data processing , hand reduction, and Engineering judgement. The AIRS data base accumulates USM statistics, and interfaces with the DTA Lab. mainframe which operates the Runstream program.

Figure 9.

This represents a typical summary plot for a measurement parameter containing all currently available edited flight test data to be entered in the Test Data Library. By edited, we mean that all data that is suspect for any reason (calibration or other problems) has been removed. The left hand side of the chart represents a summary of all data from accelerated flight regimes such as turns, pullouts, hovering maneuvers, control reversals, and transient regimes like climbs, or autorotation. Use of different plotting symbols groups data by such parameters as Gross Weight, C.G., or maneuver severity such as turn angle of bank or load factor. The right hand side contains data from steady unaccelerated flight. The line labelled E_t represents the endurance level, that is, the vibratory stress level above which the the part will accumulate fatigue damage giving it a service life below 10,000 flight hours. This life is the design target, and any part with a safe life below 10,000 hours is said to be" life limited". An equivalent line for DTA will exist, indicating the load level above which crack growth would occur.

Figure 10

The test data library consists of 120 such summaries, and selected ones, shown shaded and pulled out to the side, are the "Control" parameters which will be recorded in the L/ESS data.

Figures 11 &12

Typical histogram outputs are shown, the first of which is from L/ESS data.This will be used to determine load levels vs regimes for library comparison. The second, from USM data, will define the actual flight spectrum of the individual aircraft. Of course it will be possible to extract much additional operational information from the data, and a full programming effort to create a comprehensive AIRS data base, similar to that already in place for the C-141 and C-130, is planned.

Future Developments.

The design of the system will lend itself to further development. At some point, once the system just described enters service, various additions could include an incident recorder with a crash survivable module fed with selected data from the digital recorder, vibration monitoring, and routine maintenance data.

Force Management Requirements

Usage Spectrum Monitoring - USM

Account for flight time in terms of flight segments.
Apply to each individual aircraft in the fleet.

Loads and Environmental Spectra Survey - L/ESS

Conduct survey on selected sample of fleet.
Measure structural loads at key locations in airframe.

Fig 1

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Significant Differences

Fixed wing	Rotary Wing
Mission segments Take off Climb Cruise Descent Approach & Land	Individual Flight Maneuvers (Regimes) Liftoff Hover & Hovering maneuvers Take off Climb Turns Autorotation Control Reversals Approach to Hover Etcetera
Ground-air ground (GAG) cycles	Rotor Accel - Decel GAG cycle including rotor accel - decel
Airframe Structure	Dynamic components
Structural Loads (Gusts, Flight & Pressurization)	Vibratory loads
Static test article test results and analysis	Component fatigue tests, flight test analysis & Engineering judgement

Fig 2

UNITED TECHNOLOGIES SIKORSKY AIRCRAFT

Flight Regime Recognition

- Loads are related to flight maneuvers, or regimes, not flight segments.
- Flight test data is compiled by flight regime, and is stored in Test Data Library.
- Usage spectra are defined by regime.
- Relatively few measurement parameters are needed to recognize regimes.
- Example of a regime : Left turn 40-45 degrees angle of bank, 80-85 KIAS.

Fig 3

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CRACK INITIATION (SAFE LIFE) - PRESENT METHOD

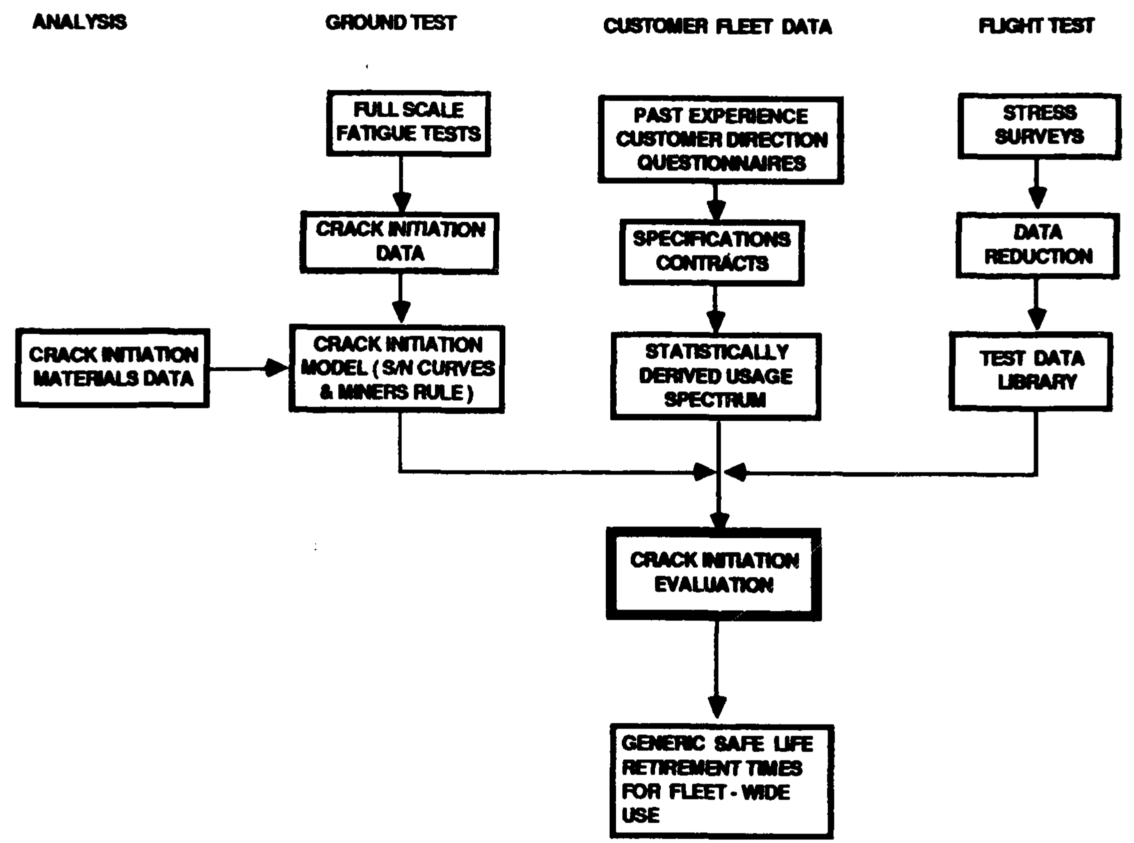

Fig 4

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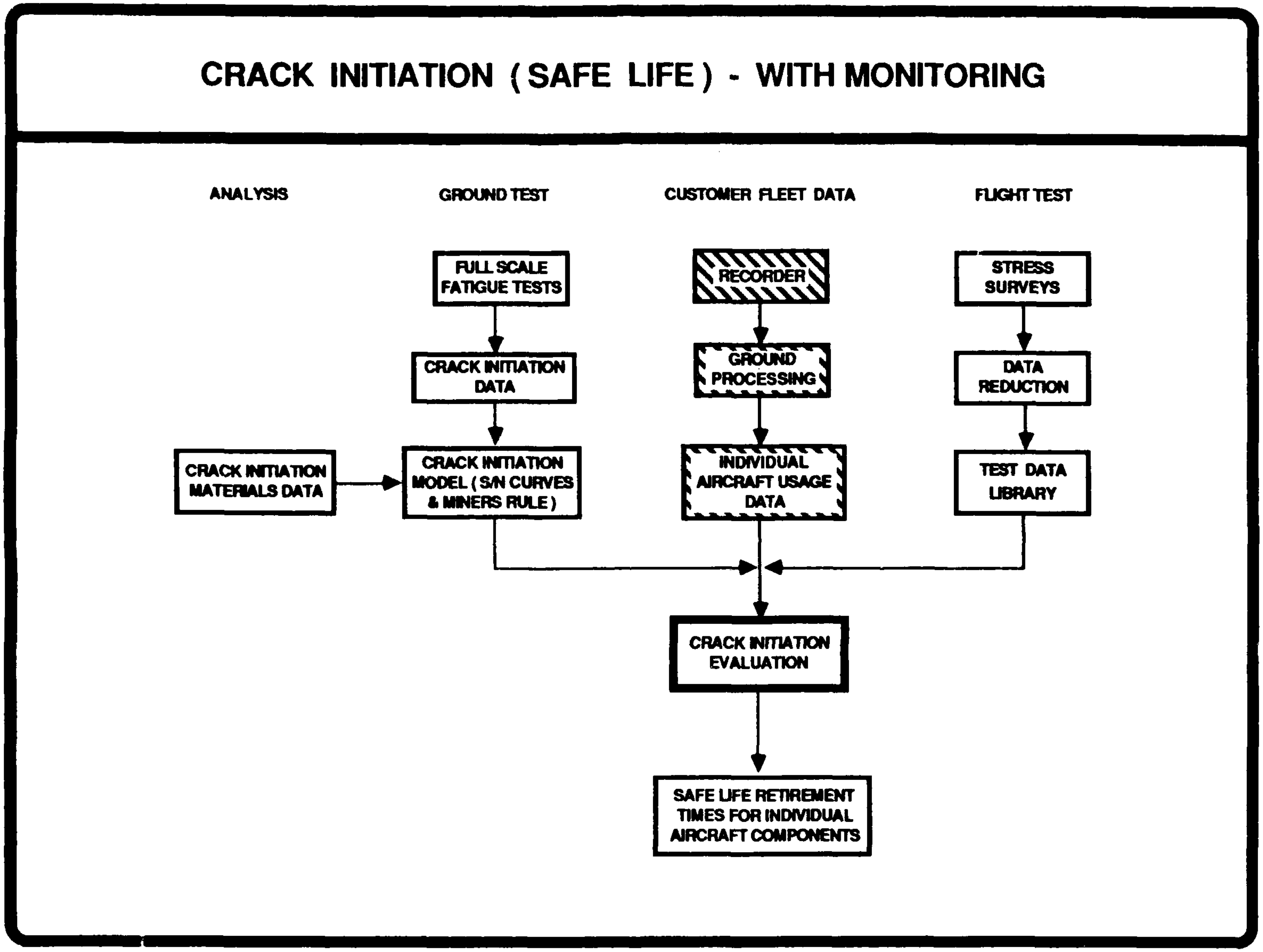

Fig 5

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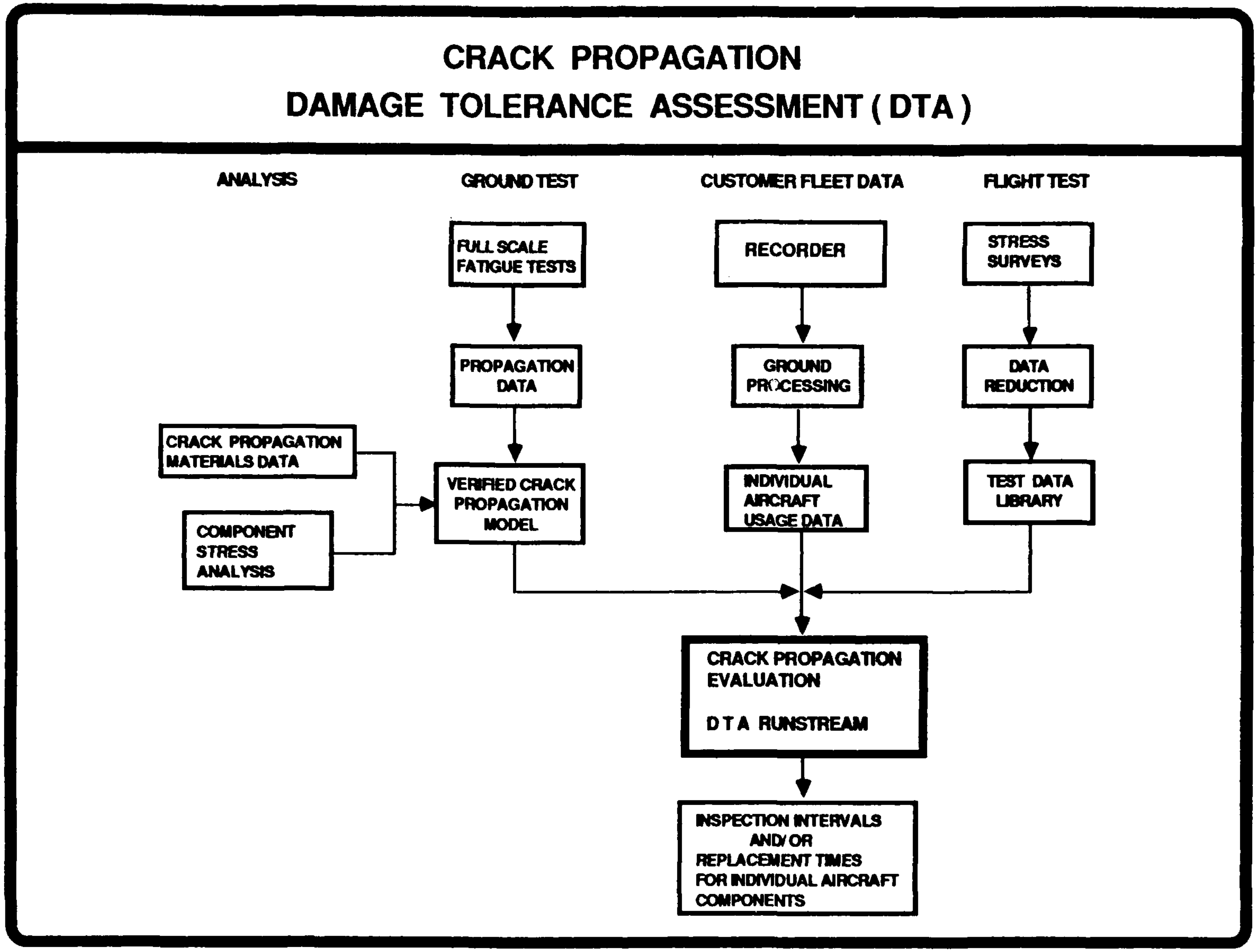

Fig 6

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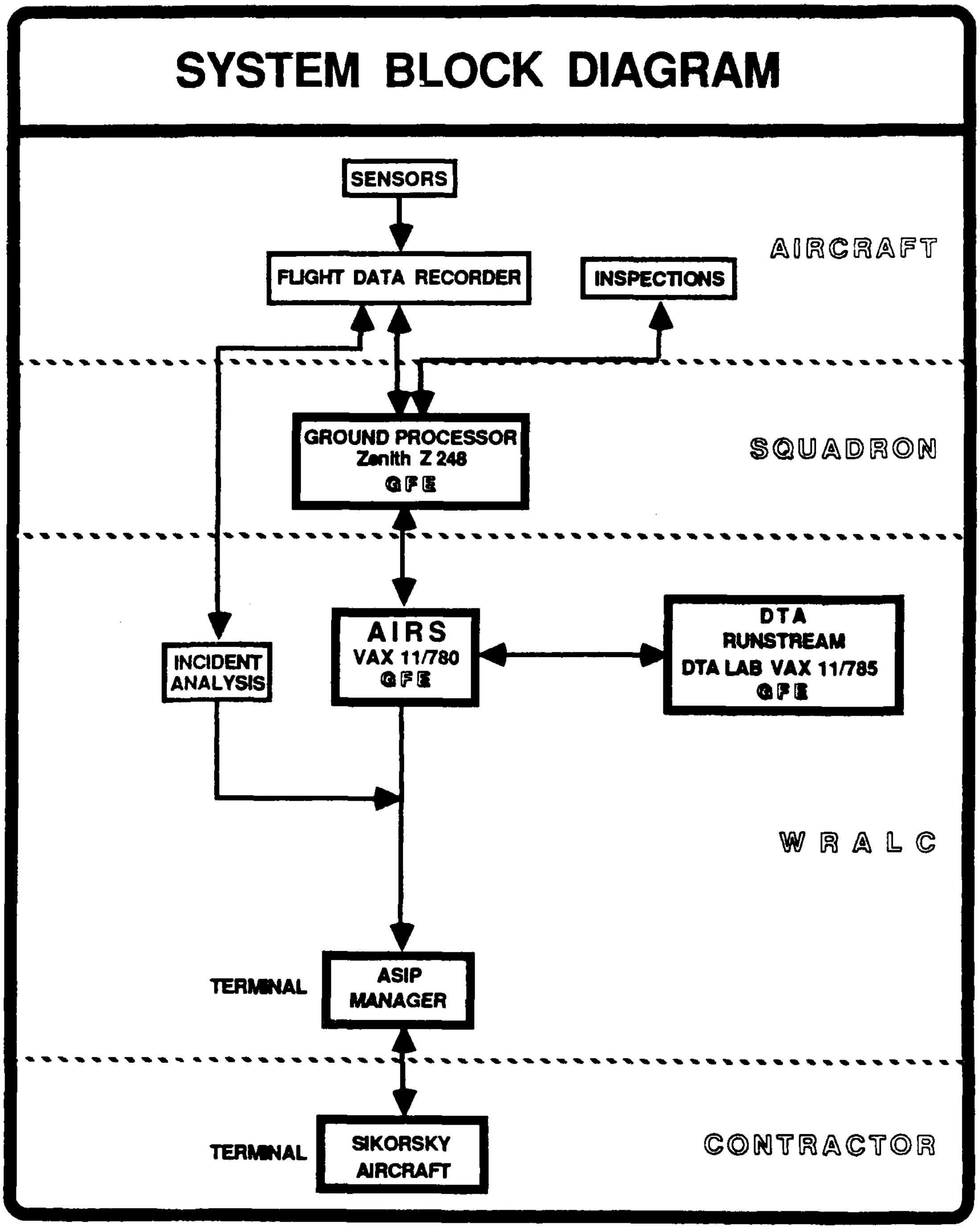

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Fig 7

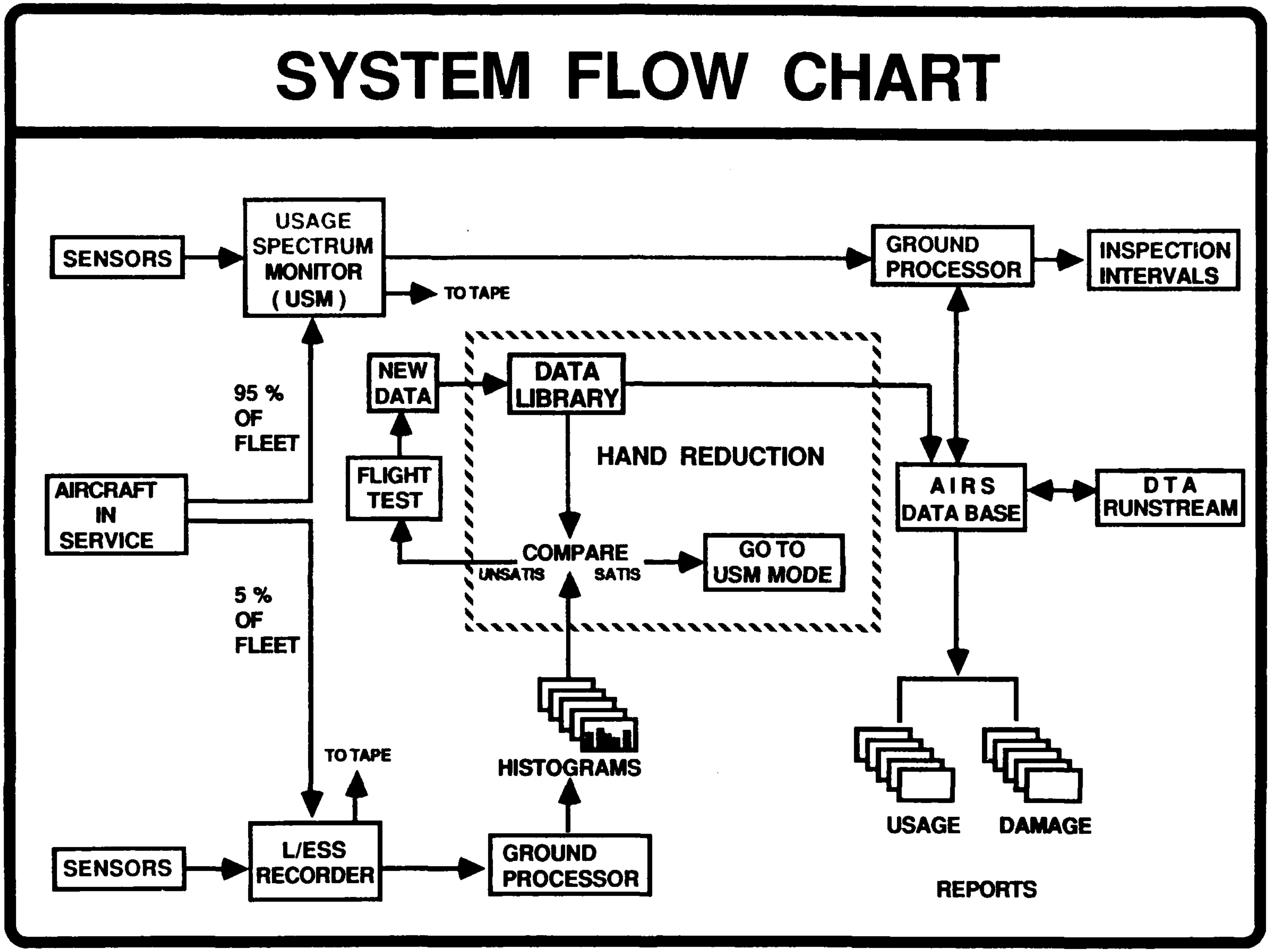

Fig 8

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Test Data Library Format

One Plot Per Measurement Location

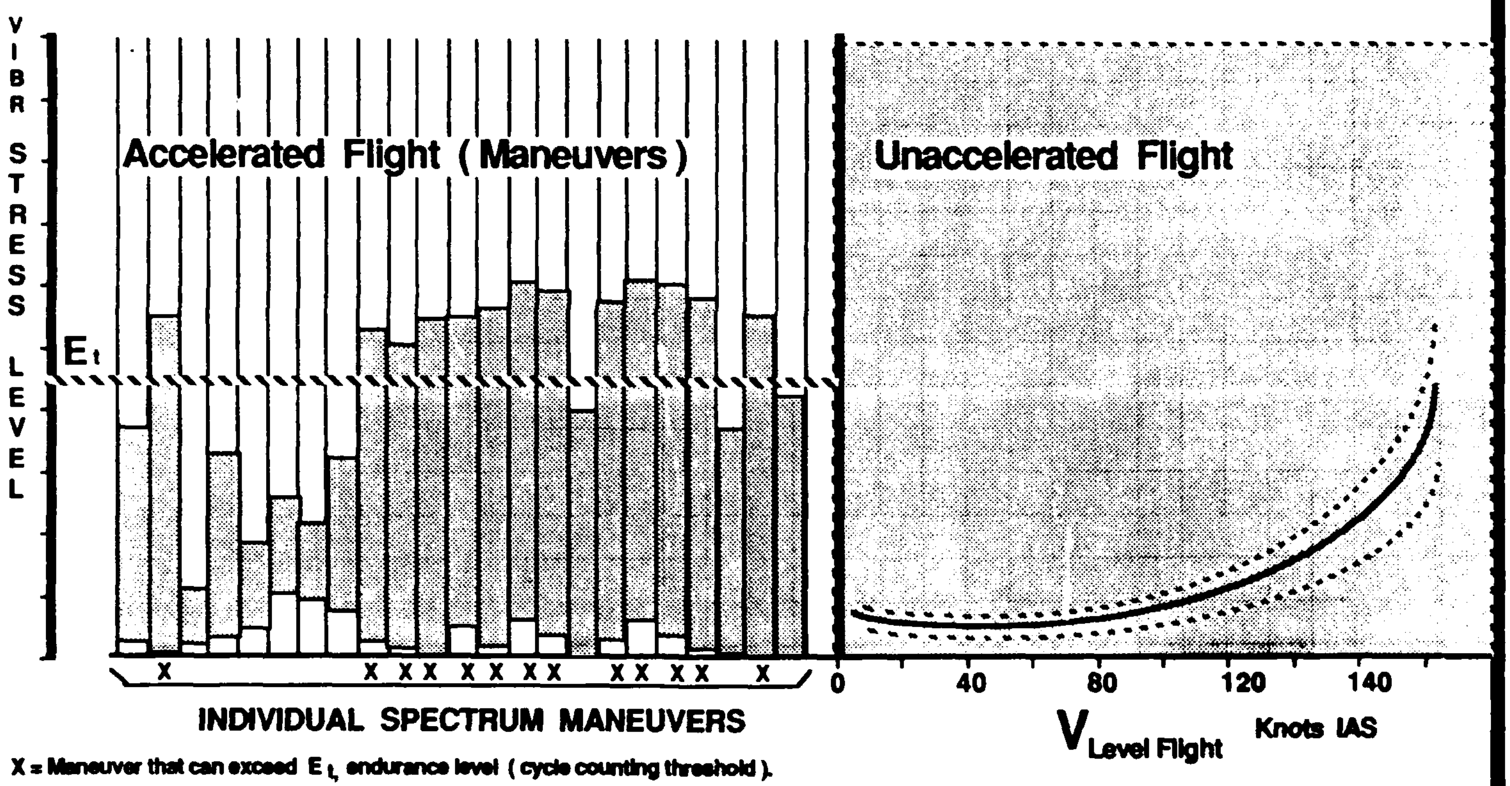

Fig 9

Computerized Test Data Library

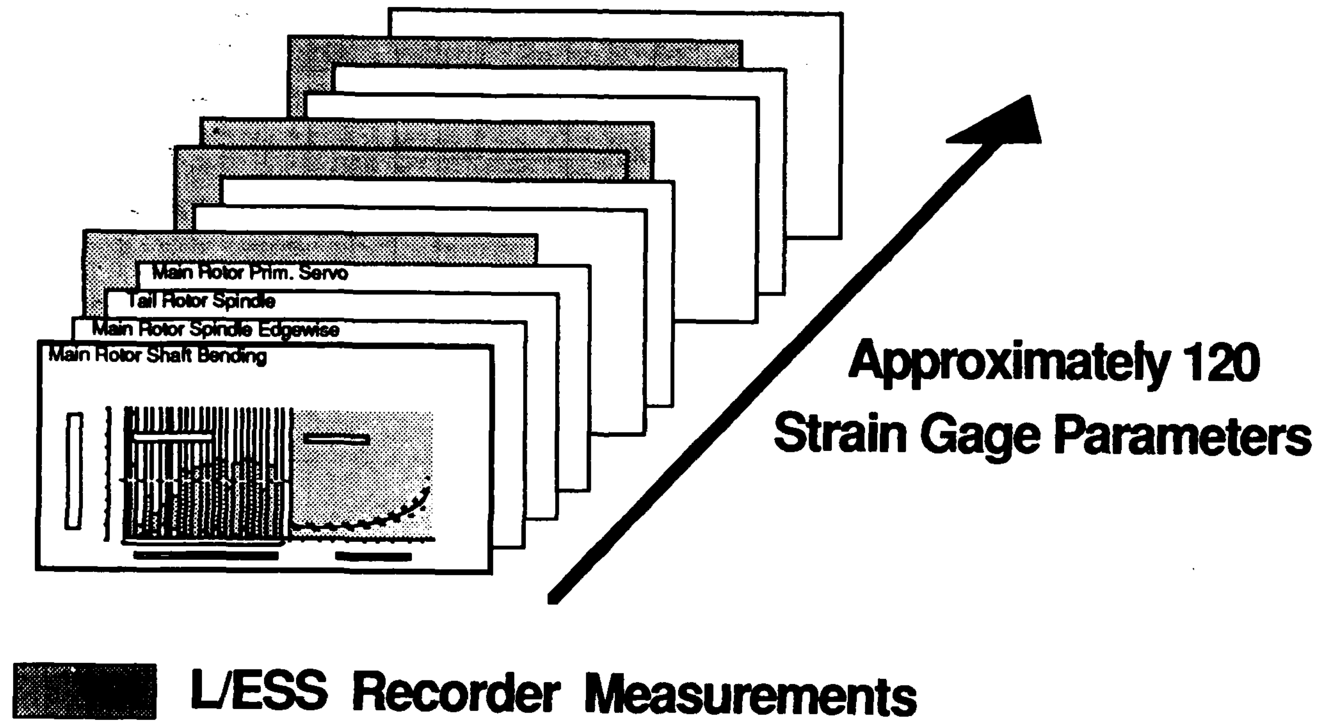

Fig 10

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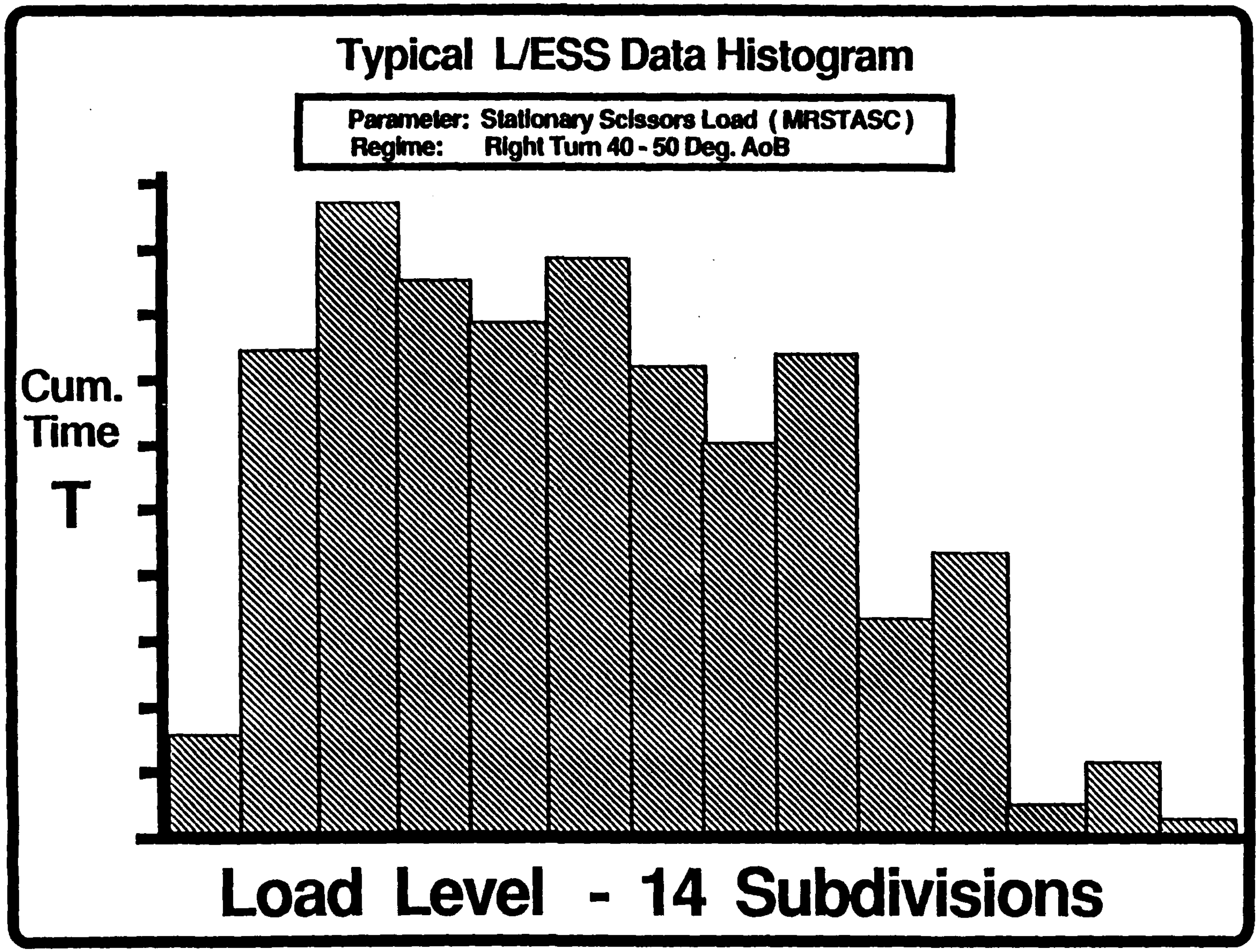

Fig 11

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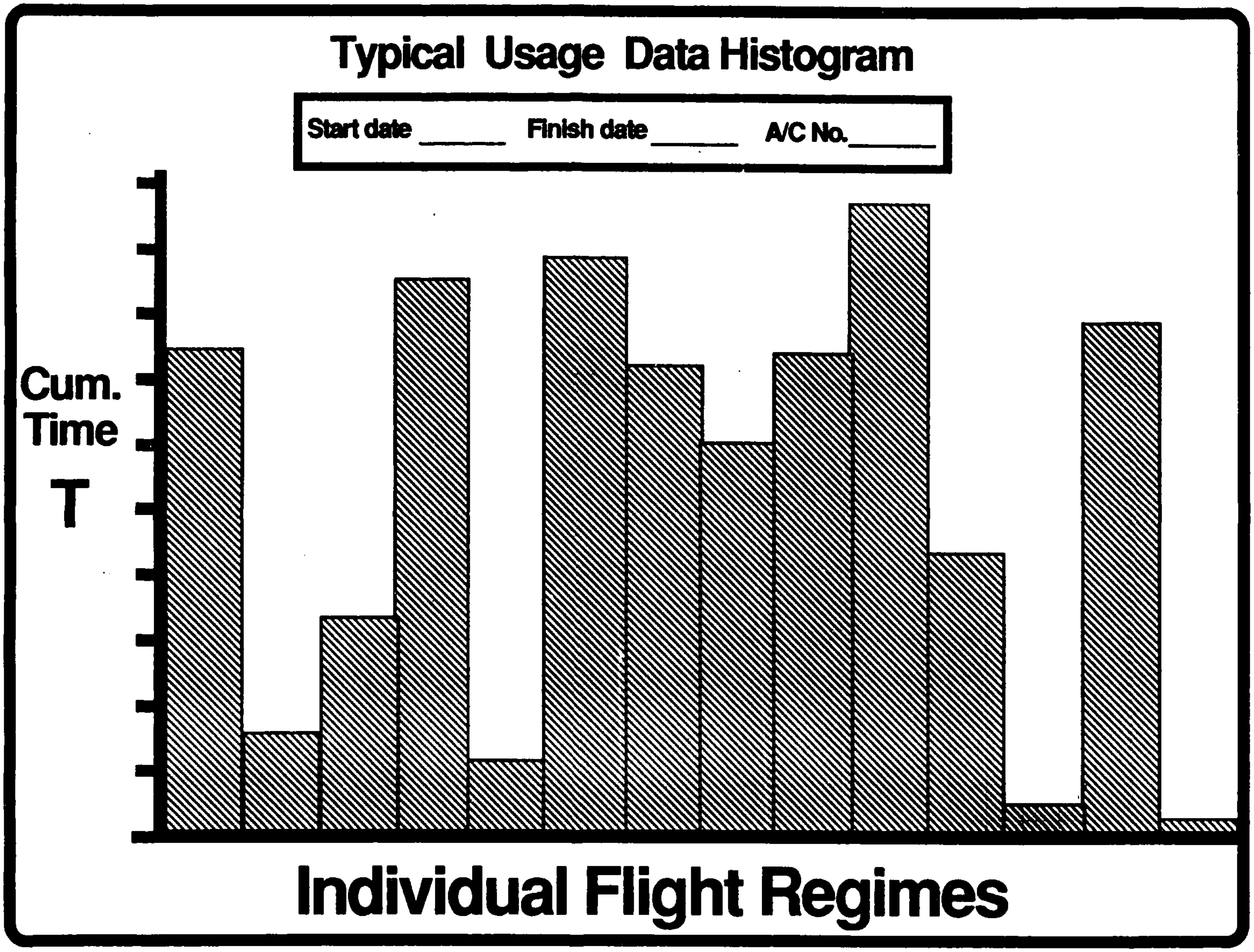

Typical Usage Data Histogram
Start date ______ Finish date ______ A/C No.______
Cum. Time T
Individual Flight Regimes

Fig 12

GARRETT TURBINE ENGINE COMPANY

F109-GA-100 ENGINE GYROSCOPIC QUALIFICATION TEST

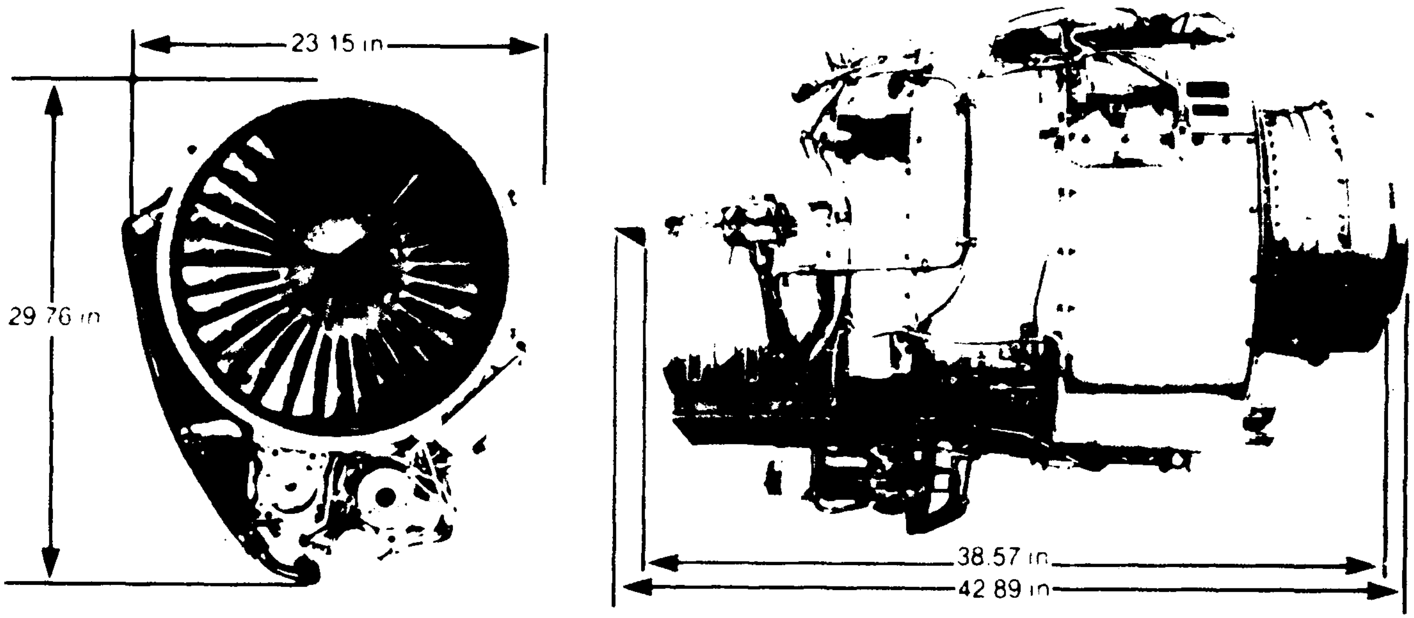
DECEMBER 1987

HANS MAERTINS
F109 PROJECT

Allied-Signal Aerospace Company

AlliedSignal

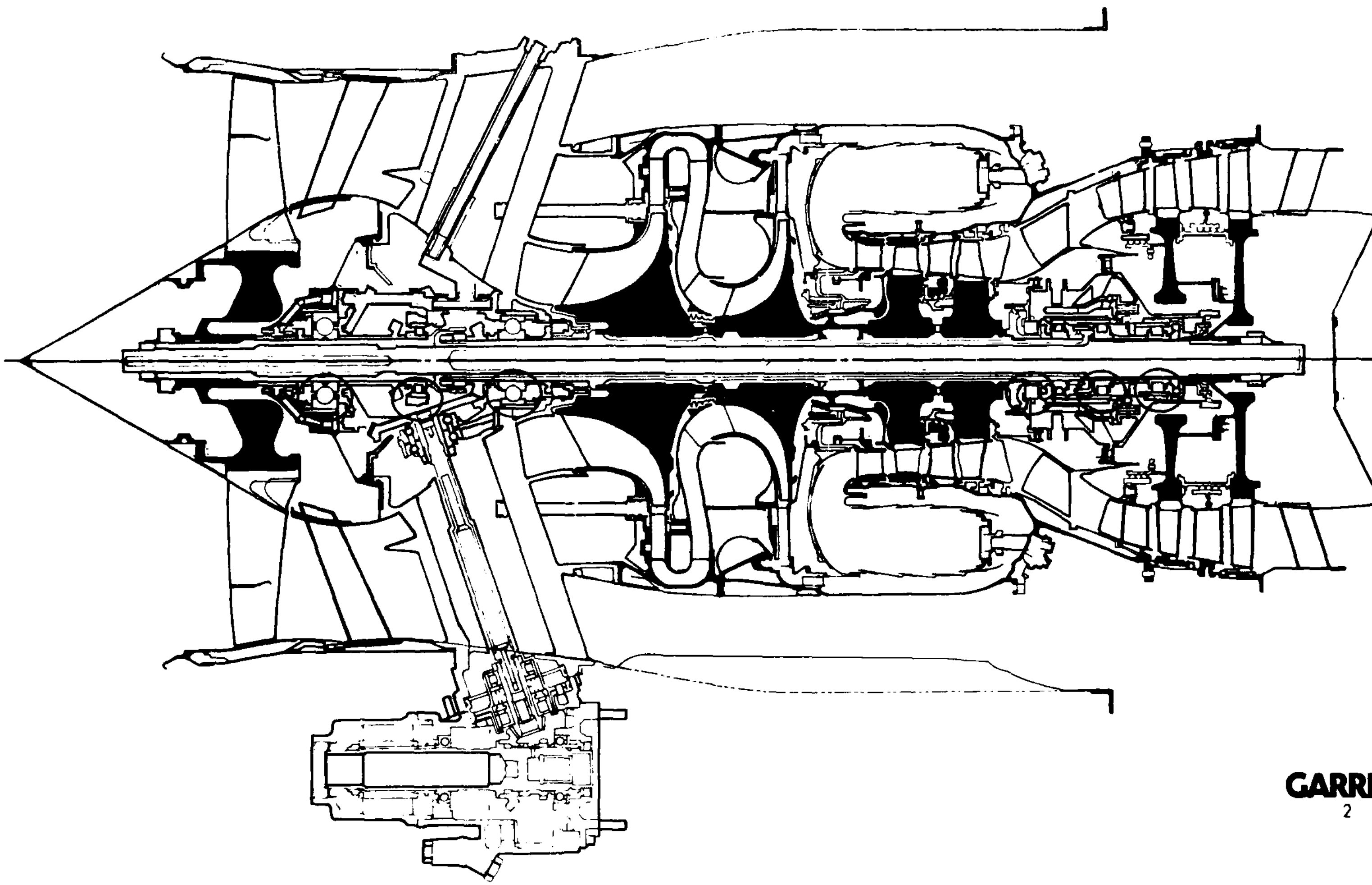
F109 TURBOFAN ENGINE



- THRUST - 1330 LBS SLS-ISA
- TFSC - 0.392 LBS/HR/LBS
- BYPASS RATIO - 5.0:1

GARRETT

F109 ENGINE DESIGNED FOR TRAINER 3.5 RAD/SEC MANEUVER



GARRETT
2

ENISIP REQUIRES RIGOROUS DESIGN VALIDATION AND VERIFICATION

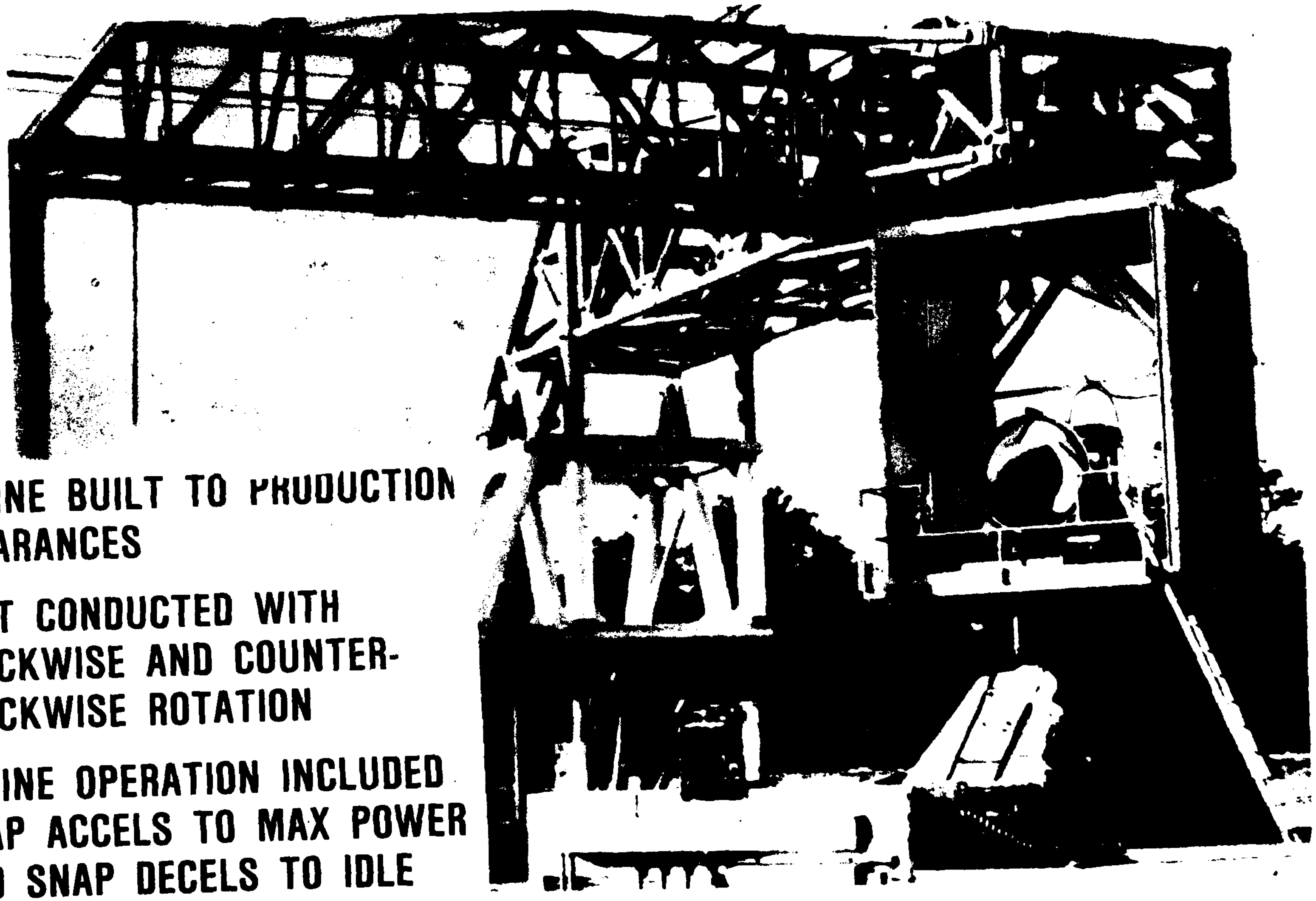
- GYROSCOPIC TEST DESIGNED TO VERIFY ENGINE OPERATION WITH MANEUVER LOADS

- QUALIFICATION TEST OBJECTIVES:

  - DEMONSTRATE SATISFACTORY ENGINE OPERATION TO A MAXIMUM OF 3.5 RAD/SEC
  - DEMONSTRATE 10^6 CYCLES ON N_L SPOOL AT 1.5 RAD/SEC

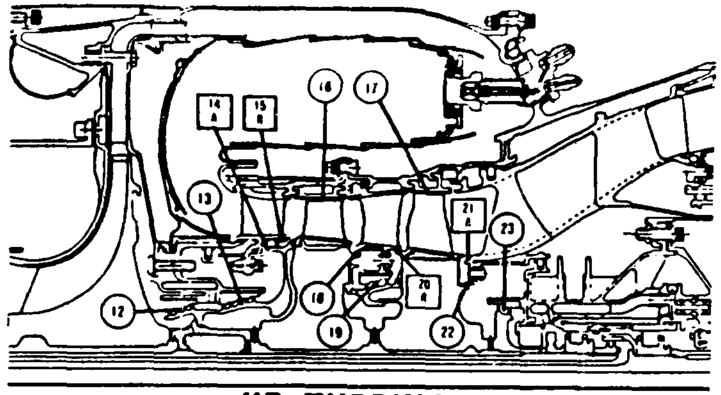
GARRETT
3

TESTING CONDUCTED AT NAPC FACILITY

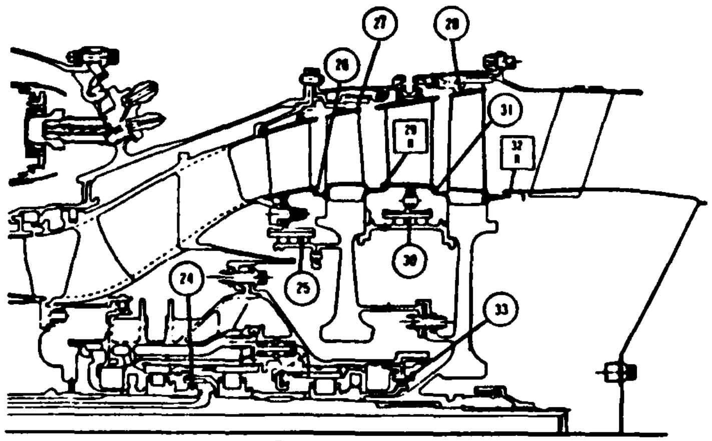


- ENGINE BUILT TO PRODUCTION CLEARANCES
- TEST CONDUCTED WITH CLOCKWISE AND COUNTER-CLOCKWISE ROTATION
- ENGINE OPERATION INCLUDED SNAP ACCELS TO MAX POWER AND SNAP DECELS TO IDLE

ENGINE BUILD INCORPORATED RUB DETECTION DEVICES TO MONITOR ROTOR DEFLECTIONS



HP TURBINE



LP TURBINE

- ALL AXIAL AND RADIAL CLEARANCES DOCUMENTATED BEFORE AND AFTER TEST
- RUB COATINGS AND GOLD RUB PINS INSTALLED TO RECORD ROTOR MOVEMENTS
- NORMAL TEST CONDITION (NON-GYRO) CLEARANCES DOCUMENTED PRIOR TO GYRO TEST

LEGEND:

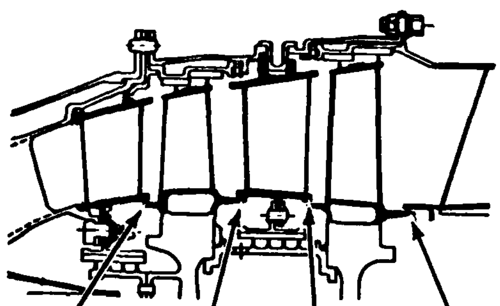
□ RUB PINS

○ DOCUMENTED CLEARANCE

GARRETT

5

GYROSCOPIC TESTING IDENTIFIED NEED FOR INCREASED LP TURBINE CLEARANCES PRIOR TO FLIGHT TEST



- INITIAL TESTING AT 2.0 RAD/SEC RESULTED IN LP TURBINE RIM SEAL RUB
- LP TURBINE CLEARANCES INCREASED TO ACCOMODATE 3.5 RAD/SEC ROTATION
- NO PERFORMANCE DETERIORATION NOTED
- ENGINE VIBRATIONS NORMAL

GARRETT
6

F109 IS FIRST ENGINE REQUIRED TO QUALIFY ON GYROSCOPIC RIG

- GYROSCOPIC TESTING PROVED TO BE EXCELLENT VALIDATION TOOL
- ENGINE OPERATION SUCCESSFULLY DEMONSTRATED TO 3.5 RAD/SEC
- 10^6 CYCLES (73 MIN.) SUCCESSFULLY DEMONSTRATED TO 1.5 RAD/SEC MAXIMUM CONTINUOUS POWER
- POST TEST INSPECTION REVEALED NO EXCESSIVE RUBS
- FINAL ENGINE PERFORMANCE AND VIBRATION CHARACTERISTICS REMAINED UNCHANGED

GARRETT
7

PHOTOELASTICITY - A COST EFFECTIVE DESIGN TOOL.

Jan Cernosek - PhotoStrain
formerly Stress-Strain Laboratories. Inc.
Dallas, Texas

Introduction.

We live in an era of technological crises. Our country is quickly losing the technical supremacy which we enjoyed just two decades ago. The indisputable indicator of this situation, an enormous trade deficit, persists no matter if dollar is up or down. The ultimate judge, the customer, turns down his thumb on products of our companies. Ross Perot, founder of Electronic Data System, wrote in the Washington Post of October 1987: "We are losing in international business competition. In 1986 we lost our position as the world leading exporter and we had a trade deficit in high-tech products, supposedly the base for future growth.

We blame the American worker unfairly for poor quality of our products. The unsatisfactory quality and appearance of many of our products is the result of poor design and engineering - not poor assembly".

We still design the best aircrafts in the world. But the warning signals are in the air. Twenty five years ago we manufactured the best cameras, motorcycles, TV sets, cars, etc. The list can go forever. Where are all these products? It does not make any sense to drive a car packed with electronic gimmicks but powered by a pitiful engine with specific output of only 30 to 38 hp/litre when competition is powered by engines with 50 to 70 hp/litre. No wonder the Japanese and German companies are grabbing a greater and greater portion of our market.

Just fifteen years ago we started the engineering revolution by introducing electronic calculators. Who is buying the US made calculators now? Are the big computers next to fall to the competition?

In order to reverse this trend, we have to go back to basic engineering. High-tech, no matter how important it is, will not save us. We need to spend more resources and efforts on design and redesign. The old proverb "if it is not broken do not fix it" does not apply any more.

Stress Analysis in design procedure - finite elements methods.

To make a design cycle more efficient we have to provide the designer with early feedback. The stress analysis should play the

prominent role in this task. It usually takes from six to twelve months to get a part from the drawing board to the final acceptance test. If the component fails the test, the designer faces a difficult task "to fix" the component within the given envelope because all other mating parts have been already designed and sometimes even manufactured. "Beefing up" the part is the most popular corrective action. The tight schedule usually prevents any attempt to make the part more efficient. The fear of repeated failure leads to "overkilling" which results in weight problems, so frequent in aerospace.

Thus, the accurate and reliable stress analysis of the component, while it exists only in "blueprint" form, should be of utmost importance. The traditional P/A and Mc/I approaches which bring the inevitable "safety coefficients" are no longer acceptable in the aircraft design. Numerical methods (especially finite element methods) whose development was closely related to the affordability of digital computers,were thought to provide a "push buttom" engineering analysis. After years of using them, the original exitement was replaced by more somber assessment of capabilities of mathematical modeling. The analyst has learned, sometimes the hard way, that the finite element methods emphasize the role of engineering judgment rather than diminish it. The numerical solutions depend not only on boundary conditions, loading conditions, size of mesh, and type of elements but also on the inner architecture of the numerical code itself.

Floyd (1) tried to compare the commercially available numerical codes in a "round robin" problem involving a relatively simple pressure vessel. He concluded that the application of the finite element method, if not compared to some other non numerical approach, could lead to a surprisingly large errors.

Sometimes the behavior of mathematical models defies logic. Fig. 1 depicts a simple axisymetric contact problem. In order to model a proper contact pressure distribution, the "gap" elements were introduced between various portions of the structure. The solution (using MSC NASTRAN numerical code) converged despite the erroneous definition of stiffnesses of "gap" elements but the displacements of various portions of the structure were comparable to the distance from the earth to the moon. When the error was corrected, the solution diverged until the specific (10 lb) preload was introduced into the "gap" elements. Why the solution converged for the definite preload and diverged for all other preloads was a mystery even to the "fathers" of this numerical code.

It has to be pointed out that there are many problems which have been very successfully solved with finite element methods. For every "horror" story one can find one or more "success" stories. The finite element methods seem to work very well for the analysis of airframe structures where typical structural elements as beams, rods and thin shells are used. But their reliability

is not so good when their extension into the analysis of the continuum is considered. The performance of "solid" elements is questionable.

Photoelasticity.

One method which is capable of helping designers is three-dimensional stress-freezing photoelasticity. This is not a new method. Principles of photoelasticity were established in the last century when British physicist C. Maxwell observed the phenomenon of birefringence induced by mechanical stress in glass and formulated the constitutive equations. The first engineering application can be dated to 1913. The major breakthrough in the industrial application of this method came in 1936 when scientists working at the University of Munich in Germany discovered a "stress-freezing" phenomenon in polymers.

In stress freezing technique the model of the component, made from special optically sensitive material, is loaded at the elevated, so called glass transition temperature and cooled, while under load, to room temperature. The optical anisotropy (birefringence) induced into the model by mechanical stress remains unchanged even if the load is removed and the model is sliced into thin slices which are examined in polarized light. This discovery enabled the extension of photoelasticity into the analysis of a three dimensional state of stress.

The stress-frozen model contains the information about stress in every point of a geometrical replica of an actual structural component. The information is "stored" in the stress-frozen model and is readily available for decoding. The way how it is done depends solely on the goal of the study. This is what distinguishes photoelasticity from all other methods of an experimental stress analysis. It can be compared to finite element methods but it has the advantage of boundary condition being established by physical laws rather than by judgment of a stress analyst.

In spite of this exceptional power of photoelasticity, its usage was limited to the solution of problems of the basic research (determination of coefficients of stress concentration, for example) because its procedure was too time consuming and, therefore, expensive. Models had to be machined from precured blocks of expensive material whose machinability was very poor. Thus, the industrial applications were limited to project which could absorb cost of this analysis, as for example projects associated with the nuclear power industry.

A considerable effort was made during several past years to transform stress-freezing three-dimensional photoelasticity into a responsive design tool. Its methodology was completely reevaluated with time being a decisive factor.

The major breakthrough was a development of new fast curing model material which enables preparing models in the extremely short time. This material also exhibits an excellent machinability, comparable to that of aluminum. The new mold making technique was also developed. Both these accomplishments made casting "on shape" possible. The expensive and time consuming machining of photoelastic models could be eliminated or, at least, very limited.

The process of three-dimensional photoelasticity, as being practiced by PhotoStrain, is quite straightforward:

The model is fabricated by casting "on shape". The mold for casting the model is prepared from the part itself or from the pattern built according to blueprint. In order to minimize the cost of fabricating the pattern, the polymethylacrylimide foam is used. This foam can be machined in very close tolerancies. Templates can be used as cutters. The cutting forces are so small that the double sensitive tape can hold this material securely to the table of the milling machine. The various parts of the pattern can be bonded together using the quick-setting epoxy. Fillets are made by wiping-on clay using tools fabricated from rubber rods with machined spherical heads of required radii. The surfaces of the pattern fabricated from the foam are filled with wipe-on/wipe-off filler especially developed for this purpose. Figs. 2a to 2d illustrate the fabrication of the pattern of the component of the helicopter control system. This pattern was prepared in less than eight hours.

The special molds to cast photoelastic models are prepared from these patterns.

The general requirements of these molds are as follows:

a) The inner surface of the mold has to have excellent releasing properties.
b) The mold has to be designed in such a way that the shrinkage of model material during polymerization will not introduce cracking or residual stress.
c) The mold has to be rigid even at elevated temperature in order to withstand hydrostatic pressure of the material without substantial deformation.

PhotoStrain developed a unique methodology of preparing these molds. The mold is composed of two layers: an elastic cushion and a rigid shell. The inner elastic cushion is formed from especially formulated silicon elastomer. The outside rigid shell

is formed from high-temperature resistant epoxy resin. This shell provides the mold with geometrical stability while an elastical cushion protects the casting. The mold making procedure is very simple. A coat of silicon elastomer is brushed on the surface of the pattern with a stiff brush. When silicon is cured into a solid coat exhibiting an aggressive tack, a layer of epoxy is brushed onto silicon. Both these layers fuse together during the curing of the mold.

PhotoStrain developed a unique fast curing, epoxy based material for casting "on shape". This is the only available model material which can be cast in large quantities into molds without developing residual stress on "as cast" surfaces. Despite that it is cured by low reactivity end exhibits negligible exothermicity, it cures overnight and exhibits excellent machinability. Thus, the model can be cast one day and demolded and machined (if necessary) the following day.

In spite of the excellent machinability of this material which can be even broached, the machining is usually limited to contact surfaces. Holes are also rather drilled than precast.

The finished model is then adjusted into a loading fixture which enables the simulation of loading which is experienced by the actual component. The actual loads are scaled down. A typical scaling factor is in 500 to 1000 range. In most cases, the loads induced by dead weights are adequate. There is no need for a sophisticated hydraulic loading system - the fact which also helps to keep the cost of the analysis down.

The stress freezing cycle is rather simple. The model is heated to the elevated, glass transition temperature (270°F for this model material), loaded and cooled slowly to room temperature. At glass transition temperature the model material experiences a sudden change in mechanical properties. The material's thermodynamic conformance changes from glass-like to rubber-like. The modulus of elasticity changes from room temperature magnitude of 400,000 psi to a rubber-like magnitude of 3000 psi. The Poisson's ratio also changes from 0.36 to 0.5.

The cooling gradient must be small enough not to introduce thermal stress which would superimpose onto stress induced by loads.

When room temperature is reached, the load can be removed. The deformation and the birefringence remain "frozen" in the model. The deformation can be easily measured if the stiffness of the component is also of some interest. This deformation is "scaled up" because of the low modulus of elasticity of the model material at the glass transition temperature.

The birefringence which also remains "frozen" in the model is not disturbed when the model is cut into thin slices. The slicing is done on a special high-speed band-saw cooled with compressed air.

The slices are examined in a polariscope. The observed fringe pattern is directly related to the mechanical stress. The normalized fringe order (fringe order devided by the thickness of a slice) when multiplied by a constant which characterizes the optical sensitivity of the model material is numerically equaled to the stress in the model. The prototype stress is determined by multiplying the model stress by the load scale factor. In typical study, up to one hundred slices are cut from the stress-frozen model and stress is measured in up to six hundred points.

Figs. 3a to 3e follow the study of the stress distribution in a typical component of a helicopter control system. In this study, the mold (Fig. 3a) was prepared from the actual component. The part was removed from the mold after its curing was completed. The part, which was absolutely undamaged after demolding (curing temperature of the mold is only 150°F), could be used for other tests (for example a fatique test) or even used as a flying article. After removing the part, the mold was again assembled using irregularities of parting lines as the guides (Fig 3b). There is no need for locks or pins to guide various parts of the mold. Some machining was done on the model. Because of excellent machinability of the model material, there is no need for complicated machining fixtures (Fig 3c). Fig. 3d depicts the component in a stress-freezing oven after completition of the stress-freezing cycle. Note a scaled up deformation of the component. Fig. 3e shows two models of this component - one "stress-frozen" and the second in the undeformed state.

Twenty four slices were cut out from the model and stress was measured in four hundred fifty points. The cost per data point was $8.52.

The chronology of the study was as follows:

Working day	Activity
1	Part received from customer
2	Fabrication of mold
3	Fabrication of mold
4	Casting of model
5	Demolding, machining of model Fabrication of loading fixture
6	Stress-freezing
7	Stress-freezing
8	Stress-freezing
9	Slicing, preparation of slices for measurement
10	Measurement
11	Measurement
R E S U L T S A V A I L A B L E	
12	Report

13	Report
14	Report
15	Report
16	Report

Figs. 4 to 6 demonstrate complexities of the parts which were sucessfully and cost-effectively analyzed by three-dimensional photoelasticity.

Conclusion.

The great error of the past years was labeling a finite element method and three-dimensional photoelasticity as being two competing methods. Where finite element method works exceptionally well (thin walled airframe structures - for example), photoelasticity is nearly helpless but it excells in the area where FE has to use solid modeling which is time consuming, expensive and of questionable accuracy.

Photoelasticity and finite element method should be considered to be complementary rather than competing methods of stress analysis.

References:

Floyd C. G.. The Determination Of Stresses Using A Combined Theoretical And Experimental Analysis Approach. (Proceedings of the Second *Conference on* "Computational Methods and Experimental Methods," July 1984.

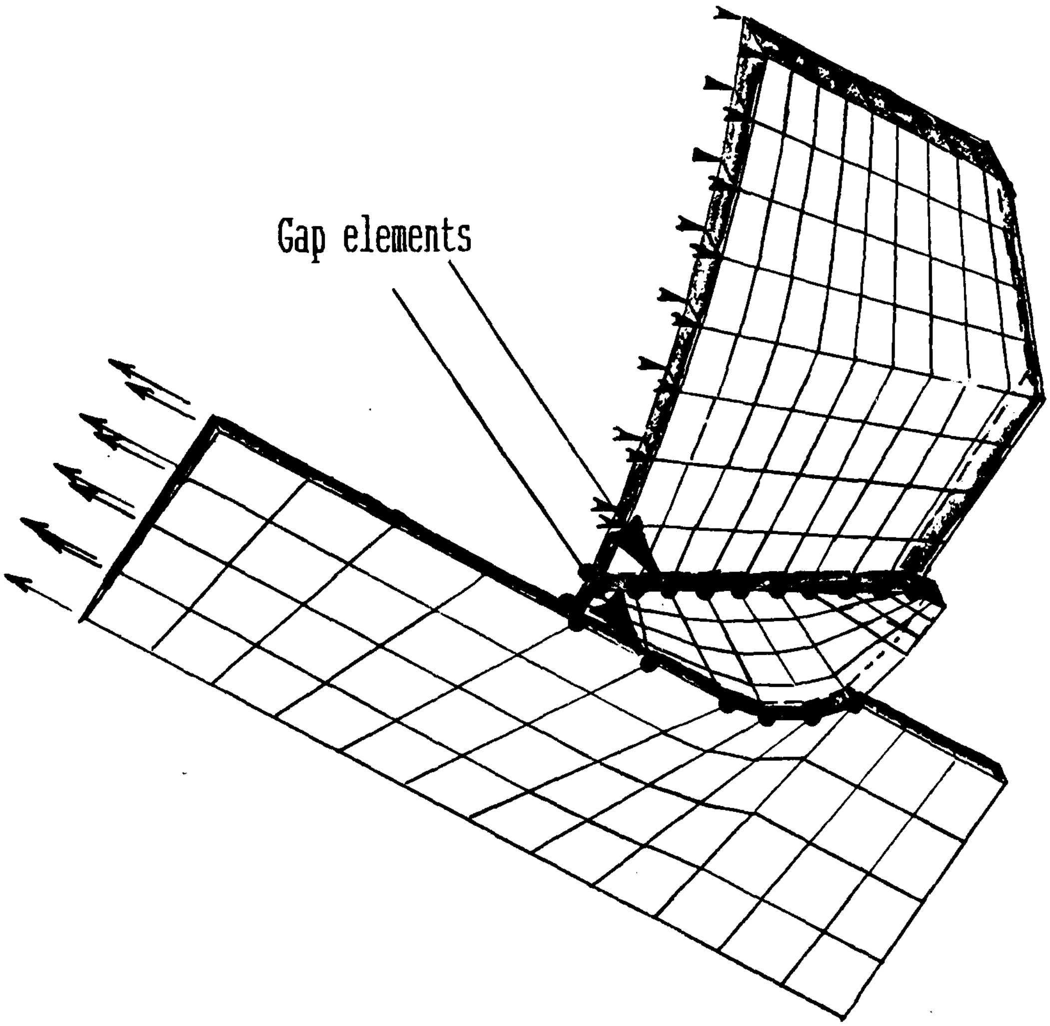


Fig. 1 - Finite element model of the spindle, split-ring and thrust bearing assembly.

Fig. 2a - Fabrication of the pattern of the component of helicopter control system. Elapsed time. 0.5 hour.

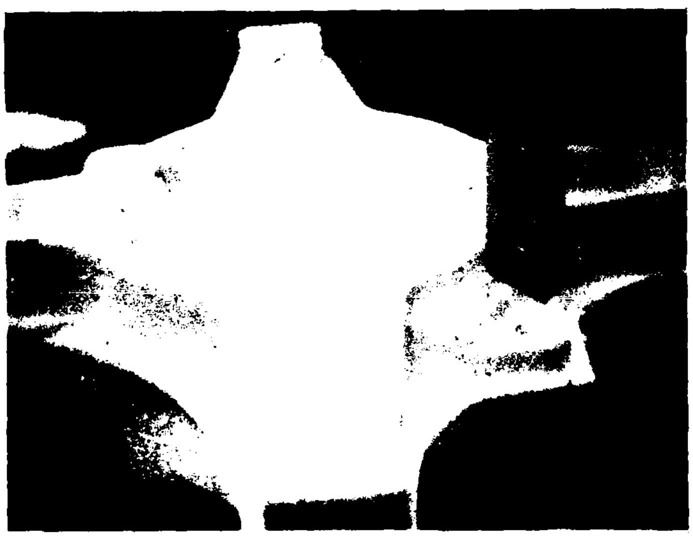


Fig. 2b - Fabrication of the pattern. Elapsed time. 3 hours.

Fig. 2c - Fabrication of the pattern. Elapsed time. 4.5 hours.

Fig. 2d - Fabrication of the pattern. Elapsed time. 7.5 hours.

Fig. 3a - Component of helicopter control system Mold for casting photoelastic model

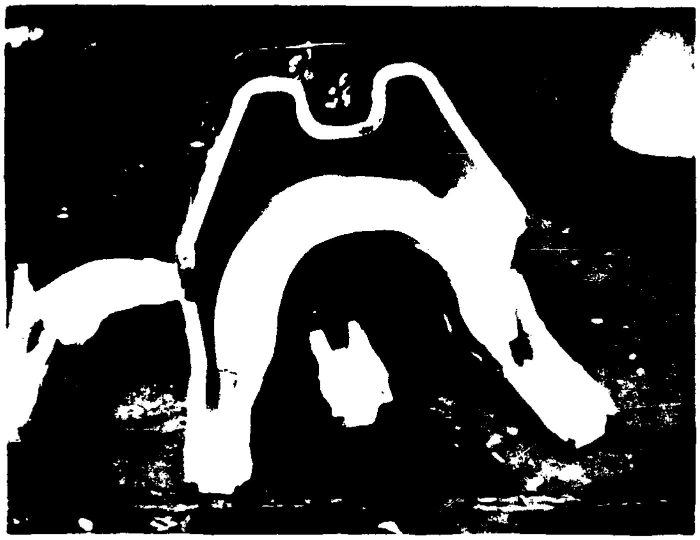


Fig. 3b - Photoelastic mold. Assembly procedure.

Fig. 3c - Machining of photoelastic model.

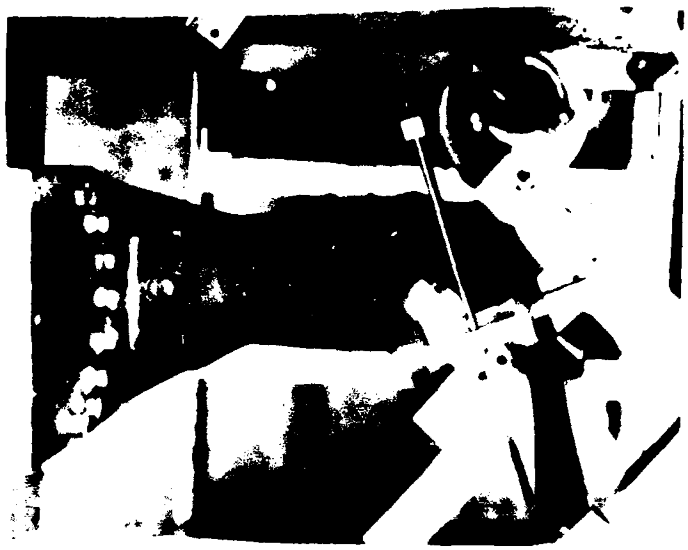


Fig. 3d - Photoelastic model in the "stress-freezing" oven.

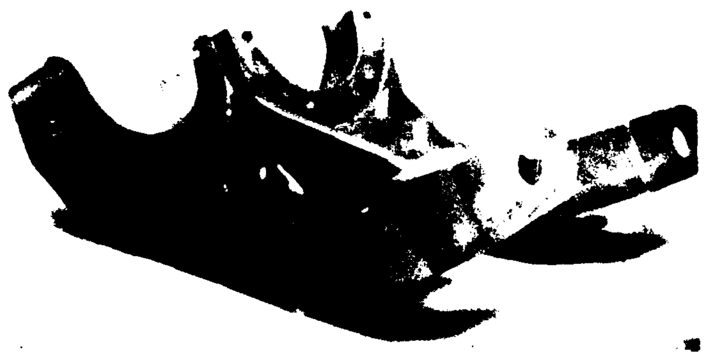


Fig. 3e - "Stress-frozen" and undeformed photoelastic models.

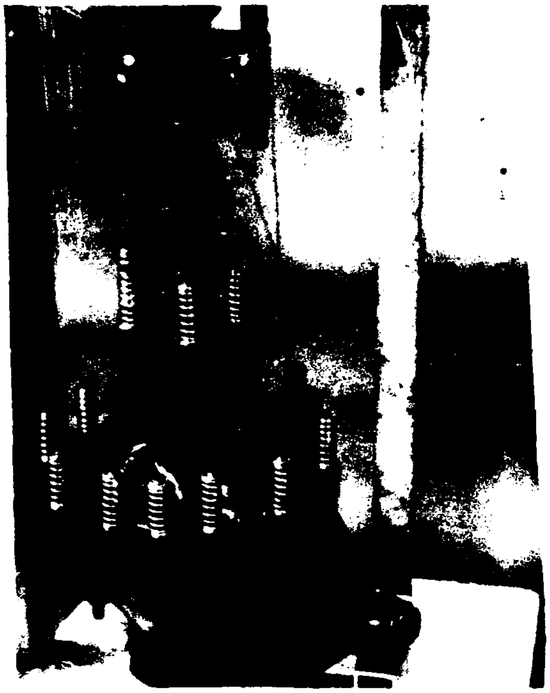


Fig. 4a - Photoelastic model of the front differential carrier of four wheel drive truck.

Fig. 4b - Disassembled model of the front differential carrier.

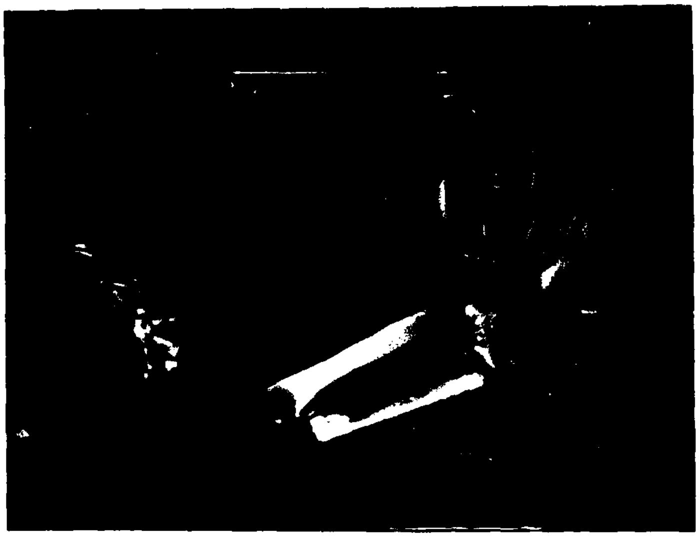


Fig. 5 - Photoelastic model of the helicopter pitch-horn.

Fig. 6 - Photoelastic model of the helicopter transmission.

FINITE ELEMENT MODELS OF USAF AIRCRAFT STRUCTURES

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INTRODUCTION

Finite element analysis is at present an industry standard for the analysis and design of aerospace structures. Significant resources are expected to be allocated for developing, testing and validating finite element models (FEMS) of current and the future USAF aircraft. The premise is that the effective use of finite element analysis can reduce (not eliminate) dependence on test procedures which are very costly both in time and resources. However, there is a lot to be desired from the way industry and Government organize and perform finite element analysis. The major deficiency is the lack of a clear definition of the analysis objectives and tailoring the models to achieve the goals in a most economical and reliable way. An even more disturbing fact is that, at present, industry developed finite element models are a wasted effort as far as future utility is concerned, because the Air Force does not take delivery of these models in an organized, predictable way. Hence, various potential users of FEMS throughout USAF do not know if and where the FEMS exist and how to gain access and use them. The result can be a duplication of developments and an unnecessary cost to the USAF. This paper explores various cost-effective ways of taking delivery of finite element model data and establishing procedures for archiving, communicating, retrieving and validating in a secure environment. It also

delineates the cost and performance benefits that can be derived during the life of an aircraft by maintaining an accurate and readily available library of finite element models of USAF aircraft.

FINITE ELEMENT MODELS AND THEIR PURPOSE

Aircraft structures are generally built up of many structural elements such as panels, beams and joints. They are highly articulated and consist of a complex arrangement of spars, ribs, skins, spar caps, rib caps, stiffeners and longerons (see figures 1-3). Before the advent of finite element analysis aircraft designers made gross approximations, such as, representing lifting surfaces by equivalent beams or plates and the fuselage by beams. A rod and shear panel representation in the context of a multi-cell box beam is the most sophistication that was available before the era of general purpose finite element codes like NASTRAN. In fact even the finite element models of the F-15 and the B-1 (circa 1970s) are made of simple shear panels and rods.

Aircraft structures are too complex or cluttered (see fig. 1) to be represented by single continuum models. These simple models do not provide enough accuracy and detailed strength and stiffness information necessary to design modern aircraft where the performance and weight requirements are extremely stringent. The behavior of the plates, beams,and rods from which aircraft structures are constructed is governed by one or more differential equations, and they can be solved with strict assumptions of continuity and complex boundary conditions. However, when they all come together, with their differential equations, at the joints it is impossible to establish compatability and make a meaningful analysis. Finite element analysis, on the other hand, allows modelling these discrete structures by approximating the differential equations by algebraic equations which do not normally require continuity and compatability beyond the first level. Also, it is easy to

represent complex boundary conditions in simple terms in a finite element analysis. An even more important consideration is that the algebraic equations can be solved very efficiently on modern digital computers. In response to this facility and flexibility numerous public domain and commercial finite element analysis codes were developed during the 60s and 70s. They are used extensively for the analysis of aerospace, mechanical, civil and marine structures. A partial list of frequently used finite element codes are: NASTRAN, ANSYS, ABACUS, ADINA and MARC. Emphasis in the 1980s is on the development of multi-disciplinary preliminary design programs such as ASTROS. They are also based on finite element analysis. In addition, they will have extensive optimization capability. When these systems are fully operational, they can really bring the impact of modern super computers to the design office in an unprecedented way in order to improve the performance at a minimum cost.

The purpose of a finite element analysis is to determine the performance characteristics of aerospace structures. The strength, stiffness, and static and dynamic aeroelastic properties can be estimated quite accurately by analysis with finite element models. When the physics of the problem is well defined by appropriate elements,boundary conditions (geometry) and loading conditions (flight environment), a finite element analysis can be very reliable and cost effective. The cost of testing can be significantly reduced by promoting quality analysis. This approach is particularly appropriate now because of the rapid developments in super computers and the reduction in computational costs. If all the benefits, such as, shorter schedules, number of parametric studies and potential for technology transfer are added up, there is no question that the finite element analysis is an indispensable tool in the competitive product development. Preliminary design systems like ASTROS, FASTOP, ASOP and OPTSTAT are all compatible with analysis systems such as NASTRAN. They provide challenging opportunities for performance improvements and weight and cost reductions in the future.

FINITE ELEMENT MODELS OF USAF AIRCRAFT

There are many airplanes in the Air Forces' current inventory. It is a tribute to the Air Force that once it develops a successful system it uses, reuses and reuses until it just about falls apart. Examples are the old faithfuls: B-52, KC-135, F-4, and C-130. Similar usage is expected from more recent systems such as: F-15, F-16, F-111, B-1, C-5A, C-17, A-10 and others. With all current emphasis on life-cycle costs, repair and maintainability considerations, future systems like the ATF and other unmentionable systems are expected to be used even more intensely. These are marvellous but extremely complex engineering systems. Yet the Air Force, at present, does not have an orderly and coherent way of receiving and archiving technical data for lessons learned from the experiences of the past. Industry develops the finite element models and the Air Force pays for them. However, due to the lack of standards and planning it does not require the contractors to deliver the data for the models with the system. Without this important technical data it is like exploring a blind alley when the time comes to adapt the systems to new missions, add new weapons, develop new derivatives or simply evaluate new repair and maintenance procedures.

The need for this technical data is becoming so acute that there is not a single month in which one Air Force center or the other is not actively looking for the data and in the process wasting countless hours and resources. There were instances where the contractors were willing (charitable enough) to give a 3 to 4 inch thick paper listing of the finite element data but refused to give the data on a computer tape simply because it was not a CDRL item. In the first place the Air Force paid (probably more than once) for the development of this data, and it would probably cost less than $50 to copy it on a computer tape and give it to its rightful owner. There are other instances where four different Air Force organizations paid for the same airplane data with minor differences. The point is that

this data is invaluable, and we are becoming smart enough to realize it. This is the time to think, organize and develop effective standards and convince the system program offices (SPOS) to take delivery of the data with the system. Development of finite element models at any time later is not only extremely expensive but also difficult to verify. Finite element models of typical fighter aircraft can cost millions of dollars to develop and verify from the drawings. The fact is that many AFLC and AFSC centers are buying finite element models of the F-15, F-16, F-111 and A-10 after they have been in the inventory for a number of years.

SBIR PROGRAM AT AFWAL

As part of a small business innovative research (SBIR) program AFWAL initiated a study to address the issues of developing A F standards to take delivery of the finite element model data and the feasibility of establishing an information center to archive, validate and distribute the data in a secure environment. The current practice of each organization buying or developing finite element models as the need arises and then throwing them away is not only wasteful but it also represents a lost opportunity for technology base validation. There are far too many Air Force systems and they are too complex for any Air Force unit to undertake this task on a voluntary (using its own budget) basis. An industrially funded information center is probably the best way to maintain and distribute the data. However, the intent is not to create a Goliath, which costs more to feed it than the benefits derived from it. The SBIR efforts should provide estimates of the comparative costs of running such a center and the benefits.

NEED FOR TAKING DELIVERY OF THE ANALYSIS MODEL

The Air Force does not build its own airplanes, and it seldom generates the finite

element models of its aircraft. As part of the systems program office (SPO) contracts, contractors develop and make finite element analyses to show the adequacy of its designs. Every contract for a new system requires that the contractor adhere to a well defined structural design criteria. This criteria specifies, among other things, the strength, stiffness and aeroelastic requirements and the margins of safety for all the safety related items. The contractor must show by analyses and tests that his designs do satisfy all the requirements. The SPO contract provides funding for all this verification. So it appears that it is a simple matter for the SPO to include the delivery of the finite element data in its Contract Data Requirements List (CDRL-AF Form 1423). Actually, it is not as simple, because if all the data becomes a CDRL item, then the contractor has to make a serious effort to assure that the data is correct. In addition, the contractor must show that sound modeling guidelines have been followed in the generation of the analysis models. Otherwise, the data can become a liability in the case of a system failure. This additional burden will certainly increase the cost of the analysis.

With hectic schedules and tight budgets the SPOs, understandably, are reluctant to assume this additional responsibility. However, if we add up all the benefits that can be derived, during the life of the system, from the readily available finite element data, it is hard to believe that any other way is prudent. For example, a successful system stays in the Air Force inventory for 20 to 30 years . During this time many changes are made to the system. Now derivatives, new weapon systems and new repair and maintenance procedures all need the analysis data of the baseline system not only for assessing the effect of the modifications but also to check the safety of the system. An even more important consideration is that we will learn to take analysis seriously.

The aerospace industry has built many successful systems over the years. Much of this success is due to excellent test programs. Every system development cycle includes coupon

tests, component tests, full scale static and fatigue tests, ground vibration tests, wind tunnel tests and flight tests. These tests are enormously expensive and time consuming. When only one or two systems were under development, it was possible to budget for all these tests. There is no way that the Air Force or the nation can afford this development cycle for all SDI space systems, National Aerospace Plane, ATF, ATB and who knows how many other systems are on the drawing board. Analysis, using finite element models on modern computers, can be the key to reducing the overall development costs. Data from systems in operation is invaluable for validating the new technology base and sharpening analysis and optimization tools. The anticipated long term benefits are too compelling to do business any other way.

AF ORGANIZATIONS INTERESTED IN FINITE ELEMENT MODELS

A number of organizations in the Air Force need the finite element models in order to do their job well. The AFLC depots need this information to evaluate new repair and maintenance procedures and for modifications to add new weapons systems. Warner Robbins AFB needs F-15 models, Ogden needs F-16 models, McClellan is buying F-111 and A-10 models now and in the future they would be in need of the ATF finite element models. Kelly AFB works with trainers (T38) and other systems in operation. When the job is done, they usually throw away these models or at least do not keep them in a form that others can use.

AFSC organizations such as ASD, ASD-AFWAL, 4950th Test Wing, Eglin AFB, the Flight Test Center at Edwards (including NASA Dryden at Edwards) need these models for the investigation of new derivatives and new stores and armaments. For example, there are four different derivatives of the F-15, and a fifth one is being studied for possible adaption as a STOL aircraft. A similar number, but perhaps not as many derivatives, is

being proposed for the F-16. AFSC laboratories can use these models very effectively for technology validation. For example, validation of a new system like ASTROS using a real operational aircraft such as the F-15 or F-16 can establish credibility and help move the new technology to the design office. There are other organizations from SAC which will be needing analytical models of the B1, etc.

The 4950th Test Wing at WPAFB makes extensive modifications to accommodate and test new avionics, radom and other surveillance systems. Flight Tests Centers and NASA Dryden at Edwards AFB need the FEM data to confirm the adequacy of critical flight safety related systems.

SAFE GUARDING FINITE ELEMENT MODEL DATA

Finite element models contain extremely valuable information about the most sophisticated aircraft in the world. The purpose of organizing, validating, and archiving this data is for the future use of Air Force units and their contractors. If they are easy for us to access, they can be just as convenient for our adversaries and competitors to obtain these models. It will be a serious matter if the wrong parties get hold of these models. This is one issue that the SBIR studies will address in depth. It appears now that the best way to protect these models is by maintaining them on a single central computer system in a binary file format and letting the users access them on a strictly enforced need to know basis. This may not be the best way and other alternatives must be investigated.

STANDARDS FOR FINITE ELEMENT DATA

Finite element model data is just useless unless sound modeling guidelines are followed in generating it. A thorough understanding of the data and the procedures for validating it

against the real structural design criteria are extremely important before archiving and distributing. In 1985, a draft Data Item Description (DID) was prepared by AFWAL/FIBRA to achieve some of these objectives. This draft DID is attached as an appendix to this paper. This DID postulates three requirements for delivery of the data. The purpose of the general requirements is to provide a concise statement of the problem with supporting data to explain the analysis objectives. The second item is the analysis data requirements in which five types of analysis are identified. In each case the data requirements are listed in some detail. The third item is listed as other requirements. This item also specifies how the data should be supplied to the government regardless of what program the contractor uses in making the analysis. The DID does not recommend for or against any particular program and allows complete freedom to exercise creativity. It only requires that the data be supplied in a standard format, so that the government agency does not have to dig in to proprietary programs in order to understand the data. This DID will be modified and enhanced when the SBIR studies are completed.

In conclusion, the authors firmly believe that it is in the best interest of the Air Force to pursue this issue vigorously for promoting technical excellence as well as for stretching scarce resources (getting bigger bang for the buck).

DATA ITEM DESCRIPTION

Form Approved
OMB No. 0704-0188
Exp Date: Jun 30, 1986

1. TITLE	2. IDENTIFICATION NUMBER
Data for Finite Element Models of Aerospace Structures	

3. DESCRIPTION/PURPOSE

This report describes the data elements and the format of the finite element models of aerospace structures to be delivered to the Air Force. This data will be used to verify the contractors structural analysis and/or to determine the effects of future modifications (or changes) to the structure or its operational conditions. It should be noted that not all the data items will be applicable to every system. The applicable items will be identified on a CDRL (DD Form 1423).

4. APPROVAL DATE (YYMMDD)	5. OFFICE OF PRIMARY RESPONSIBILITY (OPR)	6a. DTIC REQUIRED	6b. GIDEP REQUIRED

7. APPLICATION / INTERRELATIONSHIP

The finite element data generated for verifying the structural design criteria of an aerospace vehicle (designed and paid for by the Air Force) should be be the property of the Air Force and should be delivered in a suitable and understandable form for future use. This data will be extremely valuable in assessing the integrity of the system after modifications, repairs and maintenance.

8. APPROVAL LIMITATION	9a. APPLICABLE FORMS	9b. AMSC NUMBER

10. PREPARATION INSTRUCTIONS

10.1 General Requirements. The finite element data supplied in response to this CDRL item must accompany a problem narrative. This narrative must include the following items:

- Configuration version.
- Identification of the documents and/or drawings from which the model was generated. Copies of these documents must be provided if they are not available to the government.
- A key diagram showing the location of the component being modeled in relation to the rest of the structure.
- A brief description of the physical phenomena being modeled.
- A discussion on the coarseness/fineness of the grid selected.
- A rational explanation for the elements selected for the model.
- An explanation of the boundary conditions.
- Materials - Identification of the Mil Standard from which the mechanical properties were derived. Reasons for any deviations from the standard properties.
- A complete description of the flight maneuvers for which the loading conditions are attributed.
- Planform used for aerodynamic analyses showing all important dimensions.

10.2 Analysis Data Requirements. The finite element analysis models are classified into the following five categories:

I. Static Analysis Models

II. Dynamic Analysis Models

III. Aeroelastic Analysis Models

IV. Heat Transfer Analysis Models

V. Acoustic Cavity Analysis Models

The CDRL will call for the specific models required.

10.2.1 Static Analysis Model Requirements. A static analysis basically requires a good stiffness representation. However, when gravity loading or inertia relief conditions are specified, a good mass representation is also required. This mass representation must include both structural and nonstructural mass distributions. The finite element models for static analysis must consist of the following items as a minimum.

i) Geometry - (as appropriate)

Grid Point Coordinates
Element Types
Element Connections
Coordinate Systems

ii) Element Properties - (as appropriate)

Thicknesses
Cross-sectional Areas
Moments of Inertias
Torsional Constants
Fiber Orientations
Other properties as required for special elements.

iii) Material Properties - (as appropriate)

Isotropic
Anisotropic
Fiber Reinforced Composites
Temperature Dependent Properties
Stress Dependent Properties
Thermal Properties
Damping Properties
Other properties as required for special problems.

iv) Boundary Conditions - (as appropriate)

Single Point Constraints
Multipoint Constraints
Partitioning for Reduction or Substructuring

dynamic aeroelastic stability (flutter analysis), and the details of the method (references) and the necessary data shall be provided with the models. Flutter analysis is generally an iterative process and can also involve more than one flutter mechanism. There are often special techniques associated with the flutter analysis, and they can be defined in terms of the ranges of the aerodynamic parameters. Such data shall be included in the aeroelastic models. In addition, provisions must be made to include the effects of the rigid body modes on the flutter model (body freedom flutter). If it is anticipated that these models will be used for aeroservoelastic analysis, then the data shall be provided for a state space formulation. Also sensor actuator locations and their range of operation and/or limitations shall be included in the data. In addition, a flight control system block diagram shall be provided with sufficient information to define all transfer functions and gains using S-domain variables for analog systems or Z-domain variables for digital systems. The units of important parameters shall be provided.

10.2.4 Heat Transfer Analysis Models. There are three elements to heat transfer models: the heat conducting medium, the boundary conditions and the heat sources and/or sinks. The data requirements of the heat conducting medium are similar to those defined for static and dynamic analysis. For instance the geometry definition includes the grid point coordinates, element types, element connections and coordinate systems. Elements can be classified into volume heat conduction and surface elements. The element type designation for the volume heat conduction element is generally derived from the degree of approximation of its shape functions. The surface elements are used to model a prescribed heat flux, a convective flux due to the difference between the surface temperature and the recovery temperature or local ambient temperature, and radiation heat exchange. Appropriate material properties, single point and multipoint boundary conditions and description of the heat sources (applied forces) have a similar correspondence in the static and/or dynamic analysis. The surface heat convection or radiation details shall be provided (through surface elements) as appropriate. The response variables in heat transfer analysis are generally grid point temperatures or the temperature gradients and heat fluxes within the volume heat conduction elements and the heat flow into the surface elements. Four types of heat transfer analysis are contemplated:

i) Linear Steady-State Response Analysis

ii) Linear Transient Response Analysis

iii) Nonlinear Steady-State Response Analysis

iv) Nonlinear Transient Response Analysis

It is often necessary to adopt special techniques for obtaining stable solutions, particularly in the last two cases. The data pertaining to these special techniques and the limitations of the nonlinear algorithms shall be fully identified.

10.2.5 Acoustic Cavity Analysis Models. Basically there are three elements in acoustic cavity analysis models: the acoustic medium, the boundaries, and the sources of excitation. The acoustic medium model shall consist of grid points and acoustic elements connecting these grid points. The response variables are generally the pressure levels and the gradients of the pressures (with respect to the spatial variables) at the grid points. So for a general three dimensional acoustic analysis there will be four degrees of freedom per node (corresponding to four response variables) in an acoustic medium model. The properties of the acoustic medium can vary with the temperature and pressure distribution and density. The boundaries of the acoustic model can be solid walls, flexible walls, openings in the walls and walls with acoustic material which can be represented as a complex acoustic impedance. For complicated boundary conditions separate finite element models may be necessary in order to derive the boundary conditions for the acoustic model. These finite

v) Loading - (as appropriate)

Static Loads
Gravity Loads
Thermal Loads
Centrifugal Loads
Other loading conditions as required for special simulations.

For buckling or nonlinear analysis additional information is required on the following items:

- How the nonlinear matrices are derived.
- The method of solution for the nonlinear problem.
- A description of the method in the case of an eigenvalue analysis.

10.2.2 Dynamic Analysis Models. The dynamic analysis models require i) geometry, ii) element properties, iii) material properties, and iv) boundary conditions as described for the static case. In addition an accurate nonstructural mass and damping representation is required. Generally five types of dynamic analysis are contemplated.

- Normal Modes Analysis or
- Complex Eigenvalue Analysis
- Frequency Response Analysis
- Transient Response Analysis
- Random Response Analysis

In the first two cases only the method of eigenvalue analysis and the frequency (modes) range of interest need be specified. For frequency response analysis the frequencies of interest must be specified. For transient response analysis the dynamic load must be defined as a function of time or must be provided as tabular values. For random response analysis the statistical nature of the input (such as PSD, Auto Correlation) and the statistical quantities of the output desired must be specified. In addition all the information on dynamic reduction and/or modal reduction must be specified.

10.2.3 Aeroelastic Models. An aeroelastic analysis requires mathematical models of the structure and the aerodynamics. The structure is generally represented by finite element models (FEM). The requirements for the structures models are as specified under static and dynamic analysis. They include mass, stiffness and damping representation. Both structural and nonstructural mass distributions shall be included in the mass model. The aerodynamic models are generally based on paneling or equivalent methods. The requirements of the aerodynamic models are those of the panel geometry which cover all the lifting surfaces including the control surfaces, the empennage (horizontal and vertical tails) and canard surfaces. The fuselage slender body and interference panels shall be modeled to represent the flowfield adequately. The altitude (air density), mach number and other relevant aerodynamic parameters must be specified. The details of the aerodynamic theory and the limits of its validity must be clearly defined. In addition, data for the force and displacement transformations from the structural grid to the aerodynamic grid (and vice versa) shall be included in the aeroelastic models. Two types of aeroelastic analysis are contemplated. Both deal with the phenomenon of aeroelastic stability. The real eigenvalue analysis is the basis for determining the static aeroelastic stability. There are a number of methods for determining

element models are based on solid mechanics and their data requirements are similar to those described for the static and dynamic analysis earlier. The acoustic excitation source model shall have information on the spatial distribution and the statistical properties (in terms of the frequency content) of the noise. For a deterministic case, however, definition of the forcing function includes the magnitude, phasing and frequency along with the spatial distribution. The acoustic excitation is generally given as velocity or pressure applied to the medium over prescribed surfaces or at grid points. If the disturbance is from mechanical sources, separate finite element models of the sources shall be supplied as required. These models are also generally solid mechanics models and their requirements are similar to static and dynamic analysis models. Generally three types of acoustic analysis are contemplated.

- Eigenvalue Analysis
- Steady-State Solution
- Nonlinear-Analysis

In the eigenvalue analysis the acoustic natural frequencies and mode shapes are determined. The purpose is to compare the natural frequencies of the cavity with those of the forcing function and estimate the resonance effects, and to compare the natural frequencies to the resonant frequencies of any structure which may be placed in the cavity. This analysis provides useful information for design changes in the cavity either by altering the overall dimensions or by introducing noise suppression mechanisms such as baffles or by adding noise suppression material to introduce acoustic wall impedance. This analysis does not require explicit definition of the forcing function. The steady-state solution gives the response of the cavity to a given excitation. This analysis can be in the time or frequency domain. The nonlinear analysis involves an iterative solution when the properties of either the cavity or the acoustic medium vary significantly with the pressure levels and/or temperature.

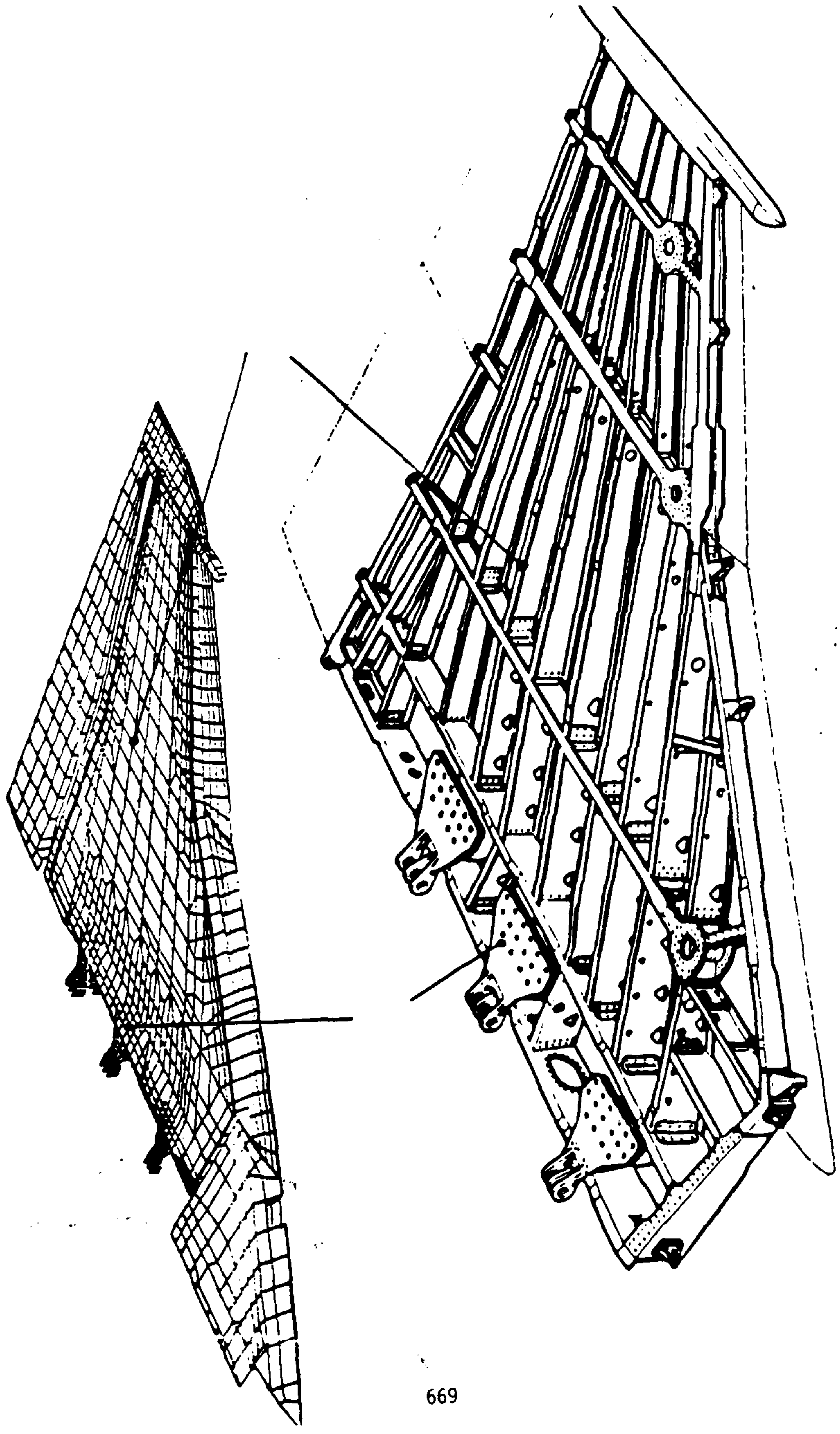
10.3 Other Requirements.

The input data for all the finite element models must be provided in a format compatible with the latest government version of NASTRAN (COSMIC/NASTRAN). If the original analysis was made with another finite element program, the data shall be converted to the COSMIC/NASTRAN format. If NASTRAN does not have compatible elements or capability, the elements that are most appropriate must be identified and projections must be provided on the expected differences.

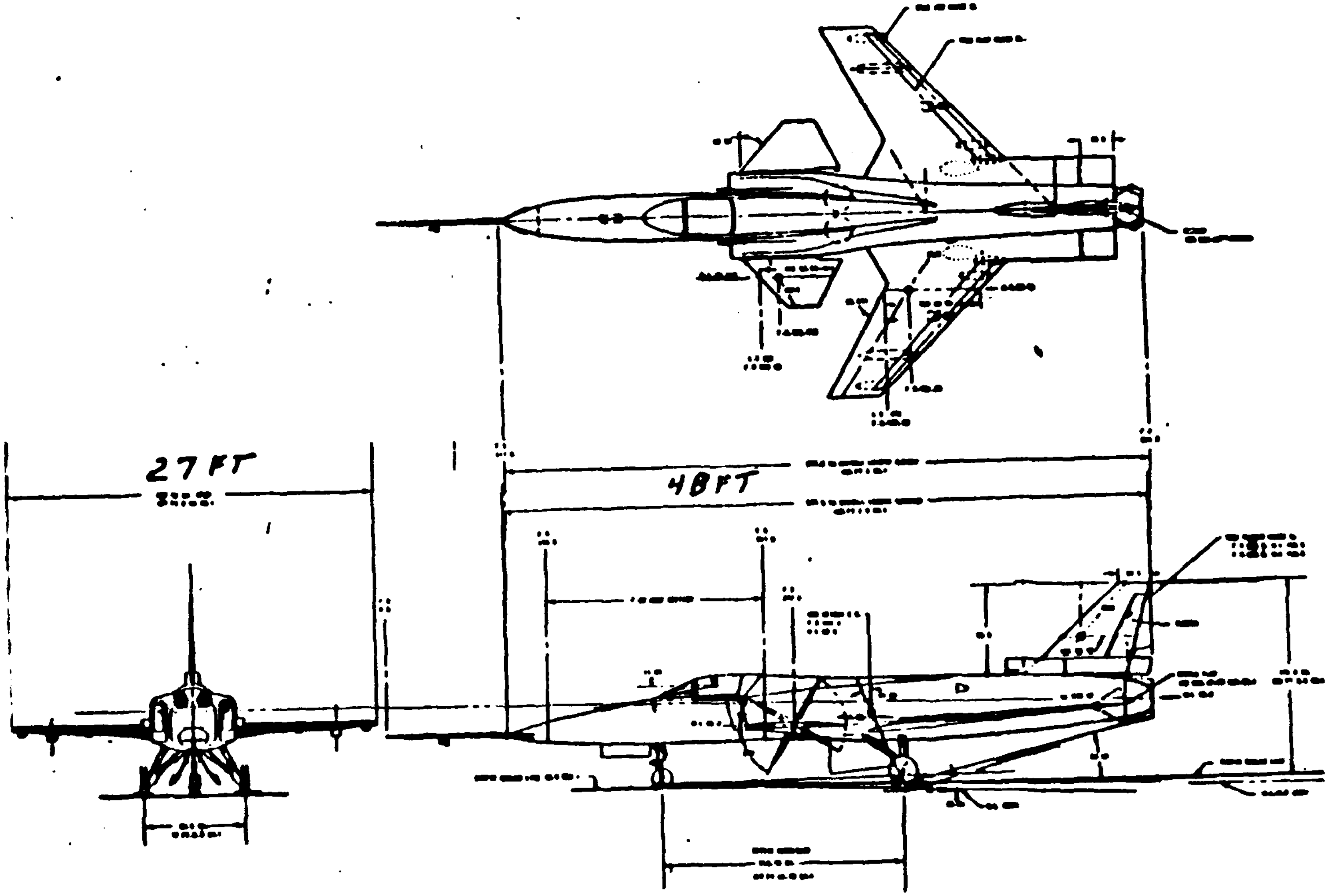
In addition to the input data a summary of output results (such as deflections, stresses, frequencies, etc. at critical areas) shall be provided for future validation of the models. Also a brief description of how these results were used to satisfy a specific design criteria. A set of undeformed and deformed plots of the structure shall be provided with all the finite element models.

For Details Contact

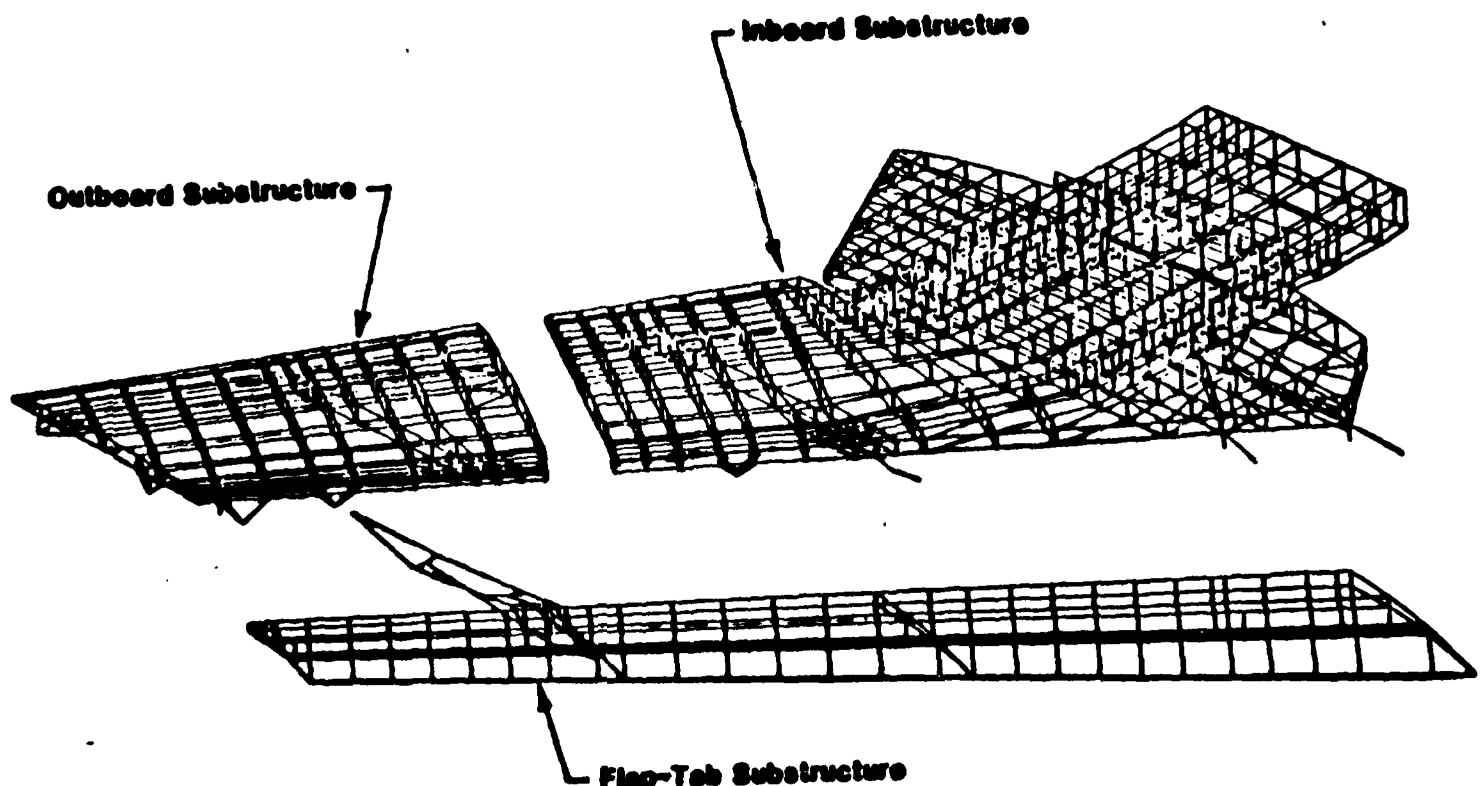
Dr. V. B. Venkayya
AFWAL/FIBRA
Wright-Patterson A F B, OH, 45433
513-255-6992



GENERAL ARRANGEMENT



WING SUBSTRUCTURE



1820 NODES
4420 MEMBERS
6060 DEG OF FREEDOM

MCS 11-7-407

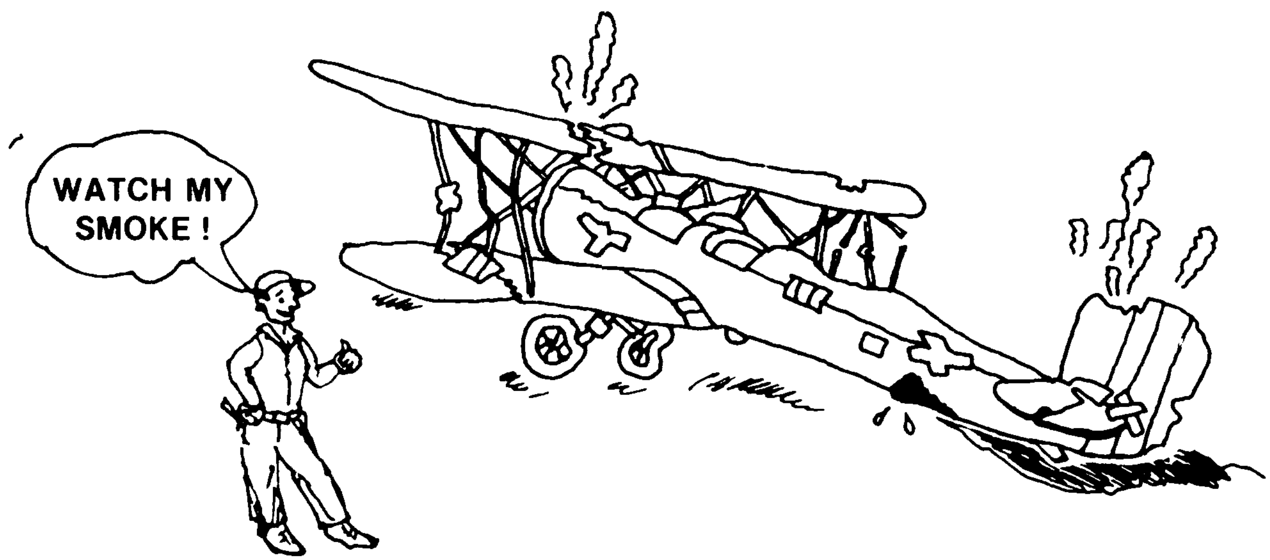
"SMART" STRUCTURES

CAPT C. MAZUR
T.G. GERARDI
G.P SENDECKYJ

AFWAL / FIBEC
WPAFB, OHIO,
45433-6553

EVOLUTION OF STRUCTURAL INTEGRITY MONITORING

- KICK THE TIRE - LIGHT THE FIRE !
- WALK AROUND INSPECTION
- MANDATORY ANNUAL AND 100 HR INSPECTIONS
- VGH RECORDERS
- FIRST AND SECOND GENERATION ASIP
- MADAR (C-5A)
- "SMART STRUCTURES'



MCS 6-7-142

"NERVOUS SYSTEM" FOR AEROSPACE STRUCTURES
PROBLEMS

AIRCRAFT:

- NO REAL-TIME STRUCTURAL MONITORING
- INDIVIDUAL AIRCRAFT TRACKING IS COSTLY AND INACCURATE
- COMPOSITE DAMAGE NOT ALWAYS DETECTABLE
- IN-FLIGHT DAMAGE CURRENTLY UNDETERMINABLE
  - FATIGUE DAMAGE
  - OVERLOAD DAMAGE
  - FOREIGN OBJECT DAMAGE
- TOO MANY COSTLY MAINTENANCE FALSE ALARMS

"SMART" STRUCTURES
CONCEPT

- AEROSPACE STRUCTURES INSTRUMENTED WITH NETWORK OF SENSORS AND COMPUTER(S) THAT
  - PERFORM REAL-TIME QUANTITATIVE NONDESTRUCTIVE EVALUATION
    - LOCATION AND NATURE OF DAMAGE
    - EXTENT AND SEVERITY OF DAMAGE
  - ACQUIRE ENVIRONMENTAL / LOAD HISTORY DATA (LESS)
  - PERFORM REAL-TIME SAFETY-OF-FLIGHT CALCULATIONS (IAT)
    - RESIDUAL STRENGTH
    - REMAINING LIFE
  - ADVISE CREW ON FLIGHT / MANEUVER RESTRICTIONS
  - SCHEDULE MAINTENANCE AND REPAIR
  - ADVISE MAINTENANCE PERSONNEL

MCS 10-7-883

EXPANDING TECHNOLOGIES

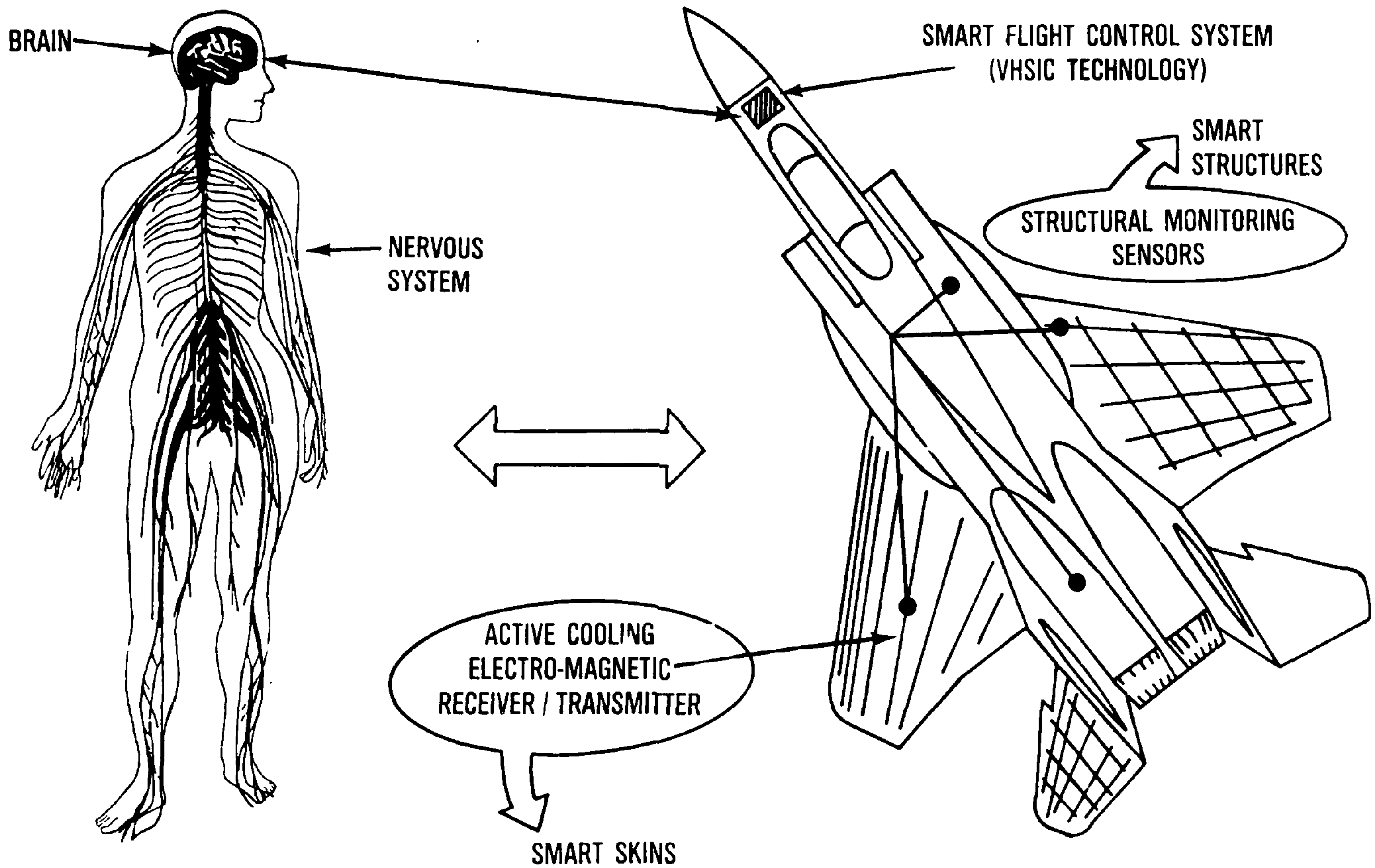
COMPUTERS

- BEYOND VHSIC COMPUTER TECHNOLOGY
- NEAR INFINITE MEMORY
- ADVANCED SOFTWARE: ON BOARD CALCULATION, AI
- ADVANCED I/O DEVICES
- SPEED DOUBLES ITSELF EVERY 5 YEARS
- UNLIMITED FUTURE: FIBER OPTICS, SUPERCONDUCTORS ... HOLOGRAPHY

SENSORS

- FIBER OPTICS
- PIEZOELECTRIC MATERIALS
- ACOUSTIC EMISSION
- Co-Ax CABLE (CAVITY RESONANCE)

"SMART" STRUCTURES
APPLICATION



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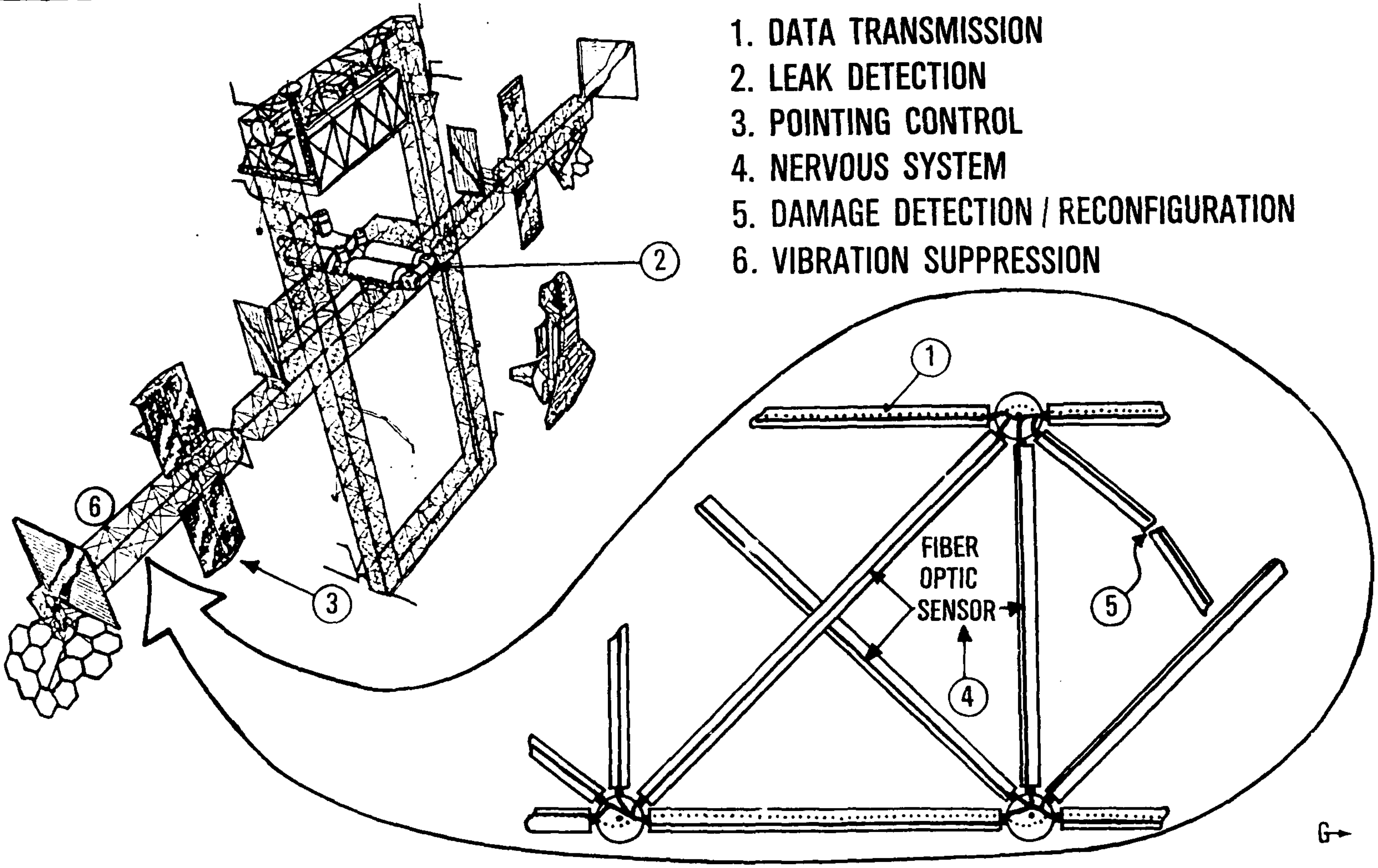
MCS 6-7-141

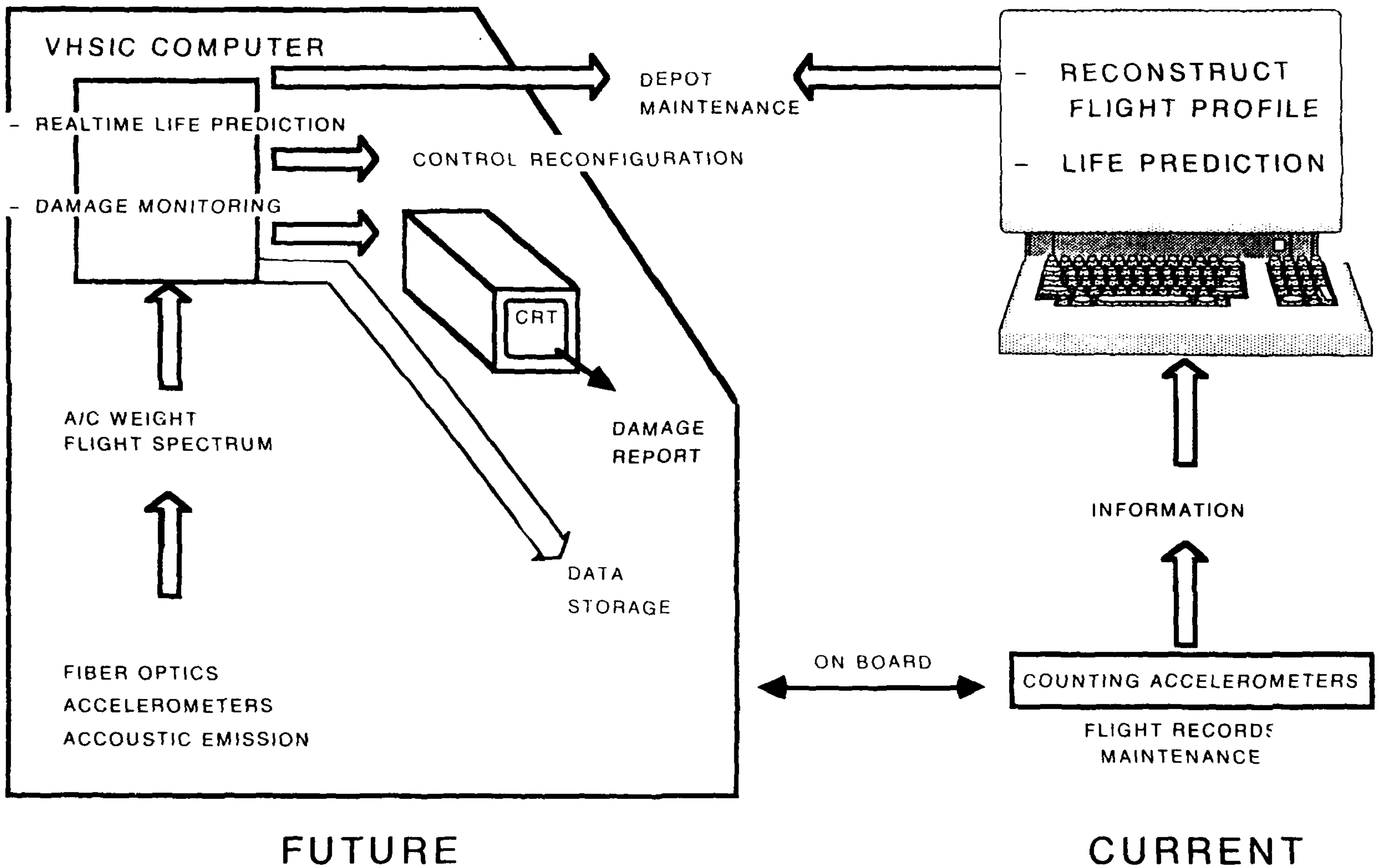
"NERVOUS SYSTEM" FOR AEROSPACE STRUCTURE APPLICATIONS

1. DATA TRANSMISSION
2. LEAK DETECTION
3. POINTING CONTROL
4. NERVOUS SYSTEM
5. DAMAGE DETECTION / RECONFIGURATION
6. VIBRATION SUPPRESSION



"SMART" STRUCTURES

INDIVIDUAL AIRCRAFT TRACKING PROGRAM



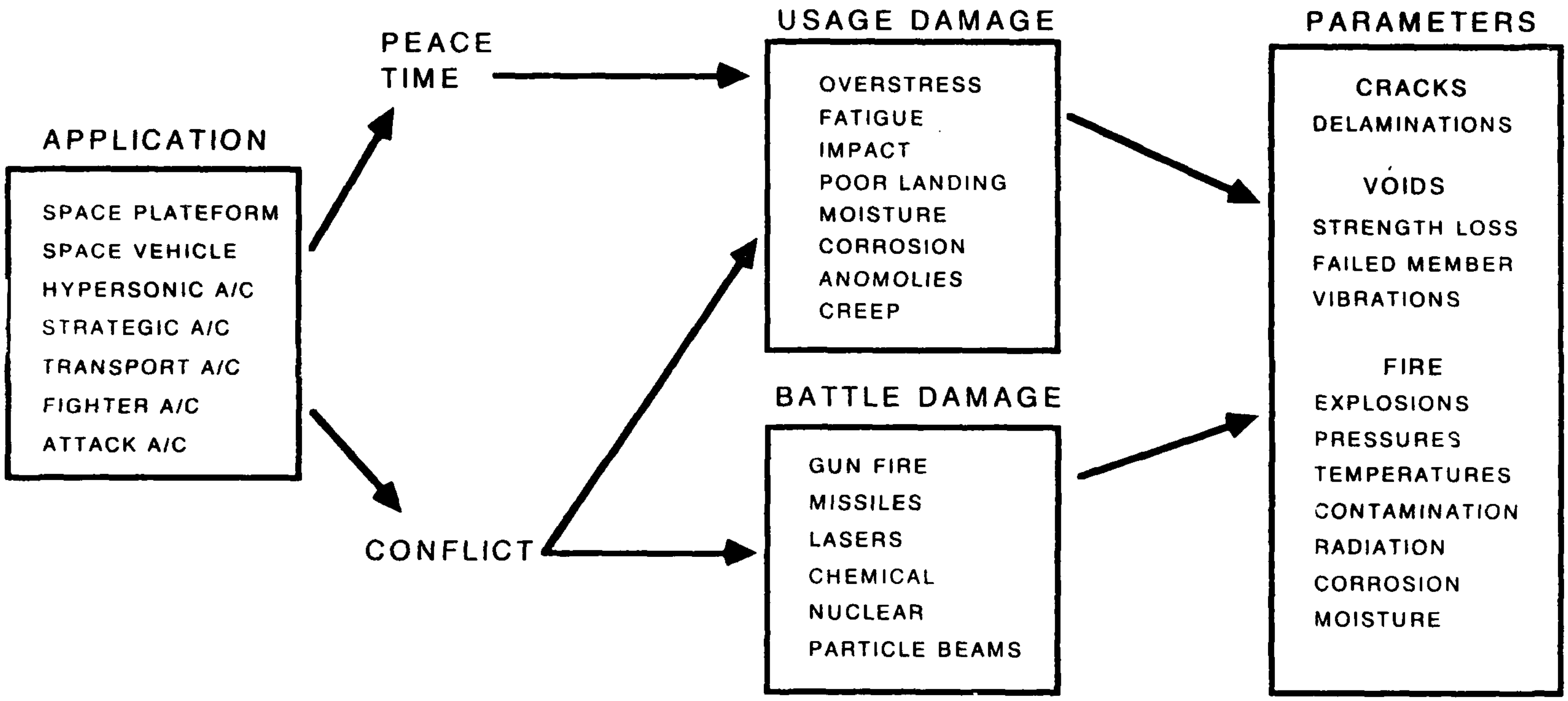
MCS 11-7-409

"SMART" STRUCTURES
SYSTEM REQUIREMENTS

- ULTRA RELIABLE
- USER FRIENDLY
- "REMOVE & REPLACE" PHILOSOPHY
- GRACEFUL DEGRADATION
- COMPOSITE & METALS APPLICATIONS
- CRADLE-TO-GRAVE

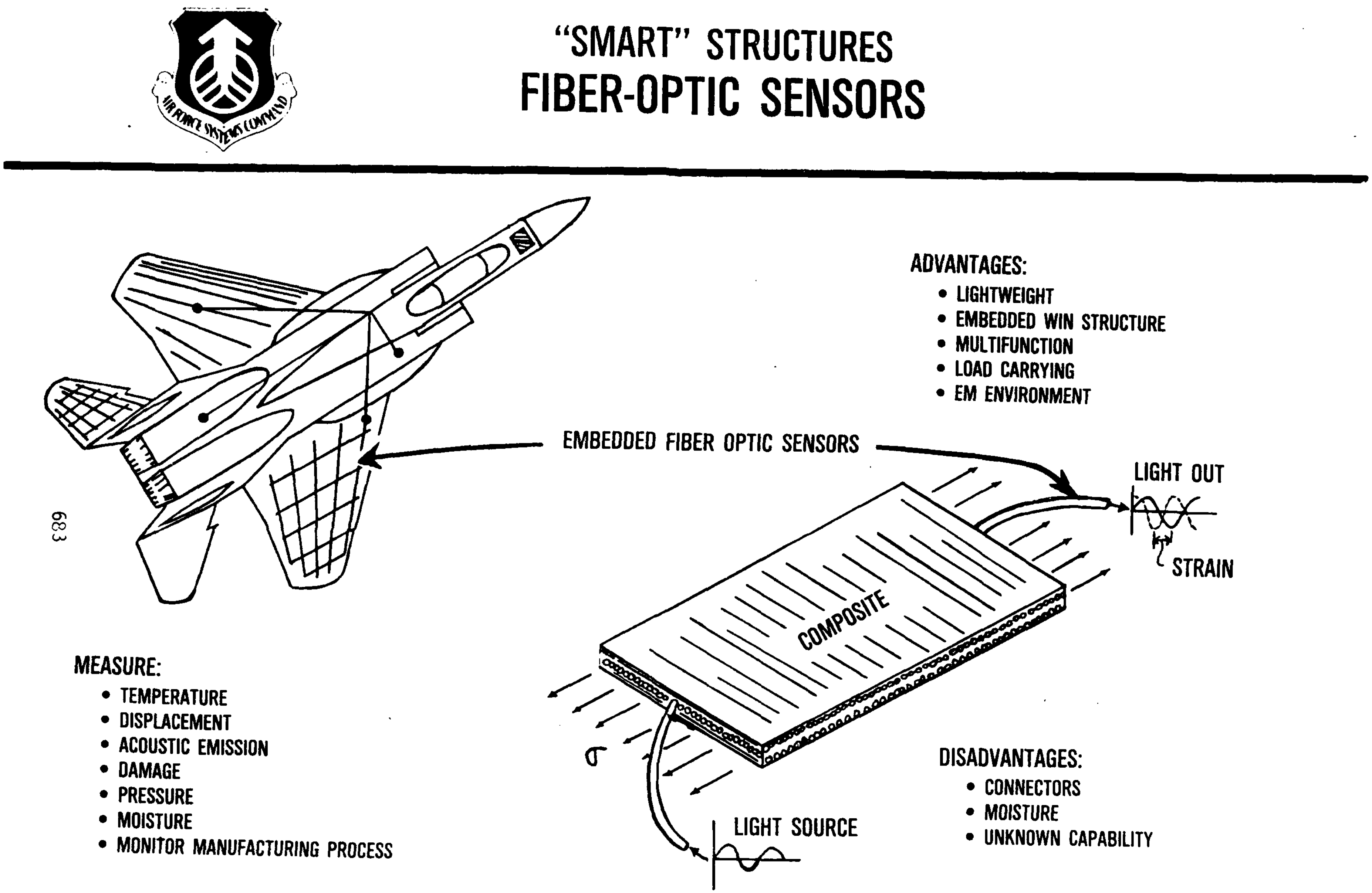
"SMART" STRUCTURES

THOUGHT PROCESS FOR ARCHITECTURAL DESIGN

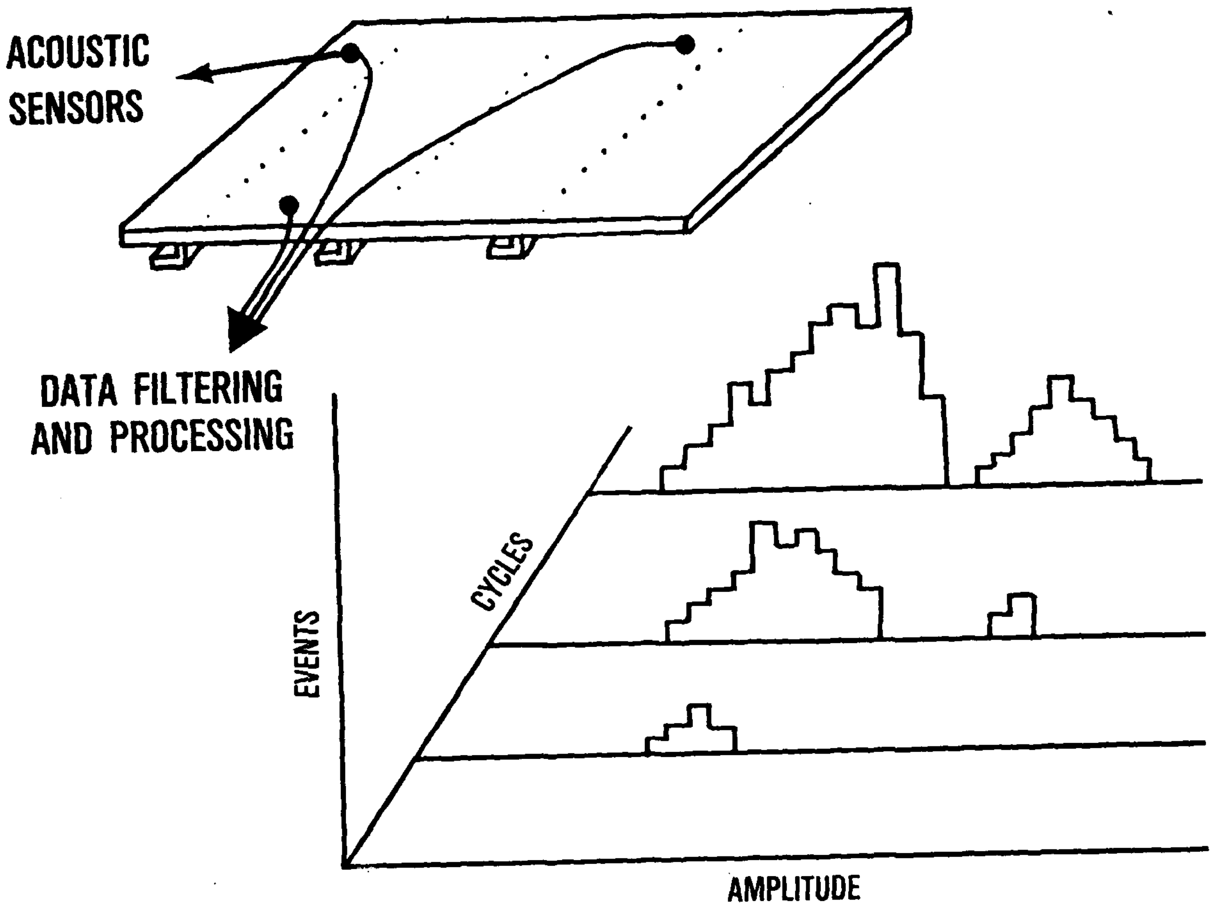


"SMART" STRUCTURES
SENSOR TYPES

- FIBER-OPTIC
- ACOUSTIC EMISSION / ACOUSTO-ULTRASONICS
- PIEZOELECTRIC
- OPTICAL
- STRAIN GAGES
- THERMOCOUPLE
- ACCELEROMETERS
- RADIATION DETECTORS
- INNOVATIVE IDEAS


"SMART" STRUCTURES
FIBER-OPTIC SENSORS
ADVANTAGES:
• LIGHTWEIGHT
• EMBEDDED WIN STRUCTURE
• MULTIFUNCTION
• LOAD CARRYING
• EM ENVIRONMENT
EMBEDDED FIBER OPTIC SENSORS
LIGHT OUT
STRAIN
COMPOSITE
σ
LIGHT SOURCE
MEASURE:
• TEMPERATURE
• DISPLACEMENT
• ACOUSTIC EMISSION
• DAMAGE
• PRESSURE
• MOISTURE
• MONITOR MANUFACTURING PROCESS
DISADVANTAGES:
• CONNECTORS
• MOISTURE
• UNKNOWN CAPABILITY

"SMART" STRUCTURES
ACOUSTIC EMISSION



MCS 11-7-412

"SMART" STRUCTURES
PROCESSING HARDWARE

- PROCESSORS LOCAL TO SENSORS
- CENTRAL PROCESSORS
- VHSIC TECHNOLOGY
- CONNECTION MACHINES
- CRAY ON A CHIP
- OPTICAL COMPUTER

MCS 11-7-413

"SMART" STRUCTURES SOFTWARE

DATA BASES:

- STRUCTURAL RESPONSES
- MATERIAL CHARACTERISTICS
- MAINTENANCE RECORDS
- FLIGHT LOAD RECORDS

RAPID STRUCTURAL ANALYTICAL METHODS

- FINITE ELEMENT
- LOOK UP TABLES
- MODAL METHOD
- REMAINING LIFE PREDICTIONS
- DAMAGE LOCATION / SEVERITY ASSESSMENT

INTELLIGENT OPERATING SYSTEM

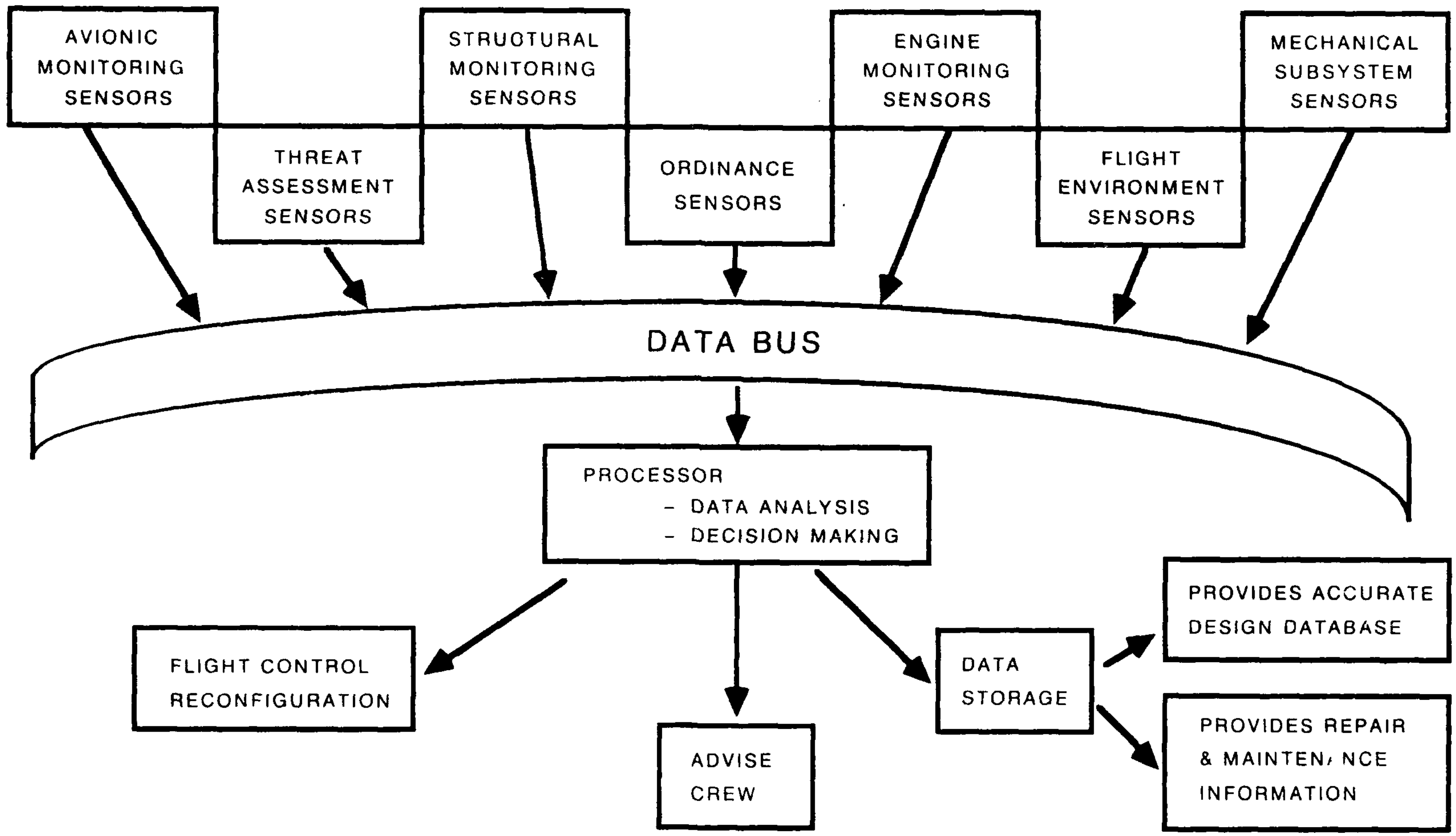
- INFORMATION MANAGEMENT
- BASIC DECISION MAKING CAPABILITY
- MAINTENANCE INSTRUCTIONS

"SMART" STRUCTURES
COMMUNICATION LINKS

- CREW / CRT
- GROUND BASED TELEMETRY
- MAINTENANCE / CRT OR GROUND BASE COMPUTER
- STORAGE
  - DOWN LOADED DAILY / WEEKLY

SMART AEROSPACE VEHICLE

BASIC SYSTEM ARCHITECTURE



MCS 10-7-885

COMPUTER AIDED ACQUISITION LOGISTICS SUPPORT

"CALS"

- SUPPORTED BY MGEN MONROE T. SMITH (RET)
- PROJECT DIRECTION FROM OSD
- HUGE DOD SPONSORED / INDUSTRY SUPPORTED EFFORT

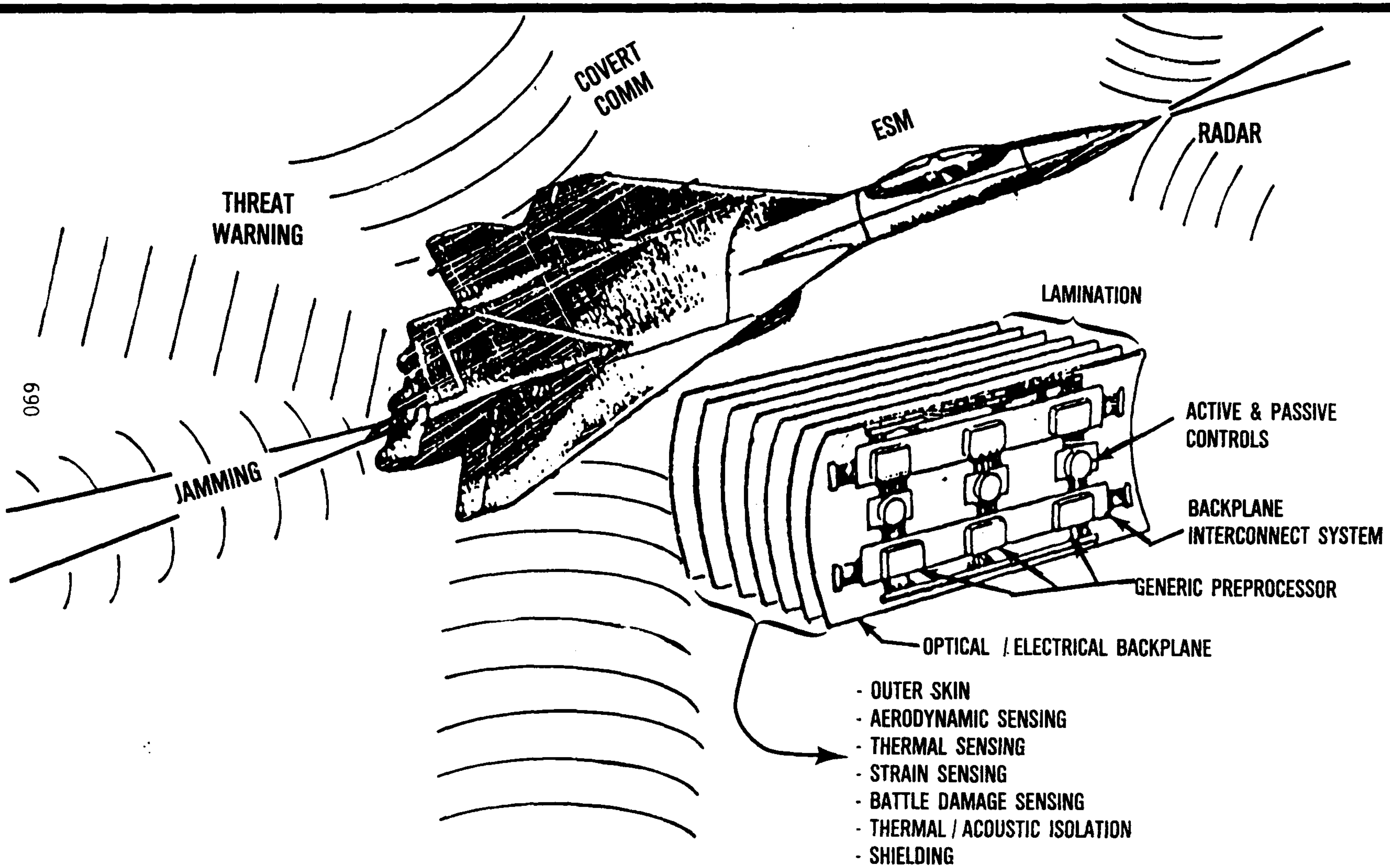
GOALS

- NO PAPER
- STANDARDIZED "COMPUTER AIDED TECHNOLOGIES"
- CLEARING HOUSE FOR RELATED TECHNOLOGIES

POC - USAF / AFSC / PLX

MS 11-7-415

PROJECT FORECAST II → PT-16
SMART SKINS

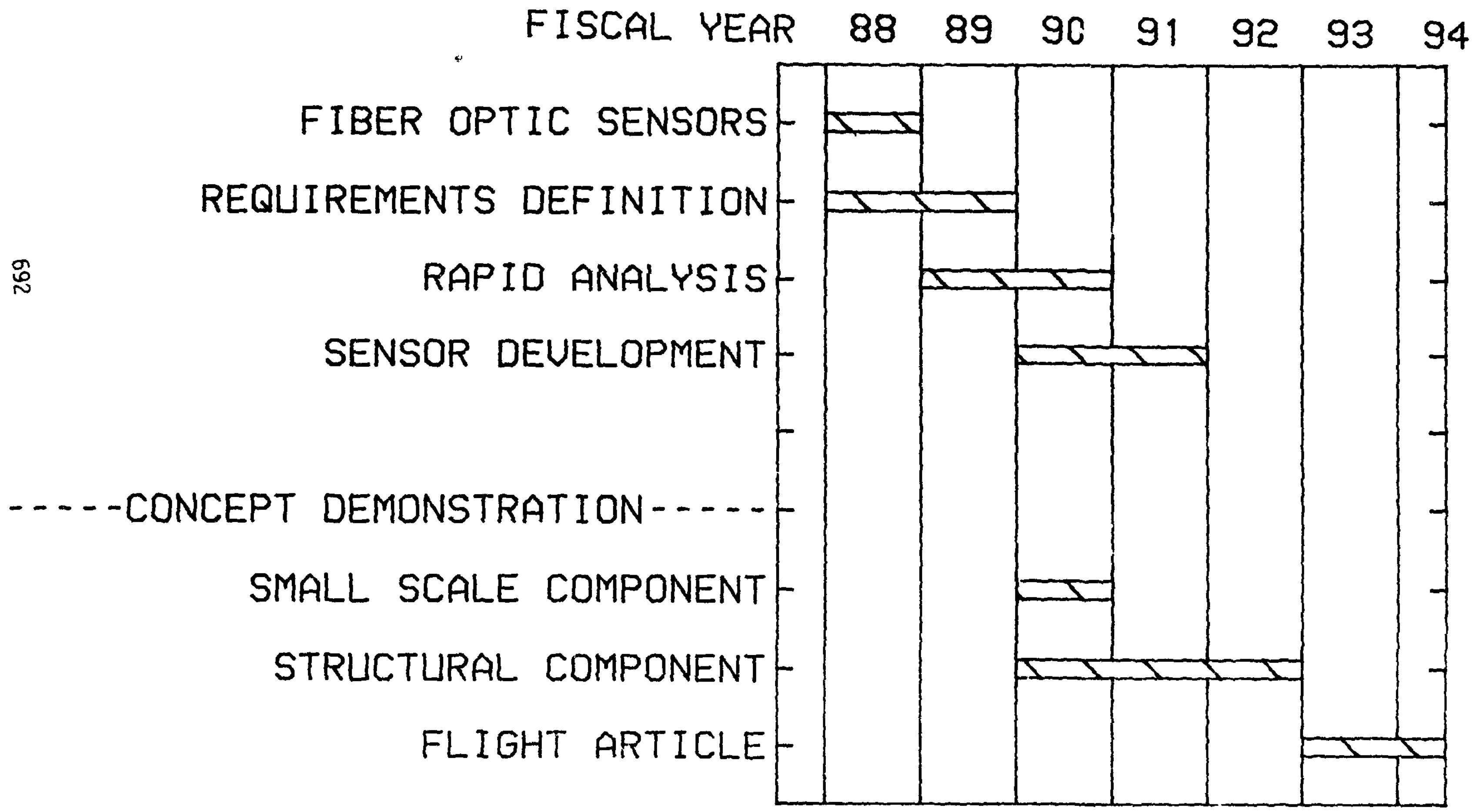


RELATED PROGRAMS

- **"SMART STRUCTURE CONCEPT REQUIREMENTS DEFINITION"
  - SMART STRUCTURE FEASIBILITY STUDY FOR MANNED AEROSPACE VEHICLES
  - SMALL DEMONSTRATION
- "ADVANCED COMPOSITES WITH EMBEDDED SENSORS AND ACTUATORS"
  - LARGE SPACE STRUCTURES
  - DEMONSTRATION PROGRAM
- **"STRUCTURAL INTEGRITY MONITORING USING FIBER OPTICS"
  - DEVELOPMENT OF EMBEDDED FIBER OPTIC SENSORS
- "EMBEDDING FIBER OPTIC SENSORS IN COMPOSITES"
  - ENGINE MATERIAL AND SENSING REQUIREMENTS
- HIGH RELIABILITY FIGHTER CONCEPT INVESTIGATION
  - FIGHTER CONCEPTS FOR THE YEAR 2000

**FIBE PROGRAMS

SMART STRUCTURES ROADMAP



MCS 7-6

"SMART" STRUCTURES
CONCLUSIONS

PAYOFFS ARE:

- REAL TIME STRUCTURAL MONITORING
- MORE ACCURATE LIFE PREDICTIONS
- INCREASED SAFETY IN NORMAL OPERATIONS AND COMBAT
- REDUCE MAINTENANCE COSTS
- REDUCE A/C DOWNTIME
- IMPROVE PERFORMANCE

SESSION VI: INSTRUMENTATION/TRACKING

RESOLUTION OF LOADS MEASUREMENT ANOMALIES

ENCOUNTERED IN

F-16C/D STRUCTURAL LOADS FLIGHT TEST PROGRAM

3 DEC 87

Douglas O. Cornog
ASD/YPEF Structural Loads and Dynamics
Wright-Patterson AFB
513 255 6605 or AV 785 6605

MAJOR TOPICS OF DISCUSSION

- o Background
  - oo F-16 Program Overview; Growth
  - oo Description of General Dynamics Loads Analyses
  - oo Structural Loads Instrumentation Method
- o F-16C/D Flight Test Program Overview
- o Desription of Wing Loads Measurement Results
- o Anomalies in Vertical Tail Loads Measurement
- o Anomalies in Forward/Center Fuselage Loads Measurement
- o Lessons Learned

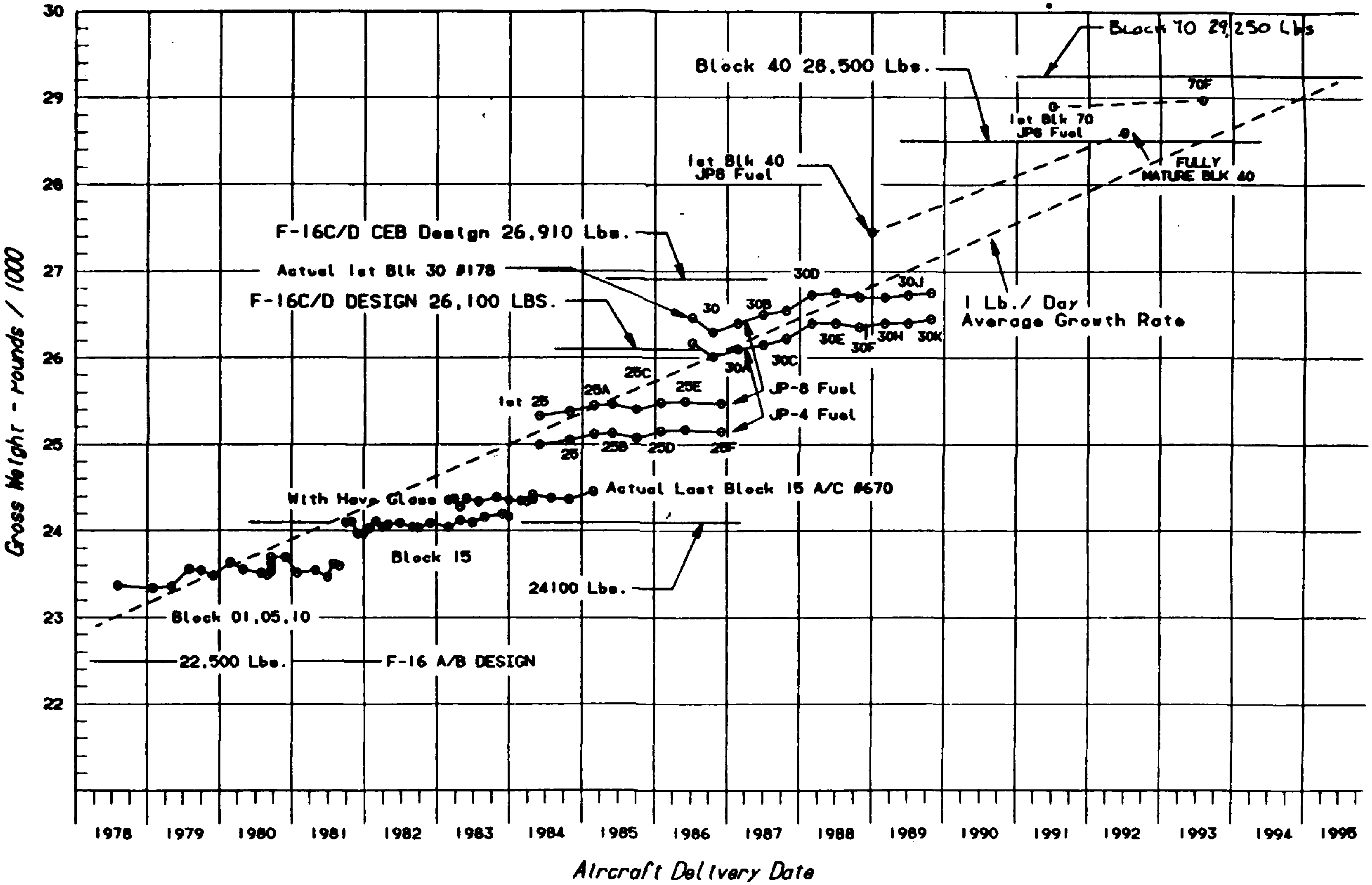
BACKGROUND

- F-16 Program Overview; Growth
- Description of General Dynamics Loads Analyses
- Structural Loads Instrumentation Method

AIRCRAFT PRODUCTION BLOCKS

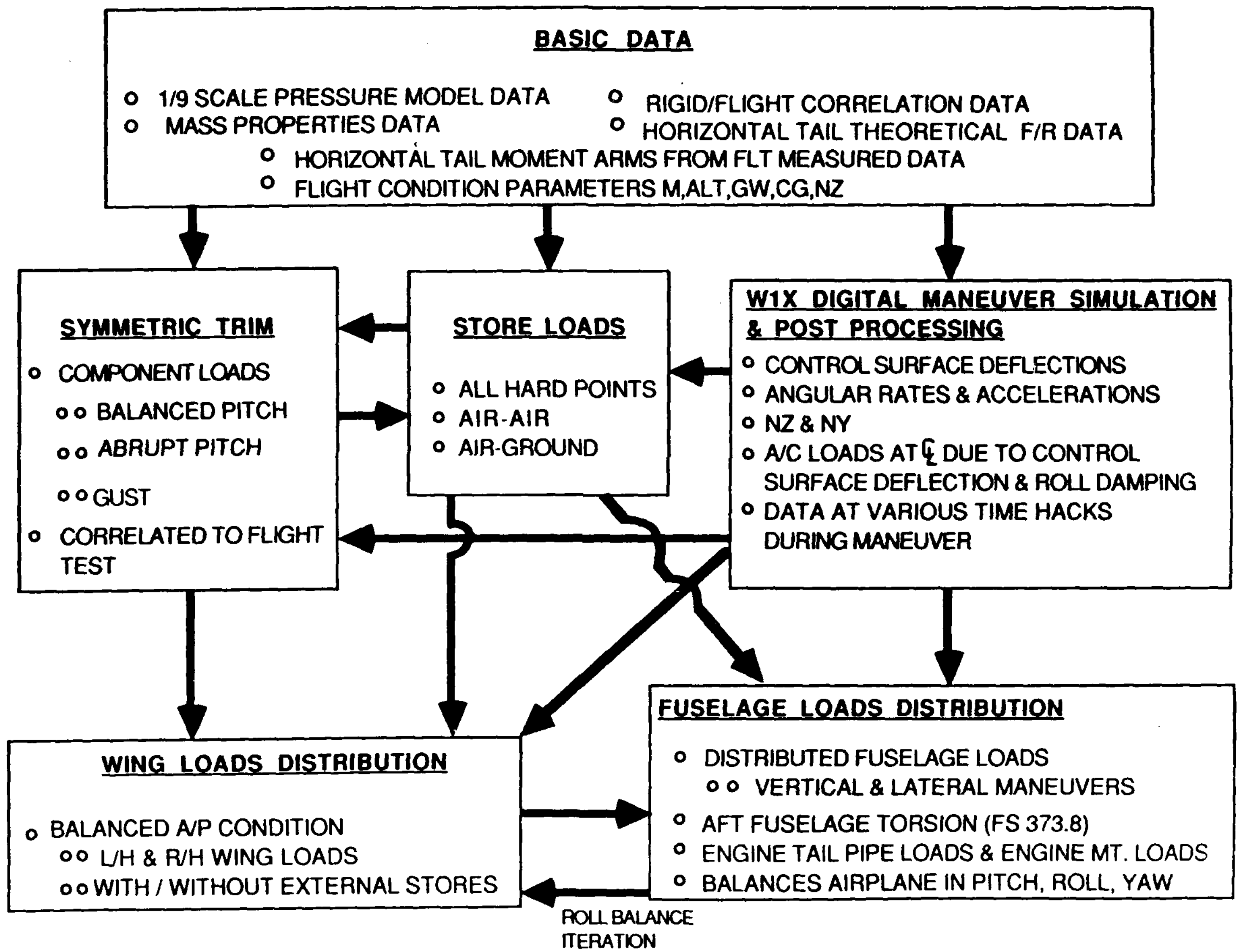
BLOCK 1 - 5	FULL SCALE DEVELOPMENT		F100-PW-200 ENGINE
BLOCK 10	FIRST PRODUCTION A/B	F-16A/B	F100-PW-200 ENGINE
BLOCK 15	OPERATIONAL CAPABILITIES UPGRADE		F100-PW-200 ENGINE
BLOCK 25	FIRST PRODUCTION C/D		F100-PW-200 ENGINE
BLOCK 30	COMMON ENGINE BAY		F110-GE-100 ENGINE
BLOCK 32			F100-PW-220 ENGINE
		F-16C/D	
BLOCK 40	SIGNIFICANT STRUCTURAL CHANGES		F110-GE-100 ENGINE
BLOCK 42			F100-PW-220 ENGINE
BLOCK 50	REPLACED BY "AGILE FALCON"? ADVANCED MATERIALS		F110-GE-129 ENGINE
BLOCK 52		?	F100-PW-229 ENGINE

F-16A/C Basic Flight Design Gross Weight



11-21-86

BALANCED AIRPLANE CONDITION ANALYSIS PROCEDURE



LOADS INSTRUMENTATION

- o Strain gages installed on all airframe components and control surfaces
  - oo Located on primary load carrying structure
  - oo Located and oriented to provide basic sensitivity to only one type of load
- o Strain gages wired into bridge arrangements to
  - oo Maximize sensitivty to primary load
  - oo Minimize interaction effects resulting from secondary effects
  - oo Provide temperature compensation
- o Most instrumentation installed during aircraft manufacturing

LOADS CALIBRATION

- o Purpose: To measure bridge outputs for a variety of known external applied loads at various locations and determine relationship of bridge outputs to applied loads
- o Procedure
  - o Airframe installed in calibration fixture
  - o Calibration point loads of known magnitudes applied to major components and control surfaces
    - oo Loads limited to 80% design
    - oo Applied with two exercise cycles and two 20% incremental data cycles
  - o Relationships of bridge outputs to applied loads are determined
- o Verification
  - oo Known distributed loads are applied
  - oo Distributed loads calculated and compared to actual applied loads

CALIBRATION LOADS

- o Point Calibration Loads
  - oo Wing: 34 concentrated point loads
  - oo Fueslage
    - Forward: 5 concentrated vertical point loads
    - Aft: 3 concentrated vertical point loads
      10 asymmetric horizontal tail loads and 4 vertical tail loads for torsion calibration
  - oo Empennage
    - Vertical tail: 10 concentrated point loads
    - Horizontal tail: 13 concentrated point loads
- o Distributed Verification Loads
  - oo Wing: 3 load distributions
  - oo Empennage:
    - Vertical tail: 4 load distributions
    - Horizontal tail: 4 load distributions

CALIBRATION DATA ANALYSIS

- o Purpose: To determine relationship of strain gage bridge response to applied loads

- o Method

  - oo Calibration data evaluated to determine most accurate relationship

  - oo For simple structure (single load path)
    - Relationship is slope (A) of unit load per unit bridge reponse (μ)

      Load = $A \times \mu$

  - oo For complex structure (multiple load paths)
    - Relationship is multi-term load equation
    - Involves responses of several bridges
    - Coefficients determined by regression analyses

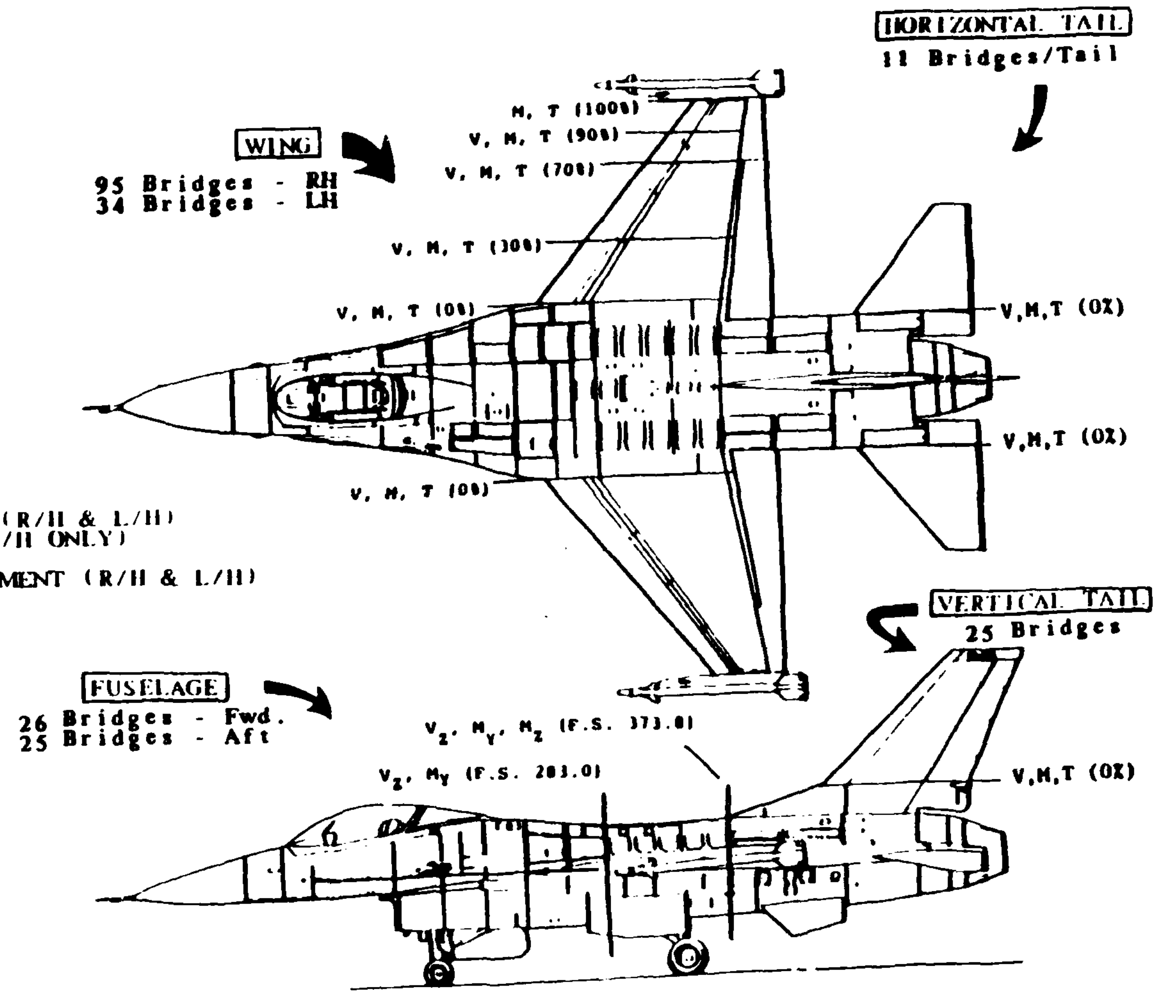
      Load = $A + B\mu_1 + C\mu_2 + D\mu_3$

- o Implementation

  - oo Equation coefficients used to calculate resistor values for electrically combining bridge response

  - oo Provides single data channel for monitoring/recording required measurement

F-16C NO. 3 AIRFRAME LOADS MEASUREMENTS



OTHER: 17 Bridges

FLAPERON HINGE MOMENT (R/H & L/H)
L.E.F. HINGE MOMENT (R/H ONLY)
RUDDER HINGE MOMENT
VENTRAL FIN BENDING MOMENT (R/H & L/H)

<u>F-16C/D FLIGHT TEST PROGRAM OVERVIEW</u>

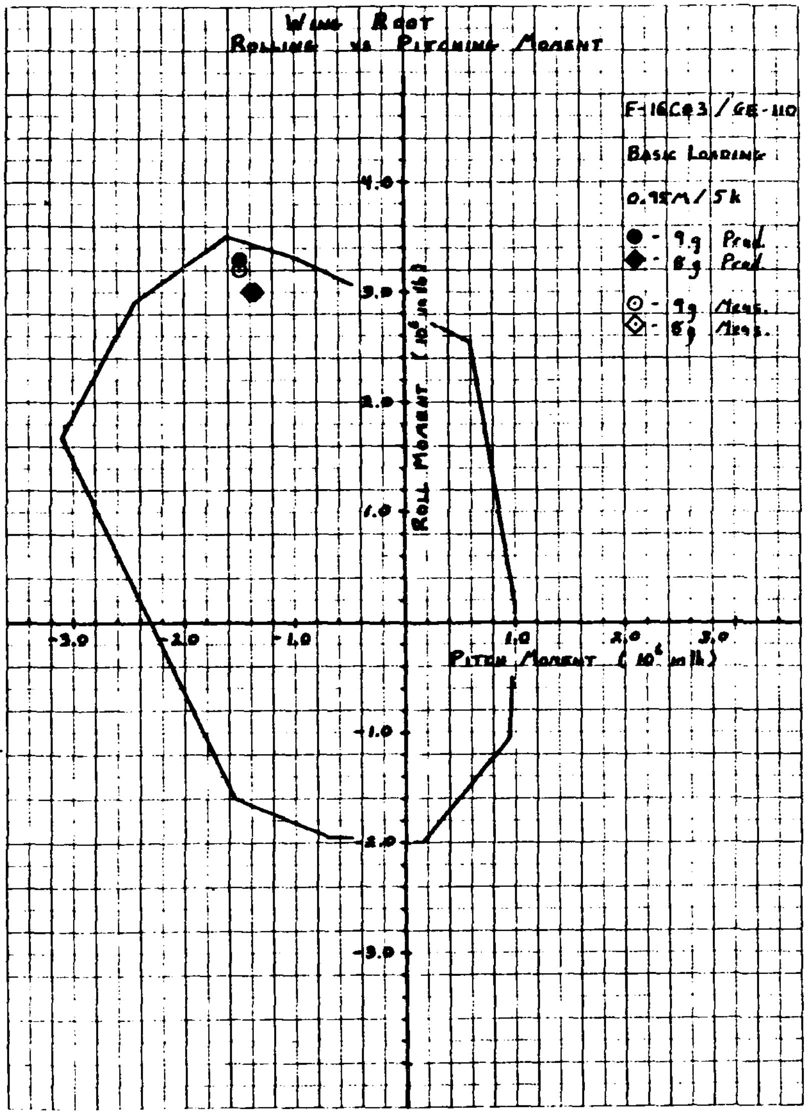
F-16C/D Loads Test: Background

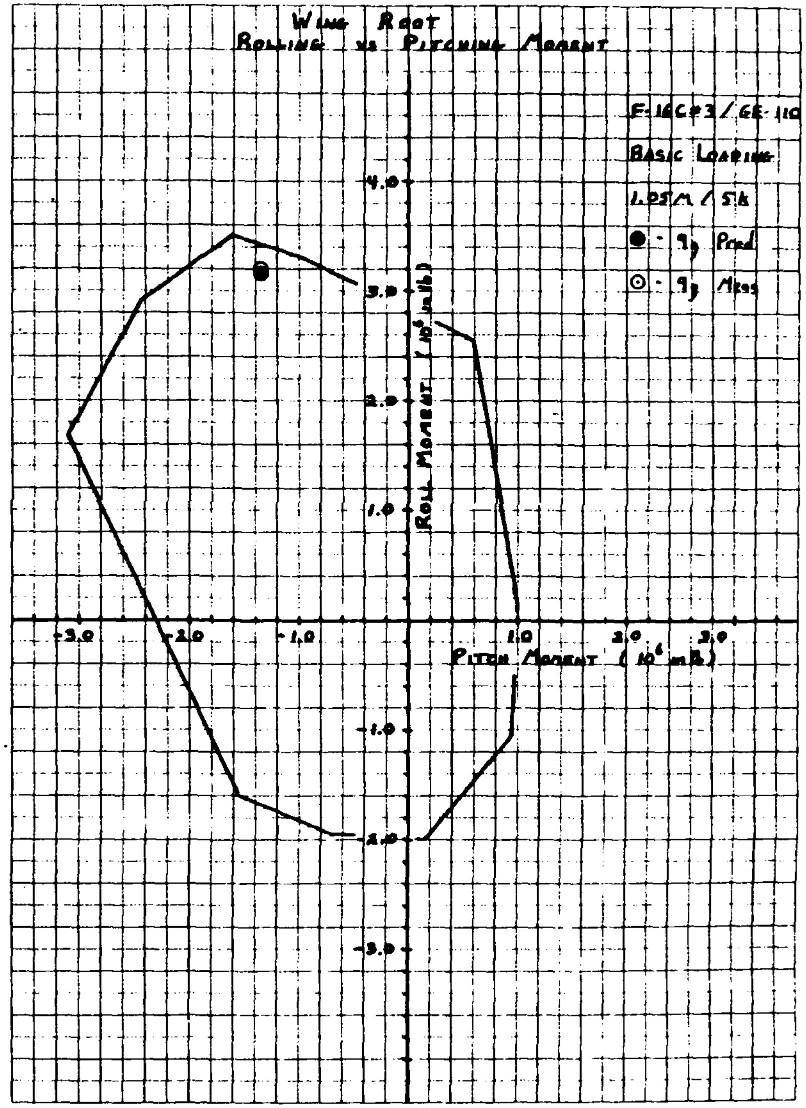
- o Test Aircraft Configuration
  - oo F-16C#3: Blk 25 structure with Common Engine Bay Modification
  - oo F110-GE-100 Engine
  - oo Fully loads instrumented and calibrated
  - oo Essentially a Blk 30 configuration
- o Test Loadings
  - oo BASIC: Clean wing + AIM-9L at Sta 1/9
  - oo BASIC + Full/Empty 370 Gal Tanks at Sta 4/6
  - oo BASIC + 3 CBU-58 on TER at Sta 3/7, 4,6 + ECM at Sta 5
  - oo BASIC + LANTIRN + ECM at Sta 5
  - oo BASIC + LANTIRN + Full/Empty 370 Gal Tanks at Sta 4/6 + 3 CBU-58 on TER at Sta 3/7 + ECM at Sta 5
- o Manuevers were intended to demonstrate Blk 25/30 limit loads for all major aircraft components

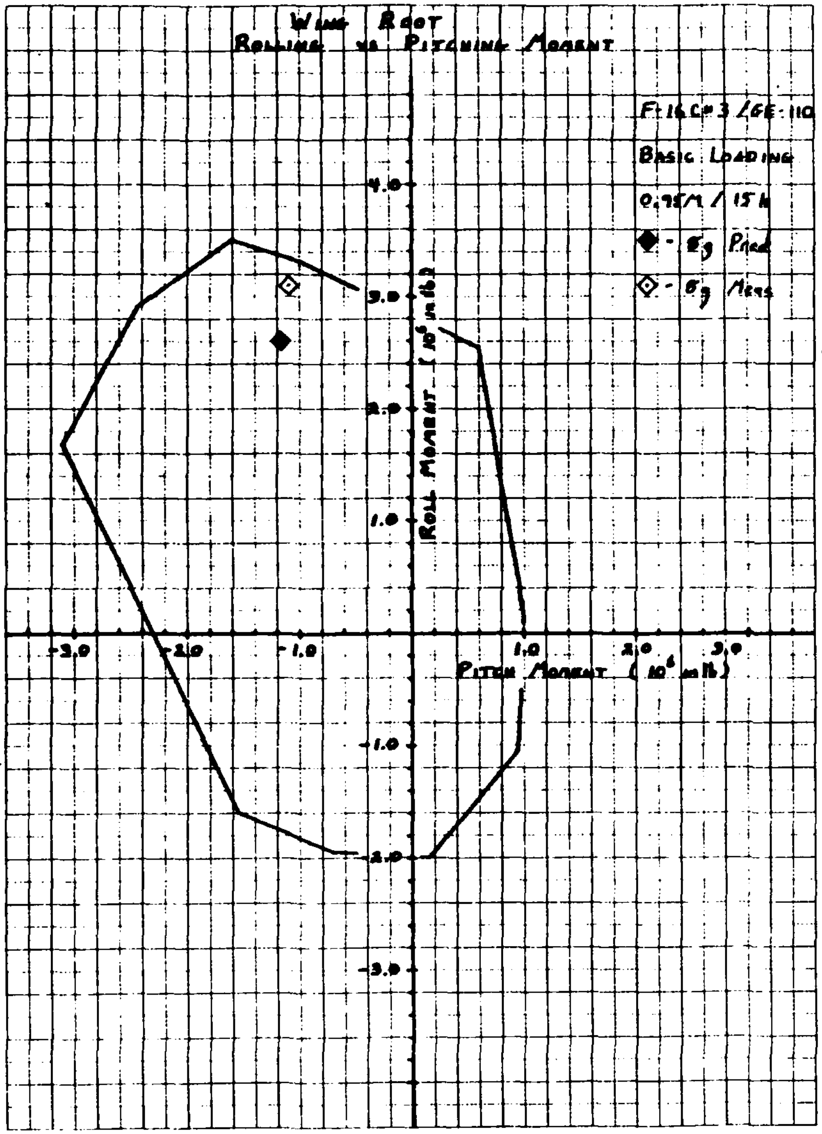
DESCRIPTION OF WING LOADS MEASUREMENT RESULTS

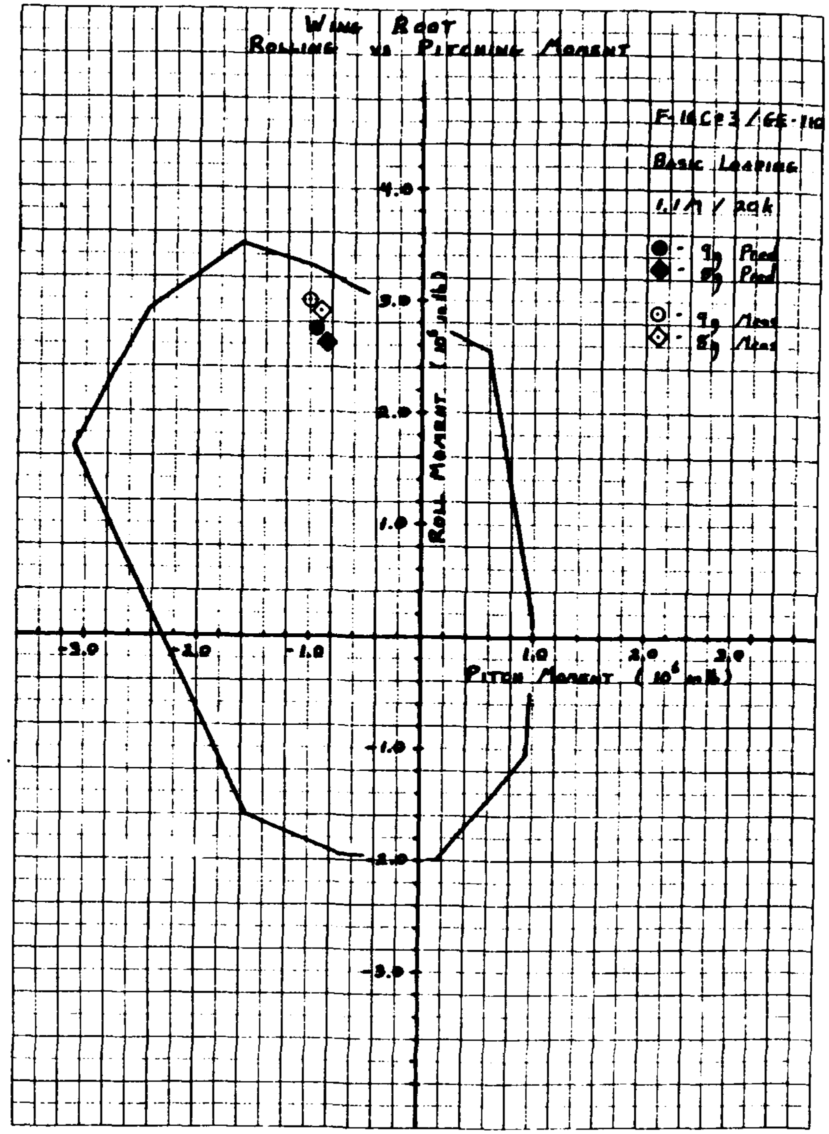
IMPLICATIONS OF HIGHER THAN EXPECTED MEASURED WING LOADS

- MEASURED WING LOADS FOR HIGH-G MANEUVERS WERE HIGHER THAN PREDICTED
- NOT ATTRIBUTABLE TO INSTRUMENTATION PROBLEMS
- NEW CRITICAL DESIGN CONDITIONS WERE IDENTIFIED
  - OLD CRITICAL 9G CONDITIONS:
    - 9G BALANCED PULL UP   0.95M/SL   BASIC
    - 9G BALANCED PULL UP   0.95M/SL   BASIC + EMPTY 370 GAL TANKS
  - NEW CRITICAL 9G CONDITIONS:
    - 9G BALANCED PULL UP   0.95M/10K   BASIC
    - 9G BALANCED PULL UP   0.95M/15K   BASIC
    - 9G BALANCED PULL UP   1.10M/15K   BASIC
- NON-LINEAR STRESS ANALYSES INDICATE ADEQUATE MARGINS OF SAFETY FOR THESE CONDITIONS
- REVISIONS REQUIRED FOR:
  - FULL SCALE STATIC TEST
  - BLK 25/30 LOADS ANALYSES
  - BLK 30 PM II LOADS ANALYSIS


WING ROOT
ROLLING vs PITCHING MOMENT
F-16C#3 / GE-110
BASIC LOADING
0.95M / 5k
● - 9g Pred.
◆ - 8g Pred.
⊙ - 9g Meas.
◇ - 8g Meas.
ROLL MOMENT (10⁶ in lb)
PITCH MOMENT (10⁶ in lb)
4.0
3.0
2.0
1.0
-1.0
-2.0
-3.0
-3.0
-2.0
-1.0
1.0
2.0
3.0


WING ROOT
ROLLING vs PITCHING MOMENT
F-16C#3 / GE-110
BASIC LOADING
1.05M / 5k
● - 9g Pred
⊙ - 9g Meas
ROLL MOMENT (10⁶ in lb)
PITCH MOMENT (10⁶ in lb)
4.0
3.0
2.0
1.0
-1.0
-2.0
-3.0
-3.0
-2.0
-1.0
1.0
2.0
3.0


WING ROOT
ROLLING vs PITCHING MOMENT
F-16C#3 / GE-110
BASIC LOADING
0.95M / 15k
- 8g Pred
- 8g Meas
ROLL MOMENT (10^6 in lb)
PITCH MOMENT (10^6 in lb)


WING ROOT
ROLLING vs PITCHING MOMENT
F-16C#3 / GE-110
BASIC LOADING
1.1M / 20k
- 9g Pred
- 8g Pred
- 9g Meas
- 8g Meas
ROLL MOMENT (10^6 in lb)
PITCH MOMENT (10^6 in lb)

EXPLANATION OF UNEXPECTED FLIGHT TEST RESULTS

- o Flight conditions tested involved the highest NzW values to date.
  - oo NzW values exceed 225000 lb for F-16C/D testing.
  - oo Previous testing produced NzW values no greater than 180000 lb.
- o Flight test data from previous tests indicated a non-linear sloping off of wing root bending with increasing NzW.
- o F-16C/D data showed that the sloping off effect was not as pronounced at higher NzW.
- o Loads analysis procedure incorporated the previous flight test data with the non-linear trends and therefore underpredicted loads for the heavier F-16C/D.
- o Non-linear flight test data trends are probably due to a stiffer than expected wing and a corresponding outward shift in the load C.P. and, thus, higher bending moments.

DESCRIPTION OF VERTICAL TAIL LOADS MEASUREMENT ANOMALIES

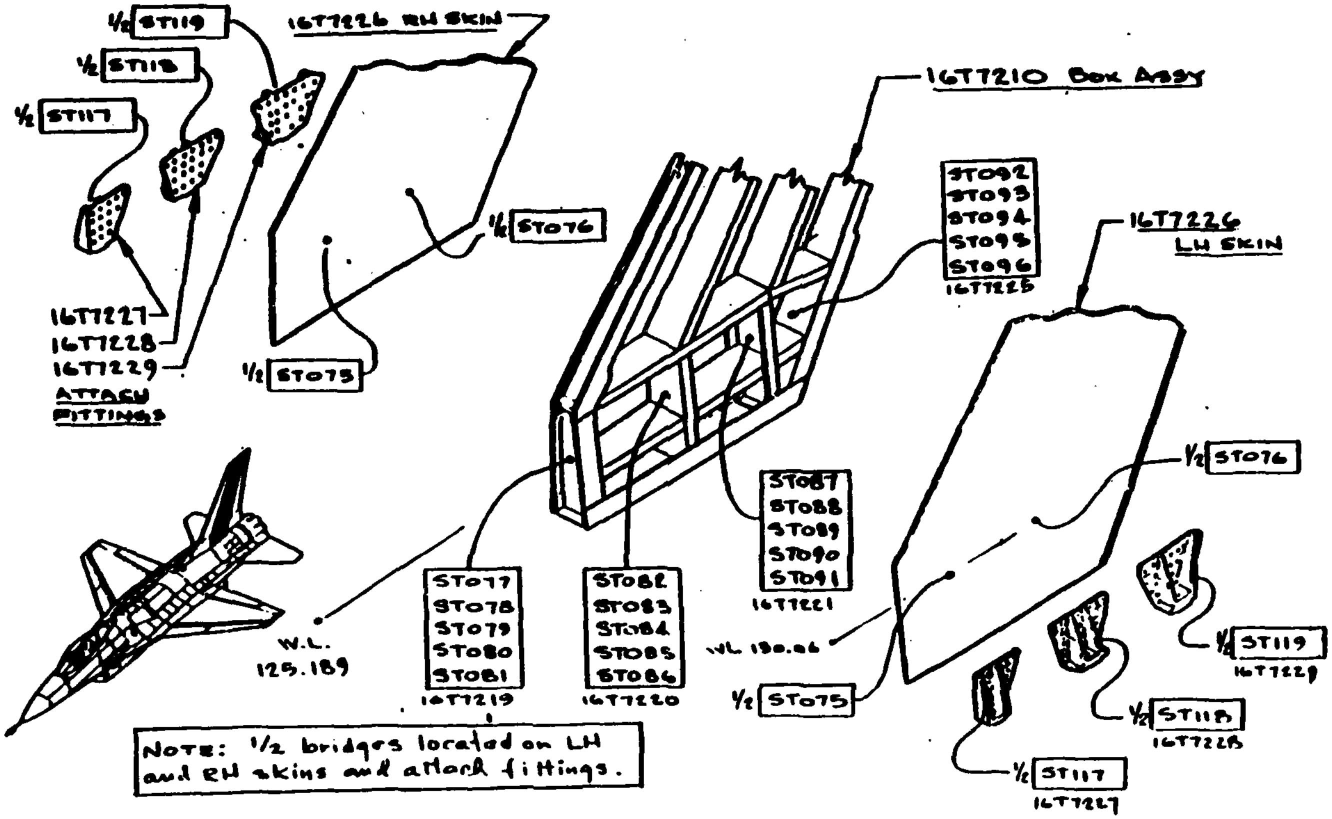
VERTICAL TAIL LOADS

- Vertical tail loads initially measured were greater than expected for sideslip and rolling manuevers
- Found that bending moment instrumentation in rear spar influenced by rudder actuator loads
- New bending moment strain gage combinations, away from rear spar, were activated and test flown
- Loads correlate well with previous flight test results
- Vertical tail load envelopes expanded to allow tolerance for "non-zero" trim loads

FIGURE 3.1.3-2

VERTICAL TAIL LOADS MEASUREMENT INSTRUMENTATION

Root Installation - General Locations



VERTICAL TAIL LOADS MEASUREMENT BRIDGES

BRIDGES SELECTED	Fwd Spar and Ftg	Skin	Ctr Spar and Ftg	Skin	Aft Spar and Ftg	Rear Spar
SHEAR						
Primary		S 075			S 089	
New Primary			S 086	M 076		S 094
Spare			M 118		S 090	
BENDING MOMENT						
Primary			S 085			M 093
Spare			S 084			M 092
New Primary			M 082		M 119	
New Spare			M 118		M 087	
TORSION						
Primary	M 077				M 119	S 095
Spare	S 080					M 092 S 094

S - *Indicates Shear Bridge*

M - Indicates Bending Bridge

DESCRIPTION OF FORWARD FUSELAGE BENDING MOMENT MEASUREMENT ANOMALIES

BACKGROUND

- o Extrapolated forward fuselage loads from flight test exceeded design by 15% - 20%   -Indicated loads highly questionable

  - oo Airplane significantly out of balance --- moments at center of gravity not zero

  - oo Imbalance increases with dynamic pressure (q)

- o Strain gages added to evaluate structure criticality

  - oo Gages located on upper, center, and lower fuselage structure

  - oo Manuever grid repeated

  - oo Strain gage analysis indicated
    - Upper skin panels less effective than anticipated --- stresses less than predicted
    - Substructure stresses greater than predicted but less than limit allowable
    - Internal loads did not support indicated bending moment levels

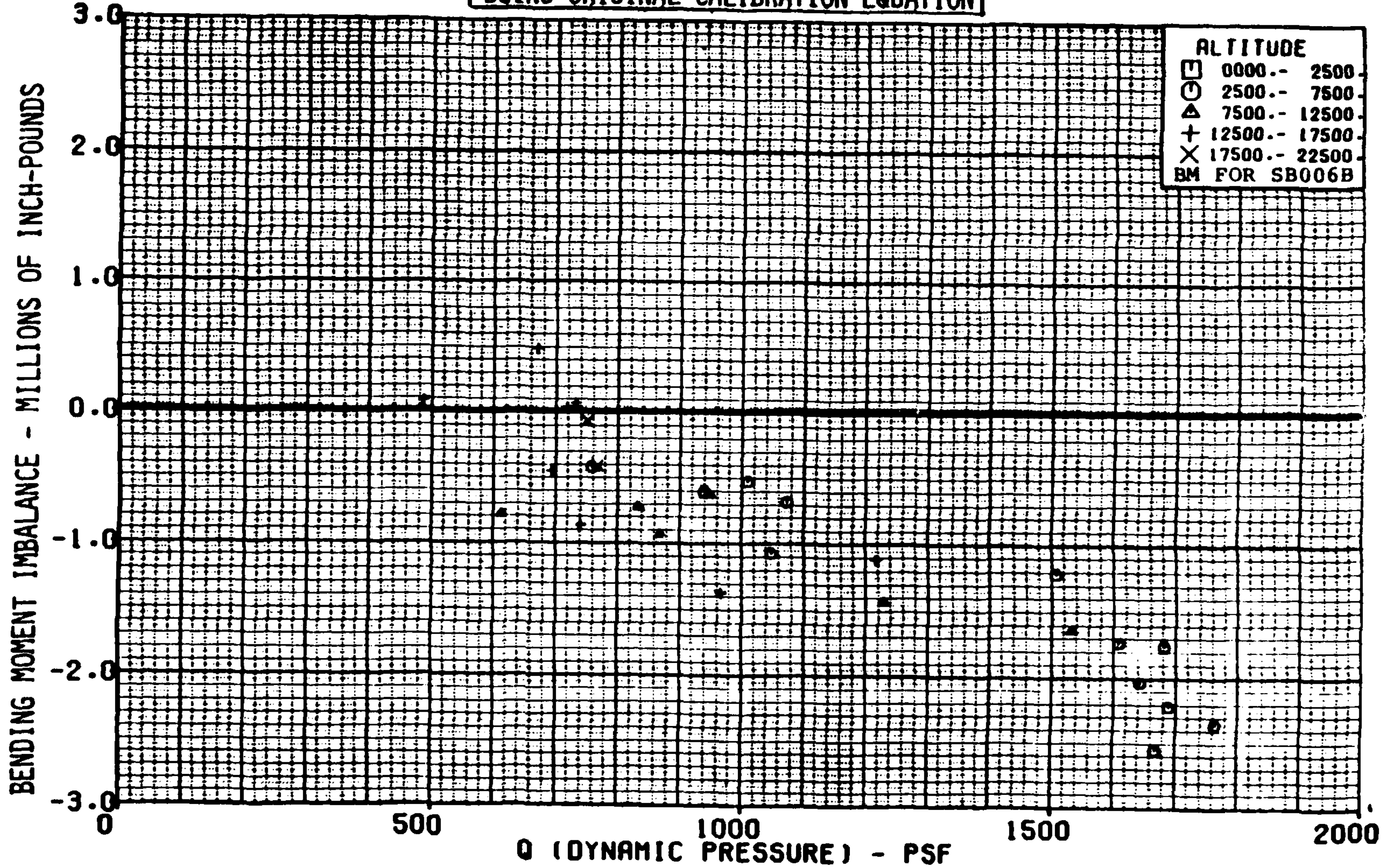
  - oo Reduced panel effectiveness could affect loads instrumentation

F-16C NO. 3

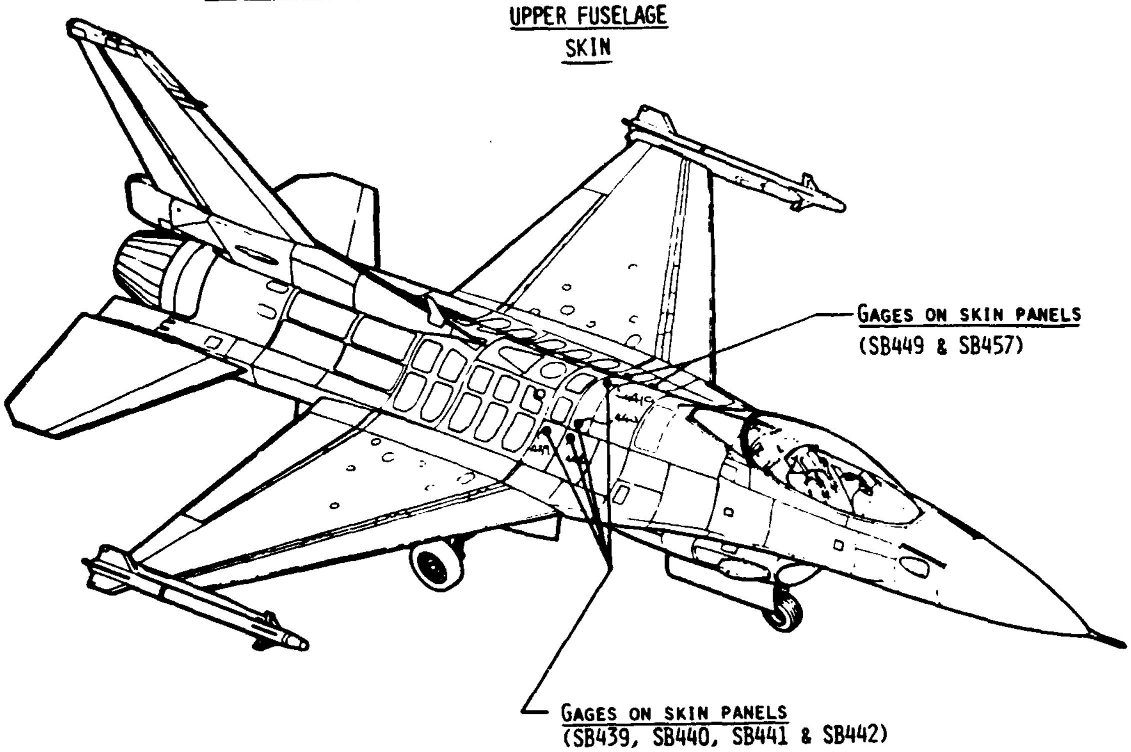
VERTICAL BENDING MOMENT IMBALANCE

AT AIRPLANE CG FOR MAXIMUM MEASURED NZW

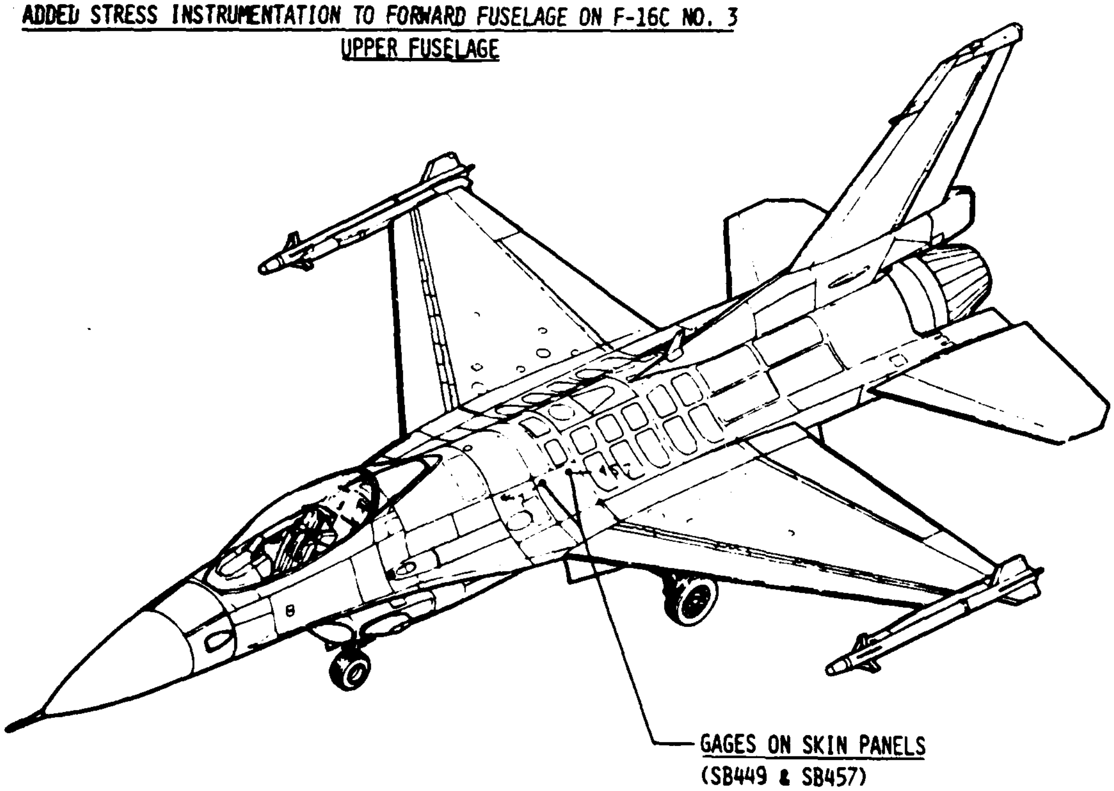
USING ORIGINAL CALIBRATION EQUATION



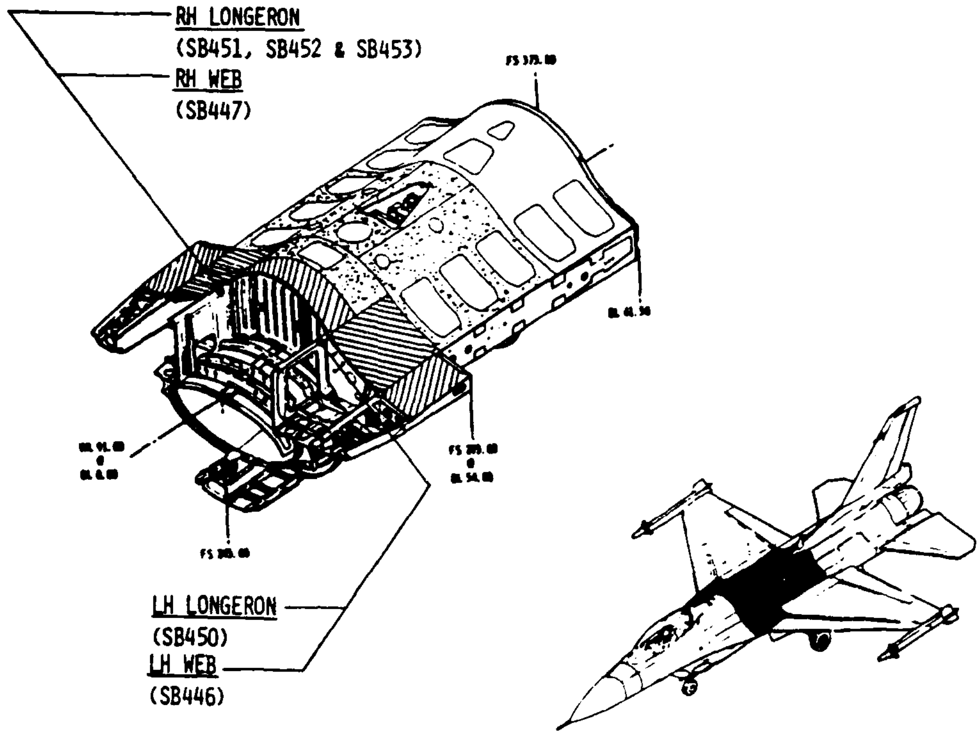
ADDED STRESS INSTRUMENTATION TO FORWARD FUSELAGE ON F-16C No. 3
UPPER FUSELAGE
SKIN



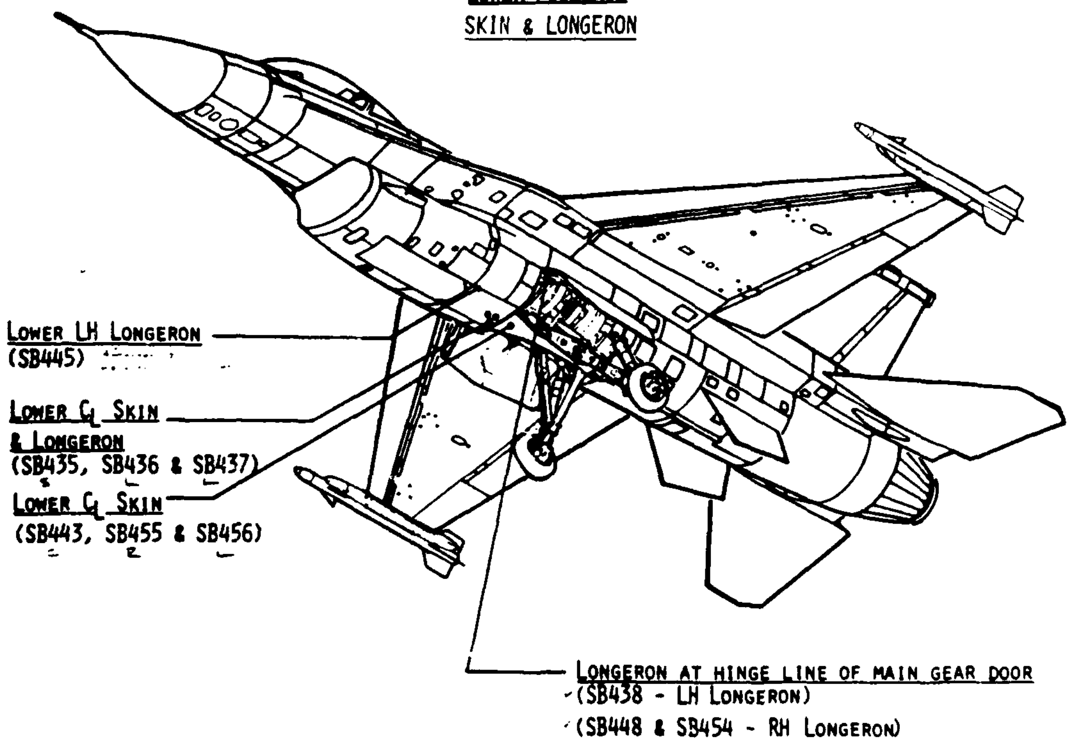
ADDED STRESS INSTRUMENTATION TO FORWARD FUSELAGE ON F-16C NO. 3
UPPER FUSELAGE



ADDED STRESS INSTRUMENTATION TO FORWARD FUSELAGE ON F-16C NO. 3
UPPER FUSELAGE
LONGERON



ADDED STRESS INSTRUMENTATION TO FORWARD FUSELAGE ON F-16C No. 3
LOWER FUSELAGE
SKIN & LONGERON



FUSELAGE RELOADING

- o Purpose
  - oo Determine changes (if any) to the original calibration equations
  - oo Investigate effects of varying skin panel effectiveness
  - oo Obtain relationships of measured stresses to known loads
  - oo Establish method for correcting flight recorded bending moments
  - oo Evaluate potential new moment equations using the reloading data
  - oo Validate performance of the airplane onboard recording system

FUSELAGE RELOADING (Cont'd)

- o Procedures
  - oo Applied both concentrated and distributed loads
    - Repeated 15 of the original calibration concentrated load conditions
    - Added four new distributed load conditions
  - oo Loads applied with various skin panel effectiveness
    - Applied all loads with upper panel attach bolts at max drawing torque
    - Applied one distributed load with upper panel attach bolts at 33% max drawing torque
    - Applied four distrubuted loads with upper panels removed
  - oo Recorded outputs from all forward fuselage stress and loads instrumentation and aft fuselage loads instrumentation

FUSELAGE RELOADING (Cont'd)

o Results

oo Original calibration equations are repeatable for the re-applied loads

oo Loads and stress instrumentation on the upper fuselage structure is affected by upper skin panel attach bolt torques (panel effectiveness)

- Original bending moment equations utilize upper fuselage loads gages combined with lower fuselage gages
- Reducing upper skin panel torques 67% increases the indicated bending moment by 10%
- Removal of the upper skin panels increases the indicated bending moment by 45%

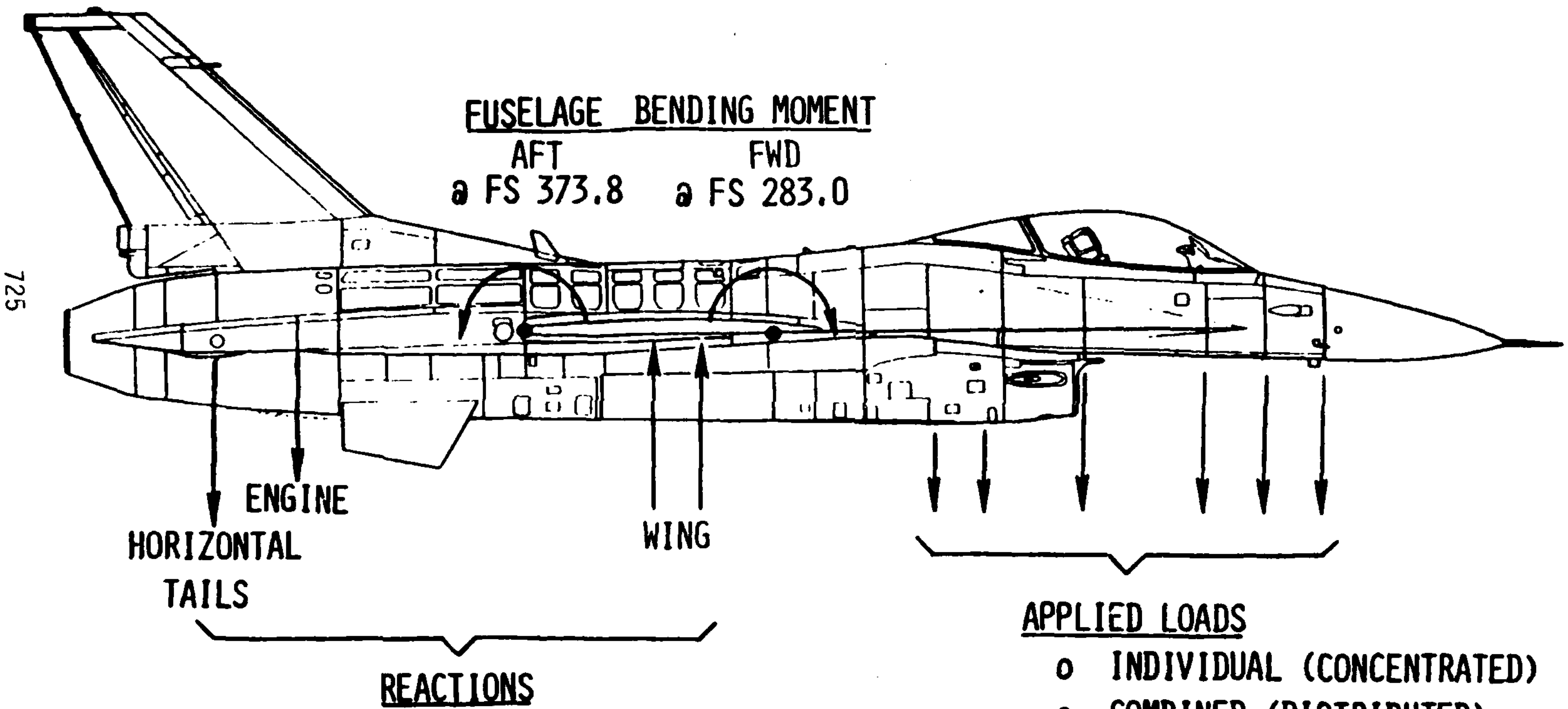
oo Upper fuselage loads and stress gages are affected by location of applied loads

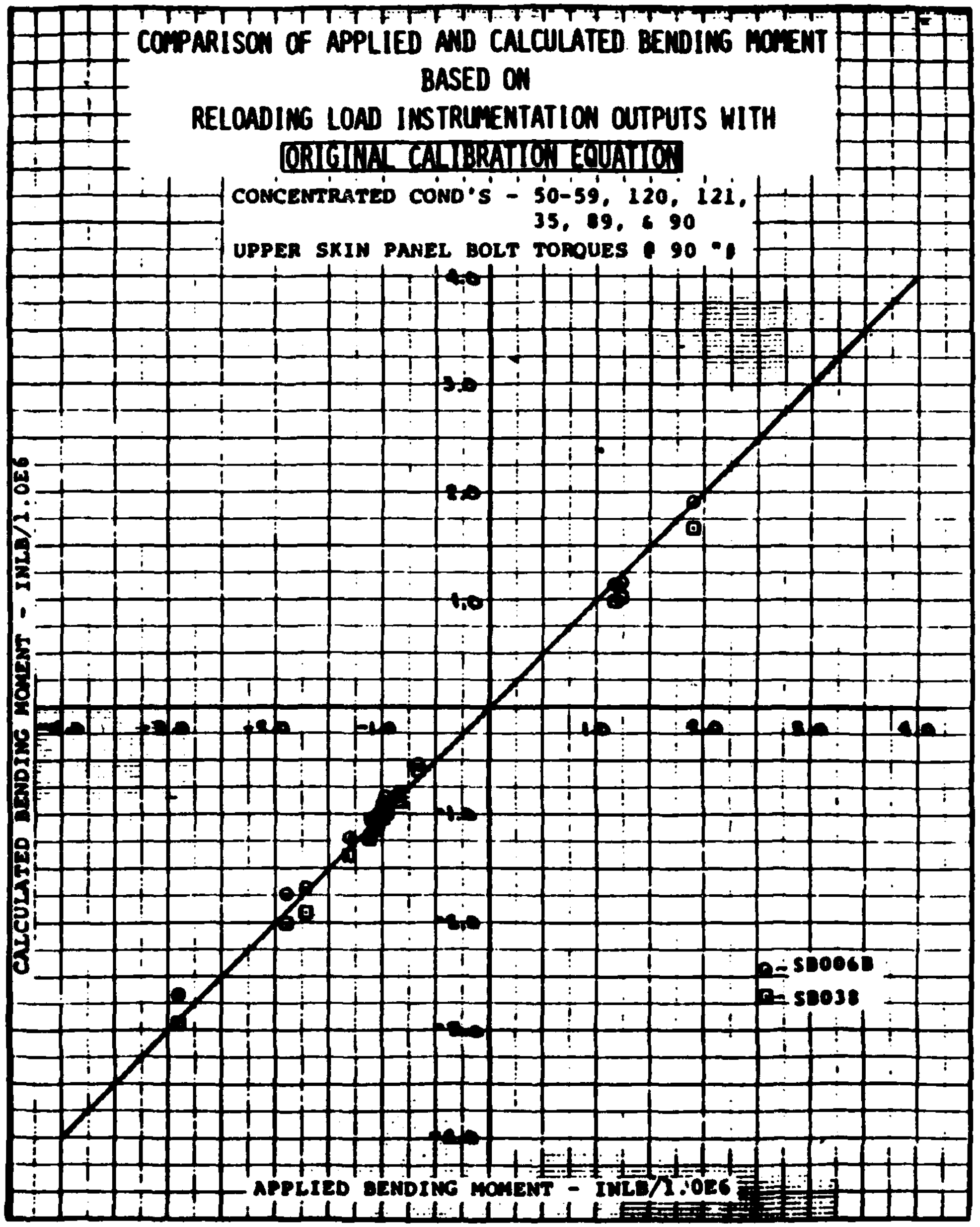
oo Linearity of strain gage response function of gage location

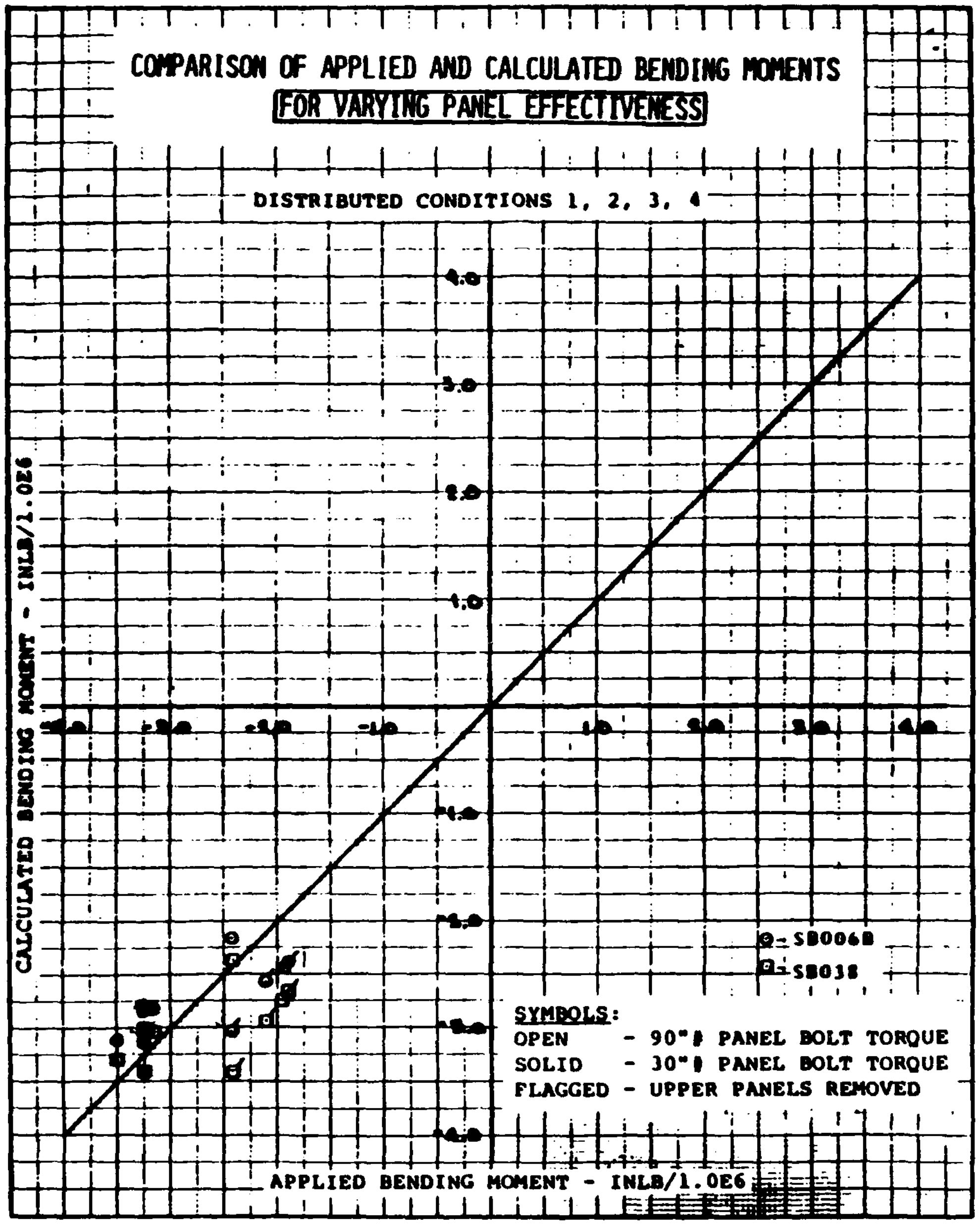
- Upper fuselage stress gage outputs are non-linear with applied load
- All other loads and stress gage outputs are linear with applied load

oo Center and lower fuselage loads and stress gage outputs are independent of the applied load locations and upper skin panel effectiveness

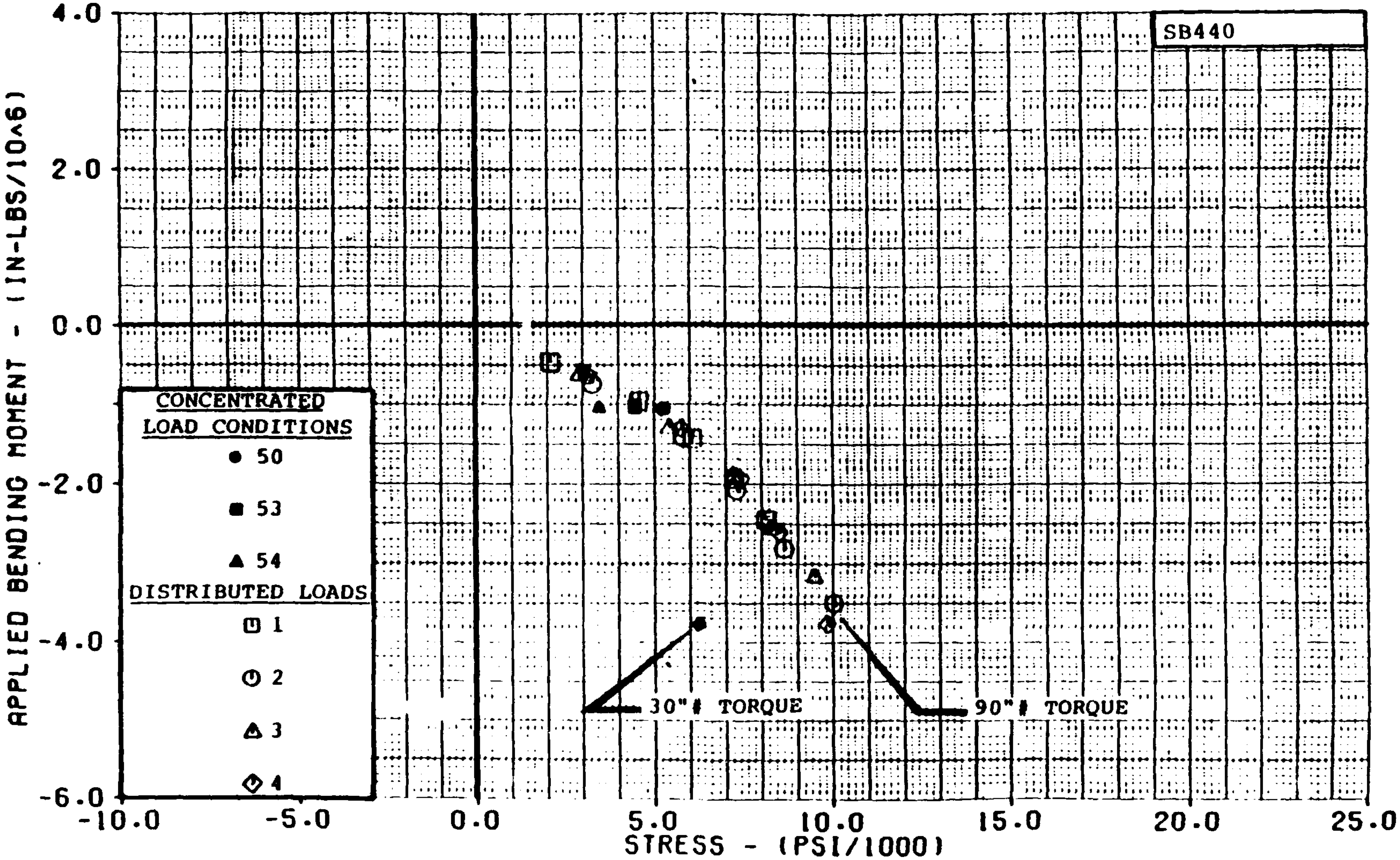
FUSELAGE RELOADING SCHEMATIC

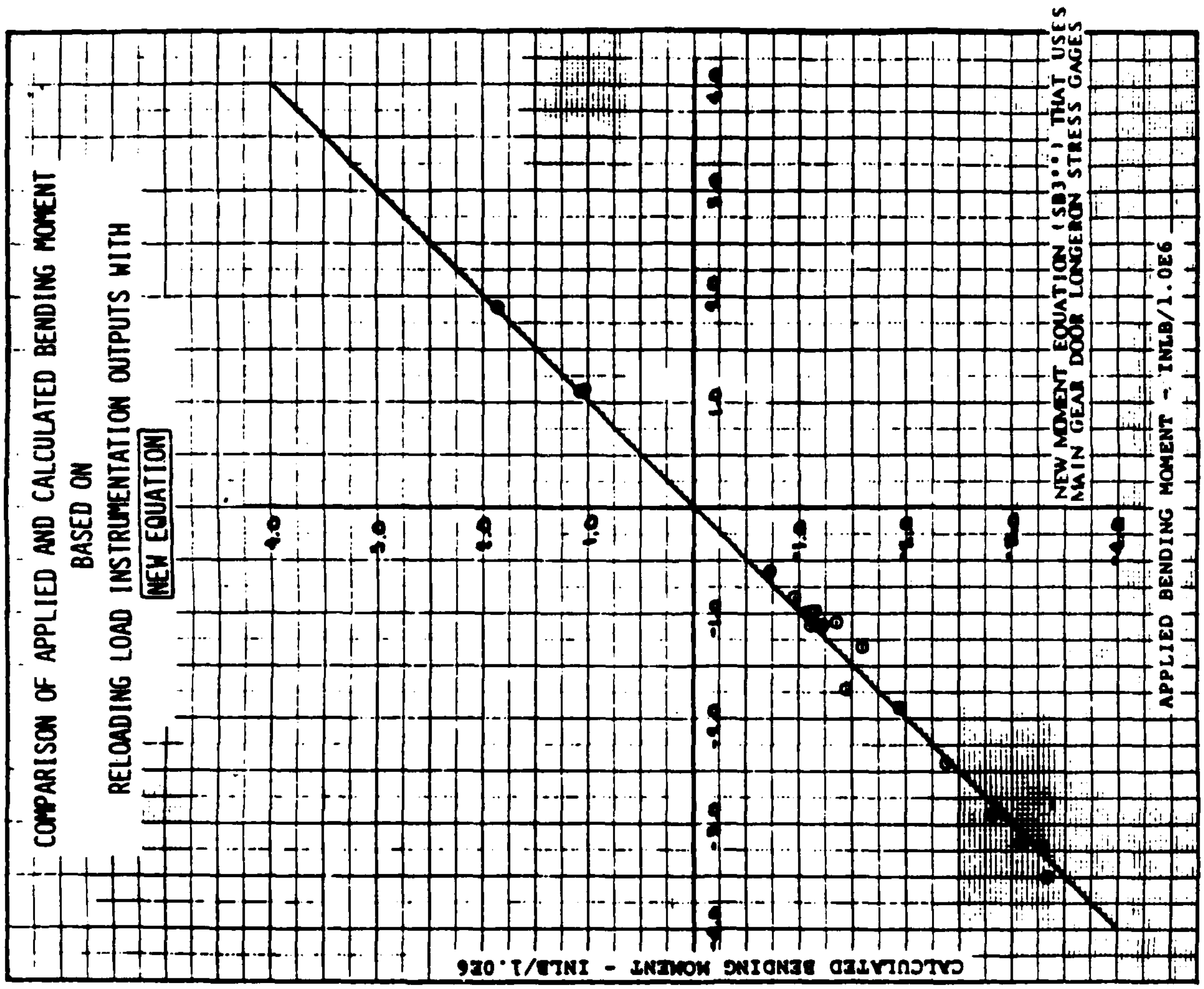



COMPARISON OF APPLIED AND CALCULATED BENDING MOMENT
BASED ON
RELOADING LOAD INSTRUMENTATION OUTPUTS WITH
ORIGINAL CALIBRATION EQUATION
CONCENTRATED COND'S - 50-59, 120, 121,
35, 89, & 90
UPPER SKIN PANEL BOLT TORQUES @ 90 "#
SB006B
SB038
CALCULATED BENDING MOMENT - INLB/1.0E6
APPLIED BENDING MOMENT - INLB/1.0E6
4.0
3.0
2.0
1.0
-1.0
-2.0
-3.0
-4.0


COMPARISON OF APPLIED AND CALCULATED BENDING MOMENTS
FOR VARYING PANEL EFFECTIVENESS
DISTRIBUTED CONDITIONS 1, 2, 3, 4
CALCULATED BENDING MOMENT - INLB/1.0E6
APPLIED BENDING MOMENT - INLB/1.0E6
4.0
3.0
2.0
1.0
-1.0
SB006B
SB038
SYMBOLS:
OPEN - 90"# PANEL BOLT TORQUE
SOLID - 30"# PANEL BOLT TORQUE
FLAGGED - UPPER PANELS REMOVED

EXAMPLE OF FUSELAGE UPPER STRUCTURE STRESS GAGE OUTPUT





COMPARISON OF APPLIED AND CALCULATED BENDING MOMENT
BASED ON
RELOADING LOAD INSTRUMENTATION OUTPUTS WITH
NEW EQUATION
NEW MOMENT EQUATION (SB3**) THAT USES
MAIN GEAR DOOR LONGERON STRESS GAGES
APPLIED BENDING MOMENT - INLB/1.0E6
CALCULATED BENDING MOMENT - INLB/1.0E6

CONCLUSIONS

- o New moment equation is valid
  - o Equation not affected by panel effectiveness
  - o Equation not affected by local aerodynamic pressures
  - o Yields more realistic forward fuselage bending moments both subsonic and especially supersonic
- o Airplance balance is reasonable
- o Stress measurements support new moment equation
  - o Upper skin panel stresses are less than limit allowable
  - o Substructure stresses are greater than predicted but are less than limit allowable

LESSONS LEARNED FROM INVESTIGATION OF INSTRUMENTATION ANOMALIES

- The potential for unanticipated secondary factors to affect flight loads instrumentation is significant.

- Altough the instrumentation calibration procedure attempts to account for these effects, the influence of secondary effects cannot always be anticipated prior to calibration.

- Laboratory calibration environment cannot duplicate actual flight environment with respect to aerodynamic and inertial distributed loads and other secondary effects.

- Careful planning must occur to adequately assess the differences between the laboratory and flight environments so that strain guage bridges can be located for maximum load measurement effectivity while minimizing the potential influences of known secondary factors.

1987 USAF Aircraft/Engine Structural Integrity Program (ASIP/ENSIP) Conference

1-3 December 1987

C-5 Forms Data Automation Via MADAR II

Terry C. Bell
Lockheed

C-5 Forms Data Automation via MADAR II

The subject of this presentation is the proposed forms data automation modification of the C-5 Malfunction Detection, Analysis and Recording (MADAR) System. The proposal addresses hardware and software revisions necessary to incorporate the automation of data acquisition for various standard forms, currently compiled manually by the flight engineer, in the MADAR System.

Proposed system definitions and implemention plans have been performed by Lockheed for San Antonio Air Logistics Center (SA-ALC) with coordination with Headquarters – Military Airlift Command (HQ MAC) and Oklahoma City Air Logistics Center (OC-ALC).

C-5 Forms Data Automation

Via

MADAR II

Lockheed

Outline

What are manual forms data? What purposes do they serve? What is required of the flight engineer? What is MADAR?

Answers to these questions as well as the manner that forms data acquisition and utility can be served by MADAR are addressed herein.

Lockheed

C-5 Forms Data/MADAR II

Outline

- Manually Completed Forms Data:
  - What Are They?
  - What Is Required of Flight Engineer?
- MADAR - What Is It?
- How Can MADAR Help with Forms?
- What Revisions Would Be Required?
- What Would Be the Benefits?

C-5 Forms Data

Forms data are manually completed records of aircraft usage, fuel requirements, performance data, calculated takeoff, emergency return, and landing information, system discrepancies, etc., required to be completed by the flight engineer.

Completion of these 7 separate forms is a significant workload for the flight engineer, often required coincident with other flight duties.

Lockheed

C-5 Forms Data/MADAR II

C-5 Forms Data

- Manually Completed Records of:
  - Aircraft Usage
  - Fuel Requirements
  - Performance Data
  - Calculated Takeoff/Emergency/Landing Data
  - System Discrepancies
  - Etc.
- 7 Separate Forms Completed by Flight Engineer
- Average of Over 400 Separate Entries per Flight

Purpose of Forms

Regardless of the effort, however, forms data acquisition is an essential part of flight operations.

These data are necessary for effective fuel management, safe and effective flight execution, maintenance requirements identification and closure records.

Portions of the data, usage, serve as the basis for Individual Aircraft Tracking and as basic elements of structural analysis, inspection planning, and force management.

Lockheed

C-5 Forms Data/MADAR II

Purpose of Forms

- Essential Part of Flight Operations
- Basis of Individual Aircraft Tracking
- Basic Element of Durability Assessment, Maintenance and Inspection Planning, and Force Management
- Planning/Replanning of Fuel Requirements
- Flight Planning and Execution
- Emergency Contingencies
- Maintenance Requirements

MAC Form 89

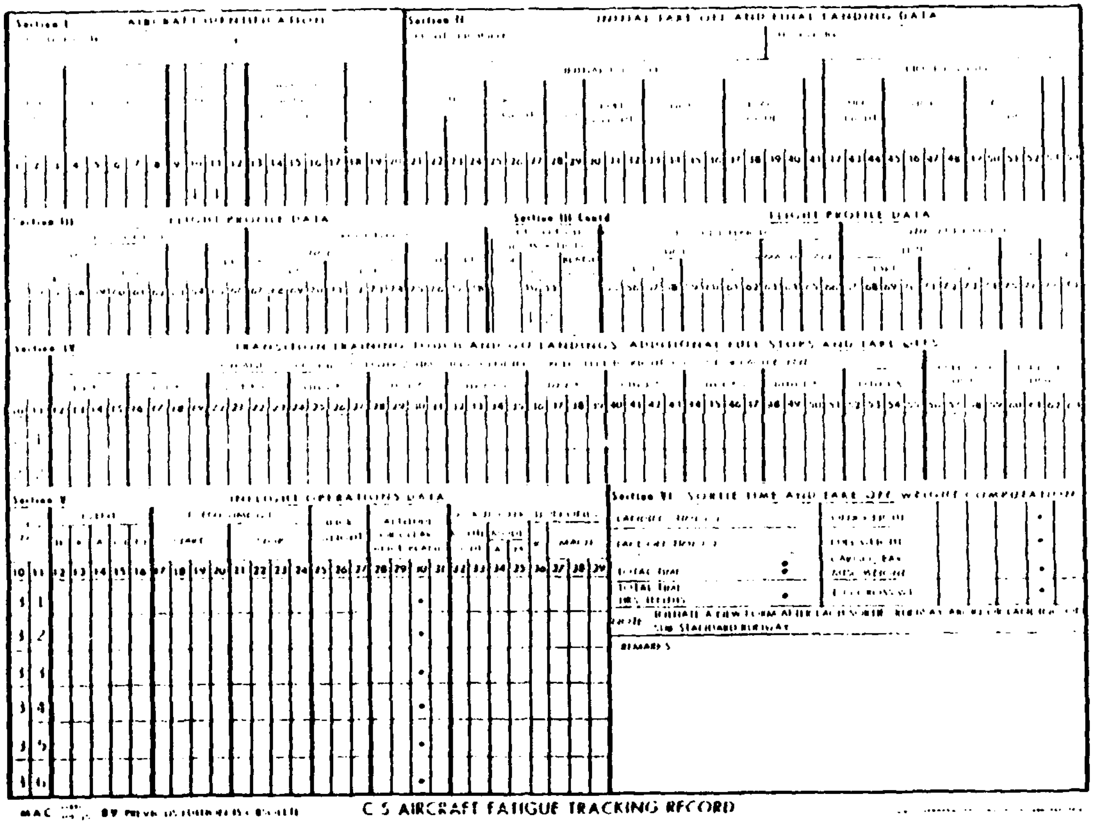
The MAC Form 89, C-5 Aircraft Fatigue Tracking Record, is utilized to record selected aircraft and flight parameter information to obtain a history of the aircraft usage and exposure to actions which contribute to structural fatigue.

These data are the basis of the C-5 Individual Aircraft Tracking Program and a basic element of durability assessment, maintenance and inspection planning, and force management.

The Air Force requires 100 percent yield for this form; i.e., a MAC Form 89 document must be completed and included within the analysis for each sortie.

C-5 Forms Data/MADAR II

Current MAC Form 89



MAC Form 95

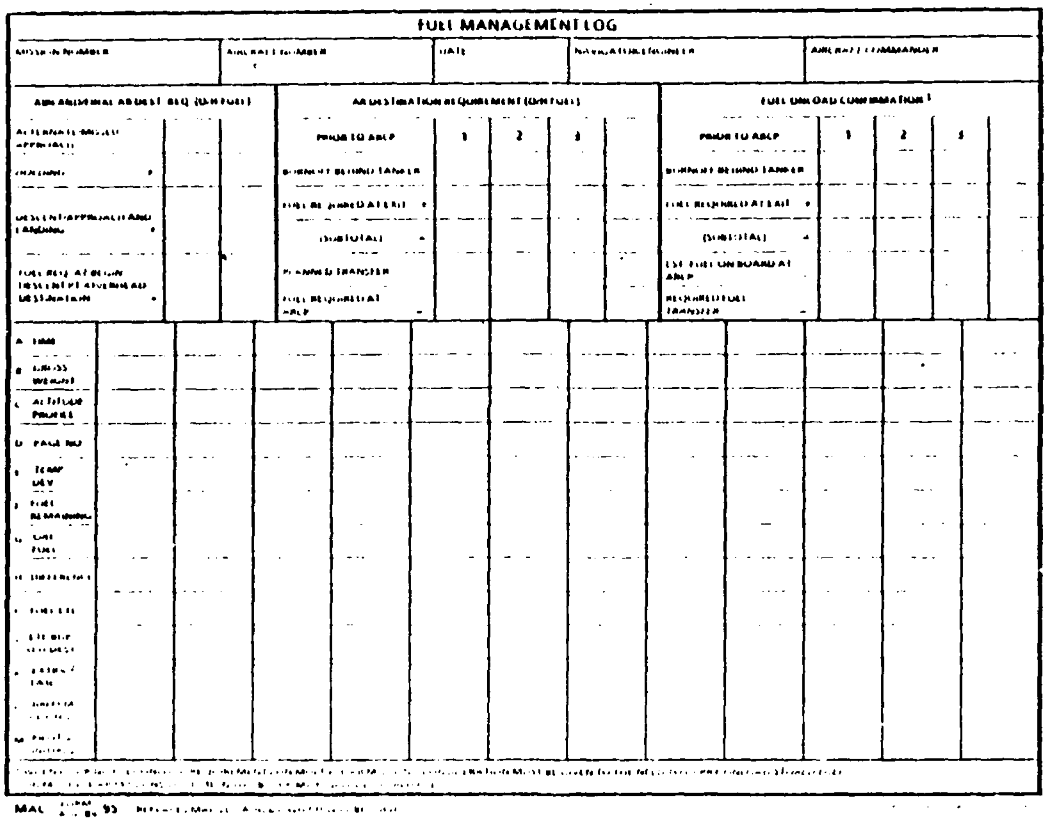
The MAC Form 95, Fuel Management Log, is utilized for inflight planning and replanning of aircraft fuel requirements on selected (Category 1 Route) missions. The form is required when the flight plan time between suitable enroute airfields, within 50 miles of course line, exceeds 5 hours.

Additionally, through coordination with the Air Force, desired elements of the AF Form 796, Aircraft Performance Log/Plan, are planned for inclusion with the revised, automated MAC Form 95. This will provide a record of selected atmospheric conditions and aircraft performance parameters throughout the mission and eliminate the need for a separate AF Form 796.

![Lockheed]

C-5 Forms Data/MADAR II

Current MAC Form 95


FUEL MANAGEMENT LOG

MISSION NUMBER | AIRCRAFT NUMBER | DATE | NAVIGATOR/ENGINEER | AIRCRAFT COMMANDER

DESTINATION REQUIREMENT (ON FUEL)

PRIOR TO ARCP | 1 | 2 | 3

BURNOFF BEHIND TANKER

FUEL REQUIRED AT EXIT

(SUBTOTAL)

PLANNED TRANSFER

FUEL REQUIRED AT ARCP

FUEL ONLOAD CONFIRMATION

PRIOR TO ARCP | 1 | 2 | 3

BURNOFF BEHIND TANKER

FUEL REQUIRED AT EXIT

(SUBTOTAL)

EST FUEL ON BOARD AT ARCP

REQUIRED FUEL TRANSFER

A TIME

B GROSS WEIGHT

C ALTITUDE PROFILE

D PAGE NO

E TEMP DEV

F FUEL REMAINING

MAC FORM 95

MAC Form 100

The MAC Form 100, Takeoff and Landing (TOLD) Card Worksheet (C-5), is utilized to compute and record takeoff, emergency return and destination landing data. Resulting predicted takeoff and landing performance data, based on aircraft configuration and meteorological conditions, are utilized for mission planning and execution.

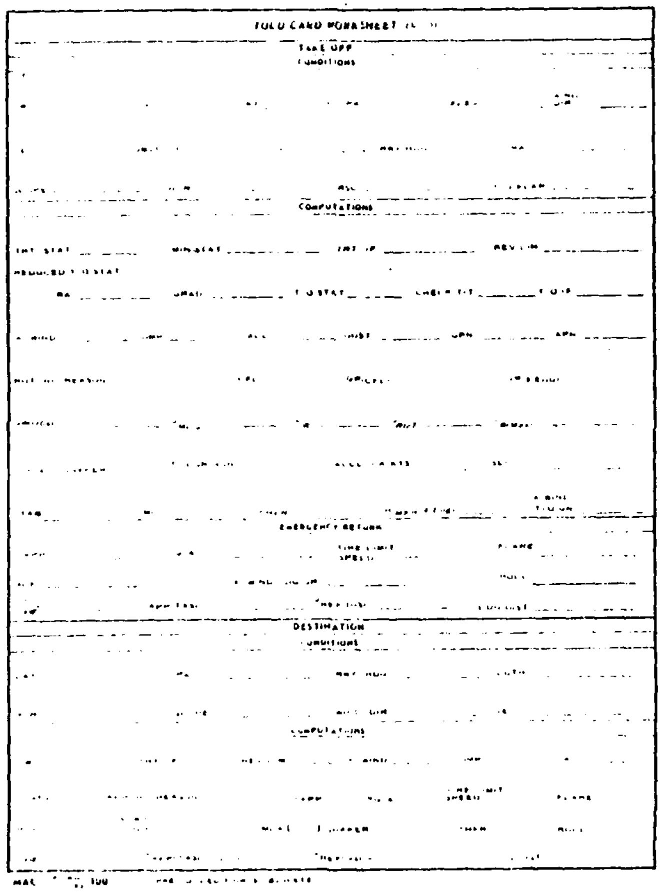
This form is also the basis for the MAC Form 100A, TOLD Card, which is directly utilized by the pilot during the mission for reference purposes.

The MAC Form 100 is judged by HQ MAC to be the most time consuming of the required forms and would provide the most benefit to the flight engineer by its automation.

Lockheed

C-5 Forms Data/MADAR II

Current MAC Form 100


TOLD CARD WORKSHEET

TAKE OFF CONDITIONS

COMPUTATIONS

EMERGENCY RETURN

DESTINATION CONDITIONS

COMPUTATIONS

AFTO Form 781A/K

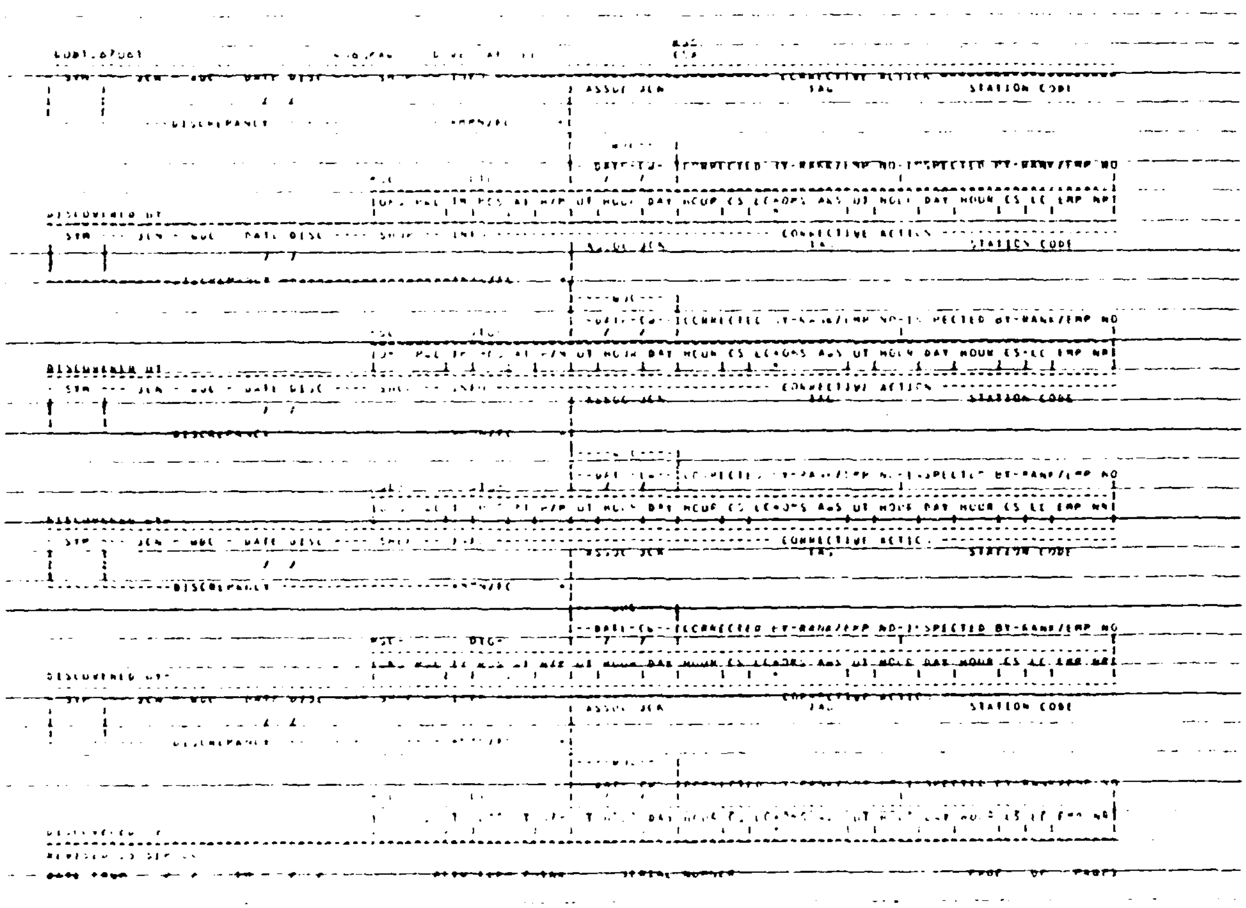
The AFTO Form 781 A/K, Maintenance Discrepancy and Work Document, is utilized to record aircraft system discrepancies discovered before, during and after flight, as well as during selected maintenance inspections.

This form is also used to record corrective action performed for each discrepancy. However, corrective action entries will not be automated since maintenance is typically performed without ship's power or MADAR operation.

Corrective actions will continue to be manually entered into the AFTO 781 log and merged with the automated data through the GPS.

C-5 Forms Data/MADAR II

Current AFTO Form 781 A/K



C-5 Flight Engineer

Lockheed

C-5 Forms Data/MADAR II

C-5 Flight Engineer

- Aircraft Systems Manager?

or

- Aircraft Bookkeeper?

C-5 MADAR System

The C-5 Malfunction Detection, Analysis, and Recording (MADAR) system is an onboard computerized avionics system which monitors the condition of many aircraft systems (engines, structural, avionics, etc).

The MADAR monitors 801 test points and interfaces with the various aircraft systems and central air data computers

Its purposes are to identify and report system malfunctions, display test point data in real time, display fault isolation and troubleshooting information, and record structural, engine, and other trend data for ground processing.

C-5 Forms Data/MADAR II

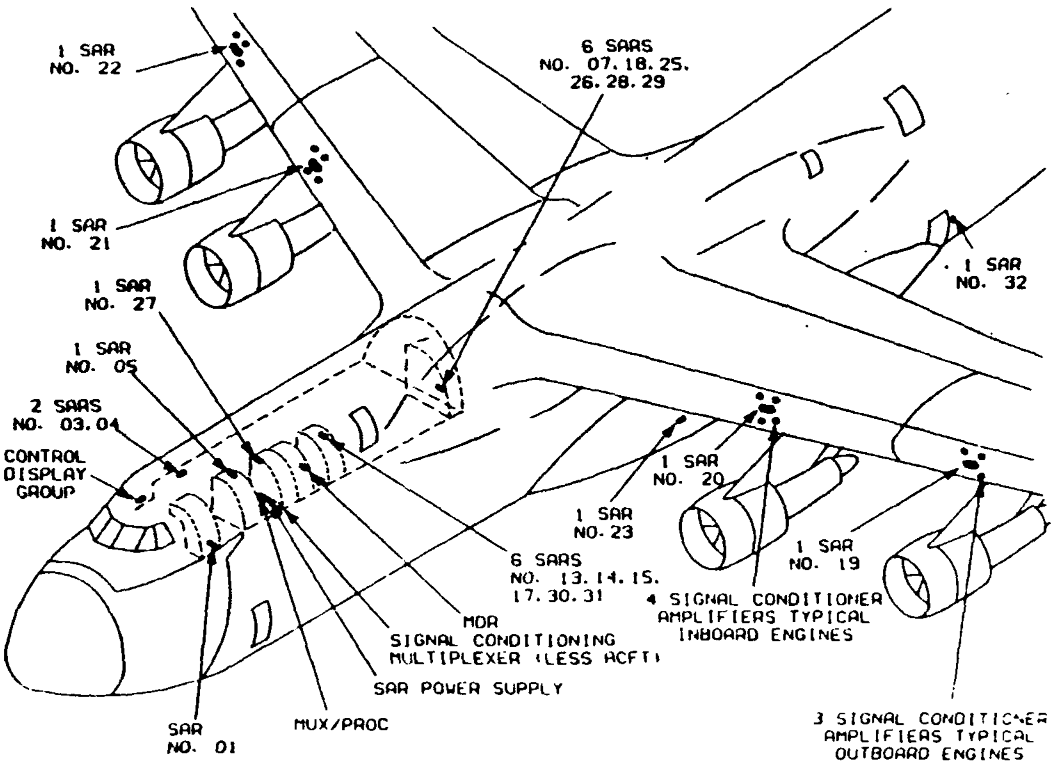
C-5 MADAR System

- Malfunction Detection, Analysis, and Recording
- Onboard Computerized Avionics System
- Monitors 801 Test Points
- Interfaces Aircraft Systems/Central Air Data Computers
- Identifies System Malfunctions/Trends
- Displays Data in Real Time
- Displays Fault Isolation/Troubleshooting Information
- Records Structural, Engine, and Other Data for Ground Processing System (GPS)

Lockheed

C-5 Forms Data/MADAR II

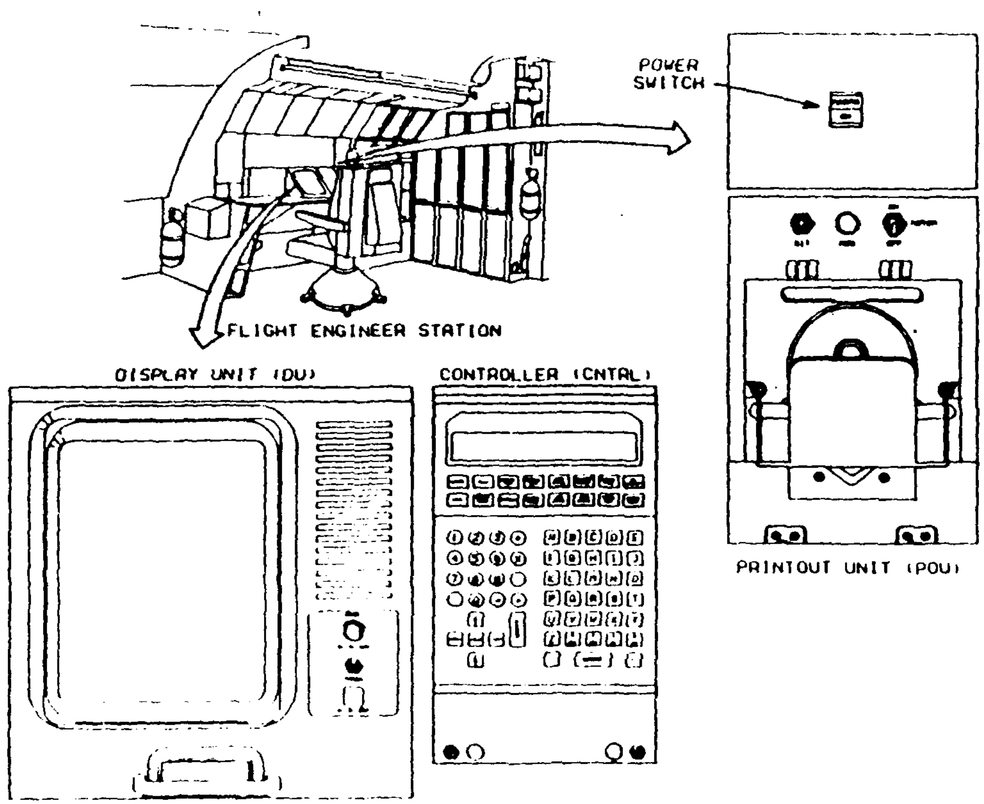
MADAR II
Component Locations



C-5 Forms Data / MADAR II

Lockheed

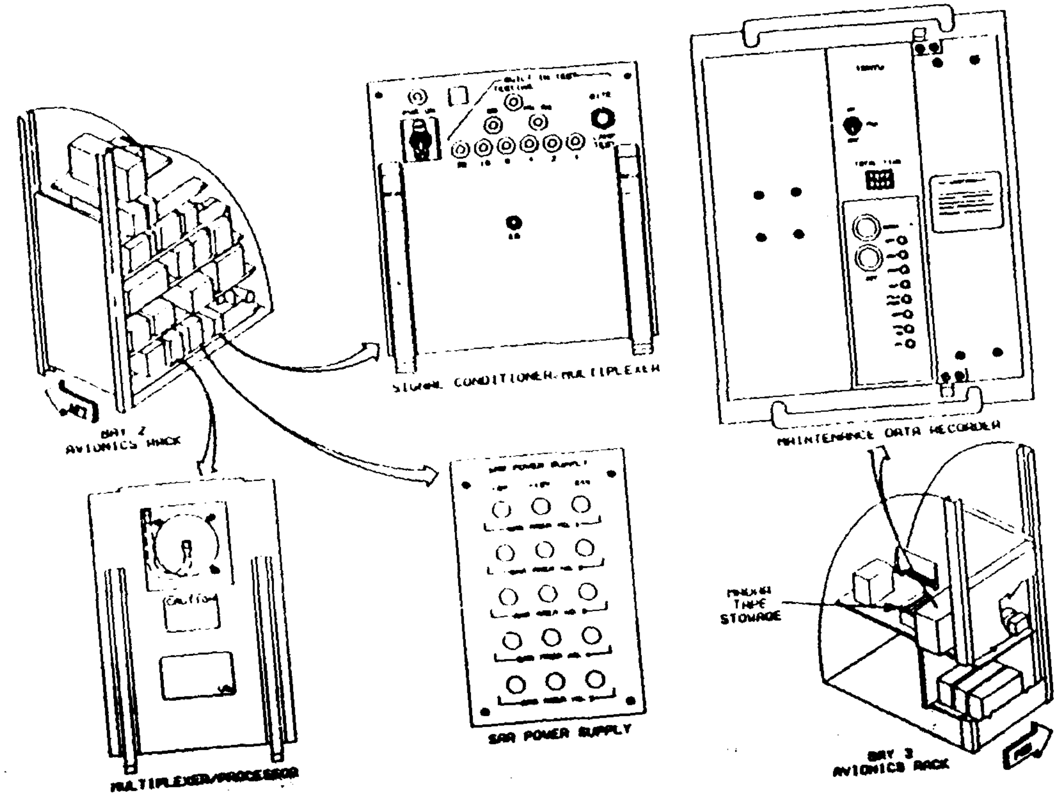
MADAR II Control / Display



Lockheed

C-5 Forms Data/MADAR II

MADAR II Processing/Storage



Existing MADAR Capabilities for Forms Data

Capability exists within the MADAR system to automate the vast majority of data acquisition required by forms data.

Parameters required for most of the forms data are already monitored by MADAR.

The computation power for the many calculations currently performed manually exists in MADAR.

Operator interface exists through the Display unit and Controller unit.

Storage capability exists in the MADAR Maintenance Data Recorder (MDR) for the forms data required for subsequent ground analysis and the interface with the Ground Processing System for automated MDR data access exists.

Communication ports already exist on the Mux/Proc and Controller units for printer and keyboard interface.

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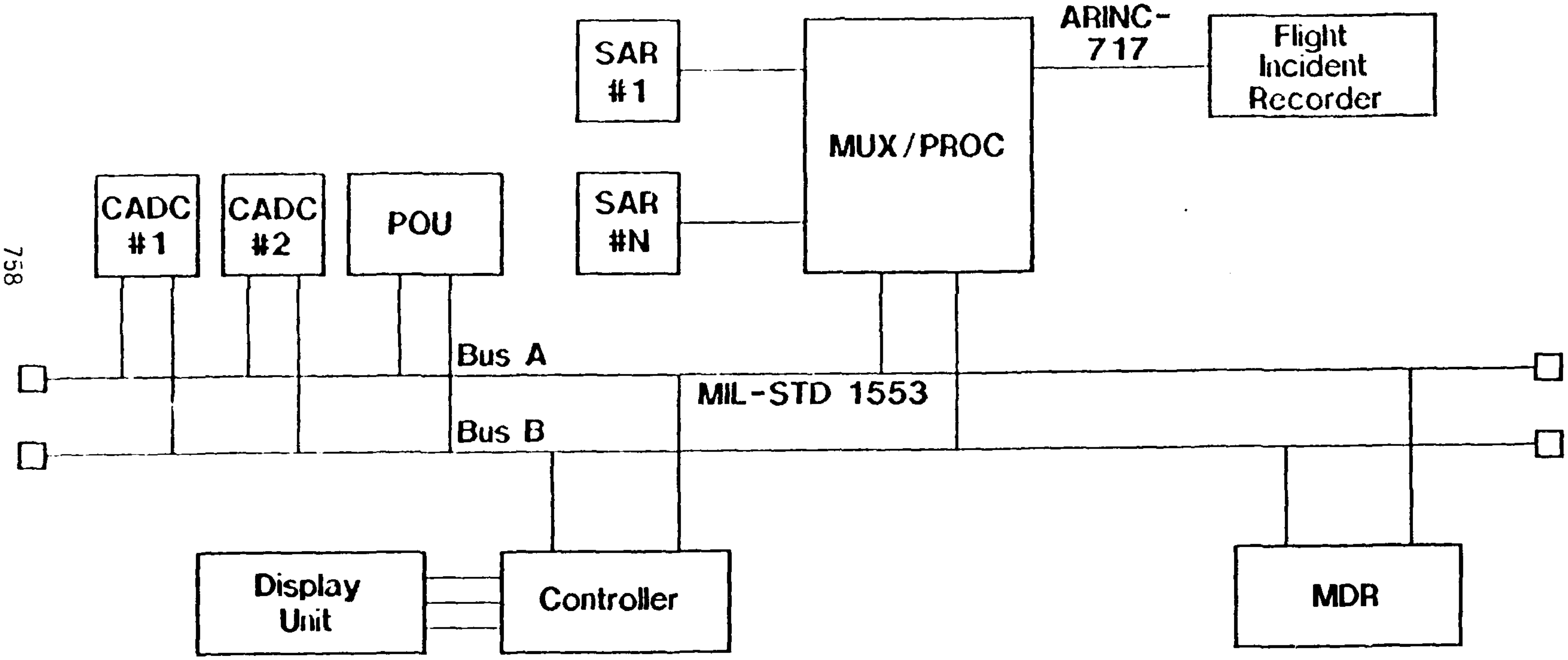
C-5 Forms Data/MADAR II

Existing MADAR Capabilities for Forms Data

- Parameters Needed for Most Forms Data Already Monitored
- Computation Power To Perform Current Manual Calculations
- Display Unit/Controller Unit for Operator Interface
- Maintenance Data Recorder for Data Storage
- Ground Processing System (GPS) Interface for Automated Data Flow
- Communication Ports for Printer and Keyboard

C-5 Forms Data/MADAR II

Current Block Diagram



Why Now?

Automation of usage data acquisition, one of the standard forms data, was investigated in the early 1970s; however, MADAR capabilities at that time were not compatible with related processing/storage requirements.

Major revisions to the MADAR were incorporated for the C-5B, MADAR II, providing the capability for forms data automation.

Retrofit of the 77 C-5A aircraft with the MADAR II system is scheduled for completion by early 1990.

C-5 Forms Data/MADAR II

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Why Now?

- Usage Data Automation Investigated in Early 1970s
- Original MADAR Capabilities, C-5A, Not Compatible With Forms Data Processing/Storage Requirements
- With MADAR II, C-5B, Capabilities Now Exist
  - Greatly Increased Data Processing/Storage Capabilities
  - Graphics Display Capability
  - Operator Interaction
- Retrofit of C-5A With MADAR II To Be Completed by 1990

MADAR Revisions Required

With moderate MADAR II modifications, forms data automation can be accomplished.

The hardware revisions required are the addition of a wide carriage printer for forms data printing and a total fuel test point on 45 C-5B aircraft. This test point is already accessed on the 5 C-5B LESS aircraft and all C-5A aircraft.

An optional typewriter keyboard addition would facilitate text input; however, the existing controller unit keys could be utilized for all forms input required.

Onboard software revisions would be required to control the acquisition and ground processing software would require revision to accommodate the stored forms data.

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C-5 Forms Data/MADAR II

MADAR Revisions Required

- With Moderate MADAR II Modifications, Forms Data Automation Can Be Realized
- Hardware:
  - Wide Carriage Printer
  - Total Fuel Test Point (45 C-5B Aircraft)
  - Optional Typewriter Keyboard
- Software:
  - Onboard Mux/Proc and Controller
  - Ground Processing System Elements

Proposed Configurations

Two proposed configurations, affecting hardware and software, have been coordinated with the Air Force and defined for Air Force consideration and selection: the Basic and Optional.

The Basic configuration is comprised of the hardware and software revisions, services and documentation required to install a wide carriage, dot matrix printer in a remote location (Avionics Bay No. 3) and to provide the necessary interfaces with the MADAR II system.

The Optional configuration is comprised of the basic plus the related efforts to install a portable, ruggedized typewriter style keyboard to the flight engineer station and provide the required interfaces with the MADAR II system.

The Basic configuration will provide full capability for the subject forms data, however, the Optional configuration will provide optimum capability for text input, and also provide interchangeability with controller keys, instructor accessibility and future expansion capability.

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C-5 Forms Data/MADAR II

Proposed Configurations

- Basic Configuration
  - Wide Carriage Dot Matrix Printer
  - Avionics Bay Location
  - Interfaces With MADAR System Mux/Proc
  - Full Forms Capability via Controller Keypad Input
- Optional Configuration
  - Includes Basic Configuration

    Plus
  - **Portable, Ruggedized Typewriter Style Keyboard**
  - **Modified Flight Engineer Station**
  - **Interfaces With MADAR System Controller**
  - **Optimum Capability for Text Input**
  - **Controller Key Interchangeability, Instructor Accessibility, Future Expansion Capability**

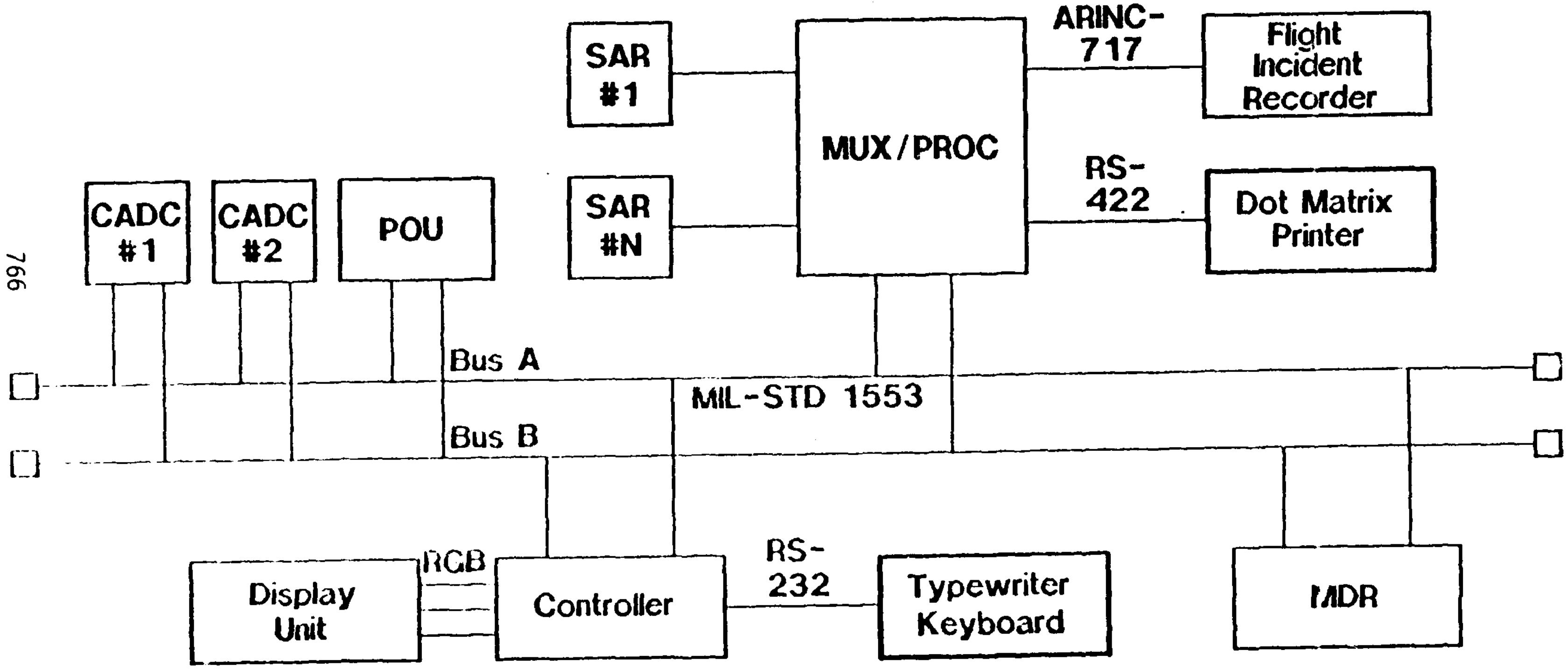
System Block Diagram

The forms data acquisition will be performed through the use of the existing MADAR test points, additional fuel quantity test point on certain C-5B A/C, and minimized operator input.

The interface of the added printer and keyboard will be accomplished through the existing RS-422 communication port on the MADAR Mux/Proc and the existing RS-232 communication port on the MADAR Controller.

C-5 Forms Data/MADAR II

System Block Diagram



Software/Firmware Updates

The appropriate onboard programs (Mux/Proc and Controller) will be revised to control the acquisition, display, printing, keyboard input, and recording of the forms data.

Likewise, appropriate ground MADAR programs, Ground Processing System (GPS) and Automated Maintenance System (AMS) programs will be revised to accommodate the expanded data and streamlined data flow.

Also, other software updates may be desired within Air Force functions to channel the available data to the Information Management System (IMS).

Appropriate documentation of these software updates will be provided.

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C-5 Forms Data/MADAR II

Software/Firmware Updates

- Onboard Programs
  - Control Acquisition, Display, Printing and Recording
- Ground Programs
  - Accommodate Expanded Data and Streamlined Data Flow
  - Channel Data to Management Information System

MAC Form 89 Data Flow

Recording of the MAC Form 89 data by the Maintenance Data Recorder (MDR), with preliminary editing within MADAR, would eliminate the redundant usage forms transcription and subsequent keypunching into the computer system.

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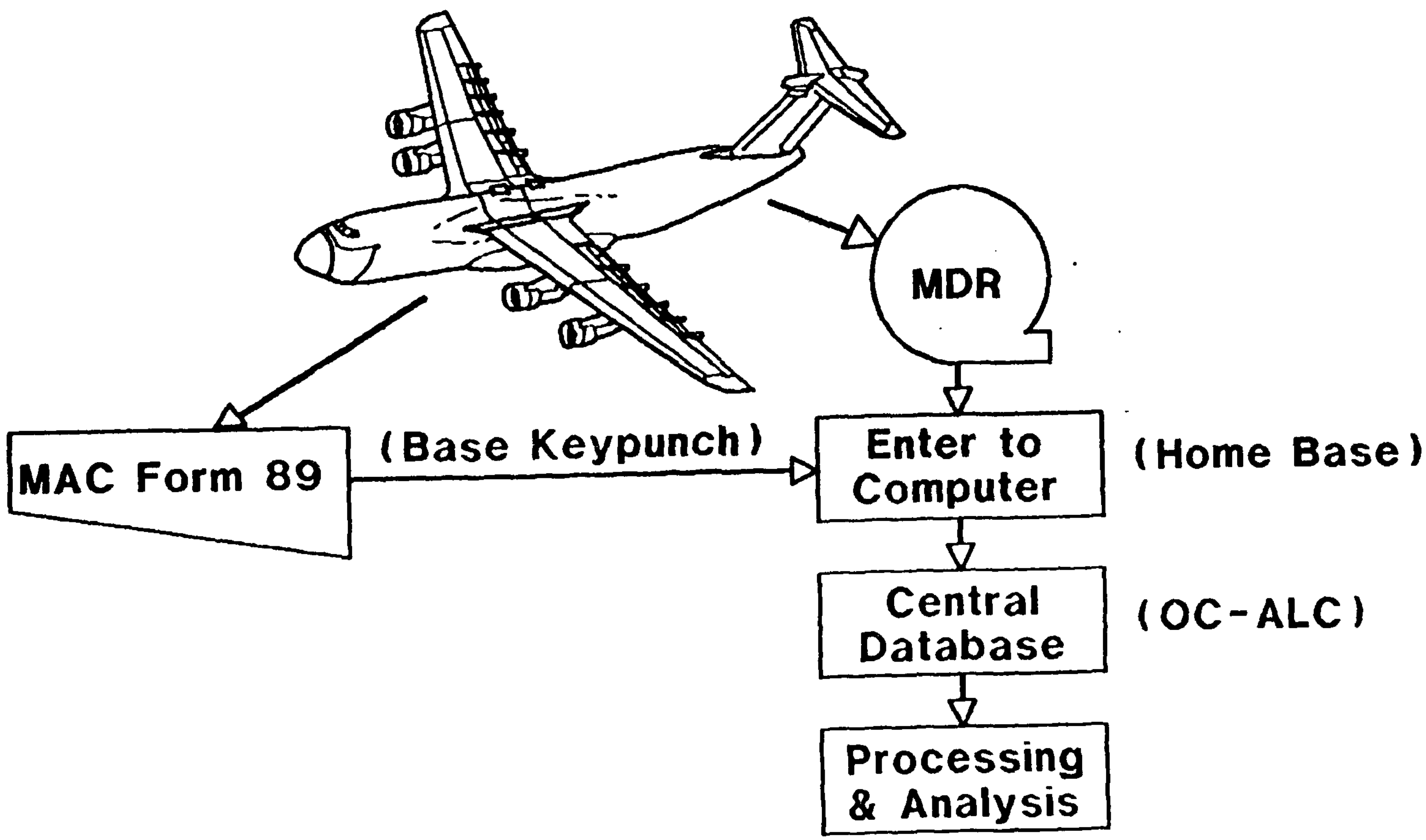
C-5 Forms Data/MADAR II

Current MAC Form 89 Data Flow

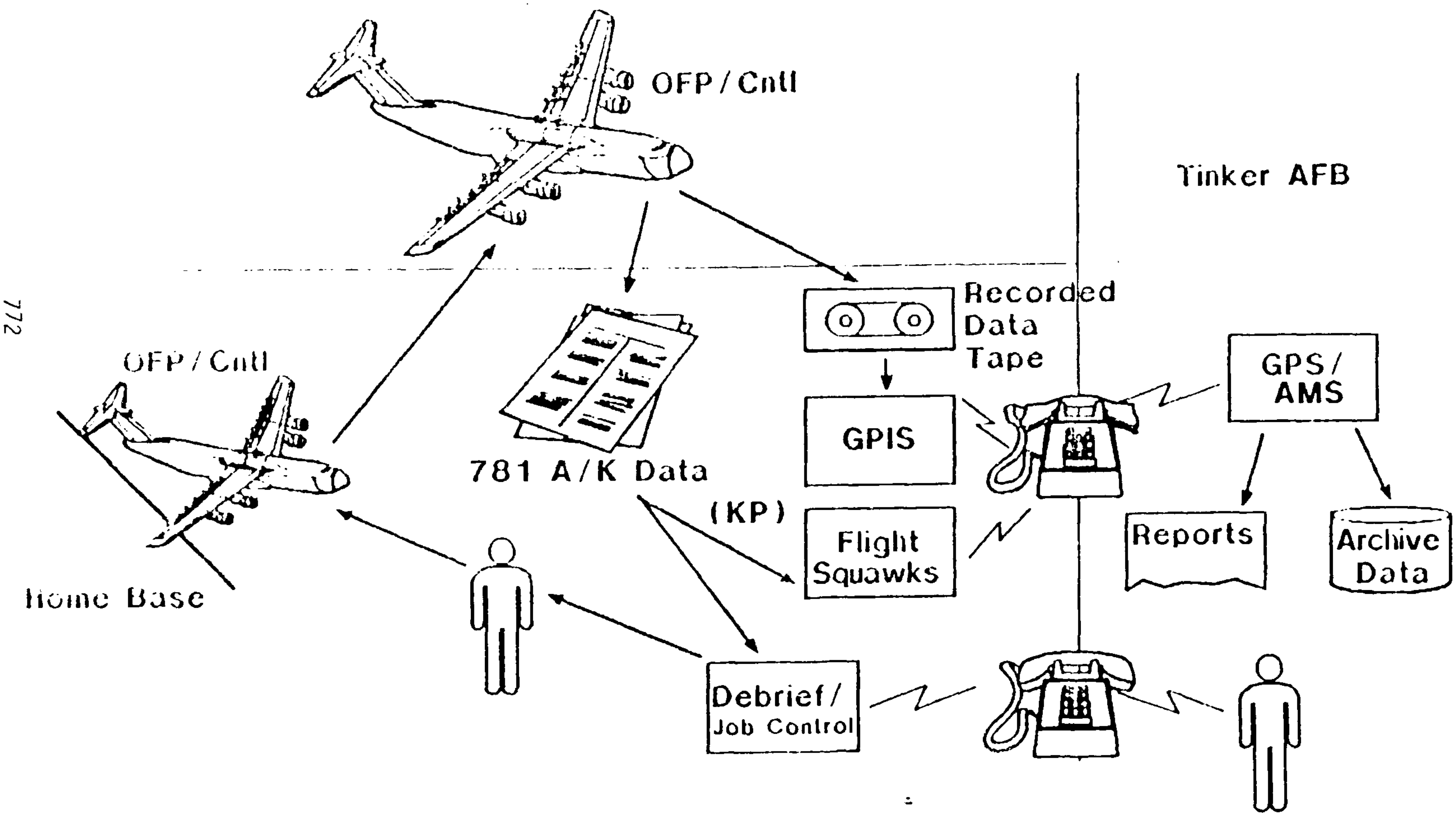


Maintenance Information Flow

Recording of the AFTO 781 A/K data by the Maintenance Data Recorder (MDR) would eliminate the redundant maintenance discrepancy items transcription and keypunching (KP) into the computer system.

C-5 Forms Data/MADAR II

Maintenance Information Flow



Data Acquisition

The design philosophy for forms data acquisition is to maximize automatic data gathering and, therefore, minimize flight engineer inputs to the system.

Current data available to the MADAR system via MADAR test points and CADC data will be utilized in addition to an added total fuel weight test point for the 45 non-LESS C-5B aircraft. (Already accessed on 5 C-5B LESS aircraft and all 77 C-5A aircraft.)

The system will automatically perform necessary calculations for provided data as required to complete the forms data.

The system will also control the interaction of operator input/verification through the existing MADAR display unit including menu selection, forms display, editing, printing and MDR recordings.

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C-5 Forms Data/MADAR II

Data Acquisition

- Maximize Automatic Acquisition; Minimize Flight Engineer Input
- Utilize Current MADAR Test Points
- Add Fuel Totalizer Test Point (On 45 C-5B Aircraft Only)
- Automatic Calculations
- Flight Segment Monitoring
- Display Unit Interactive Operation
- Menu Selection, Edit, Display, Print and Record by Operator Command

Operator Input Data/Verification

Even with maximum emphasis on automation, there are certain data items that are not possible or practical to obtain except by direct flight engineer input.

Examples are initialization/finalization data (cargo weight, takeoff and landing base, etc.) for MAC 89 or MAC 95, takeoff and landing conditions (CG, wind direction, runway length, etc.) for MAC 100 and text/descriptive data for the MAC 89 remarks section and the AFTO 781 A/K.

Maximum utility of the optional keyboard would be realized in input of AFTO 781 A/K discrepancy data.

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C-5 Forms Data/MADAR II

Operator Input Data/Verification

- **MAC Form 89 - Usage**
  - **Initialization/Finalization Data**
  - **Special Segment Data**
  - **Incident/Remark File Text**
- **MAC Form 95 - Fuel Management/Performance**
  - **Initialization Data**
  - **Enroute Updates as Required**
- **MAC Form 100 - Takeoff, Emergency Return and Destination Data**
  - Takeoff Conditions
  - Destination Conditions
- AFTO Form 781 A/K - Discrepancy Data
  - Discrepancies Identified by MADAR
  - Other System Discrepancies
  - Maximum Utility of Optional Keyboard

Menu Frames

Each of the forms will be displayed on the MADAR display unit in standard MADAR template form. Access to the specific forms and individual template types and selection of operations will be controlled through menu item selection.

An illustration of the forms index menu which allows selection of specific form types and example forms menus for the MAC Form 89 are presented.

C-5 Forms Data/MADAR II

Main Forms Menu

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[illegible]

[illegible]

ITEM	CHOICE
MAC FORM [illegible] AIRCRAFT [illegible] RECORD	1 (PF [illegible])
MAC FORM 55 FUEL MANAGEMENT LOG	2 (PF 955)
MAC FORM [illegible] TOLD CARD WORKSHEET	3 (PF 965)
AFTO FORM 781 A-K MAINTENANCE DISCREPANCY AND WORK DOCUMENT	4 (PF 975)
CLEAR DATA FROM ALL FORMS	5

END OF INDEX

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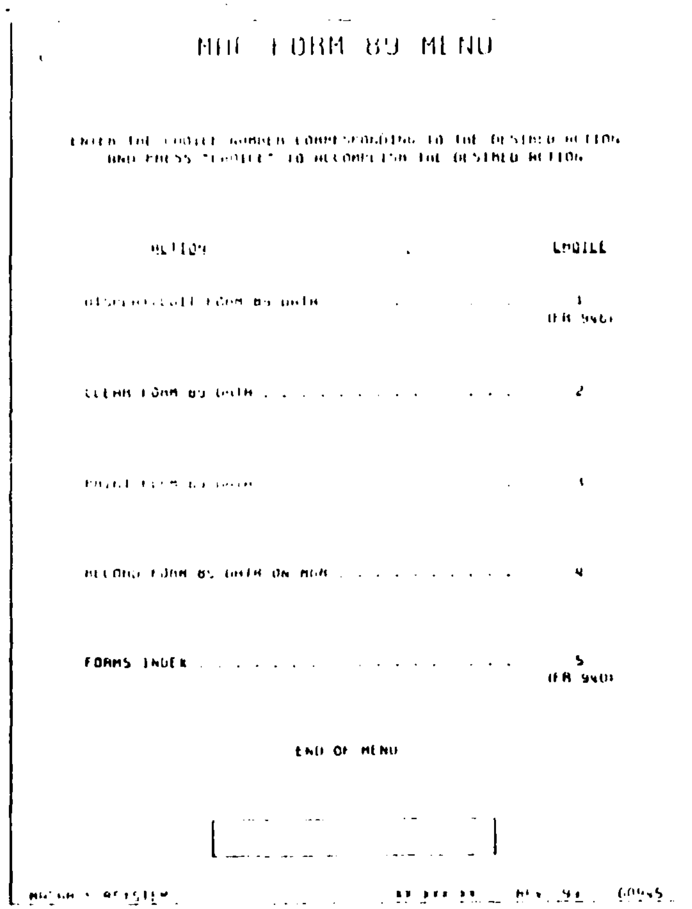
C-5 Forms Data/MADAR II

MAC Form 89 Menu



C-5 Forms Data/MADAR II

MAC Form 89 Display/Edit Menu

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MAC FORM 89 DISPLAY/EDIT MENU

ENTER THE CHOICE NUMBER CORRESPONDING TO THE DESIRED SECTION
AND PRESS "ENTER" TO ACCESS THE DESIRED SECTION

SECTION	CHOICE
AIRCRAFT IDENTIFICATION DATA	1
AIRCRAFT LOADING DATA	2
INITIAL TAKEOFF AND FINAL LANDING DATA	3
FLIGHT PROFILE (CRUISE) DATA	4
MAXIMUM INFLIGHT VERTICAL ACCELERATION .	5
REMARKS	6
INFLIGHT OPERATIONS (AERIAL REFUELING, AIRDROP/CARGO JETTISON, TERRAIN FOLLOWING) DATA . .	7
TRAFFIC PATTERN DATA	8
TOUCH AND GO LANDING DATA (ADDITIONAL FULL STOP LANDINGS AND TAKEOFFS) . .	9
FORMS INDEX	10 [illegible]

END OF MENU

Forms Data Operations

All of the subject form data are displayable through operator interaction. Data fields are entered automatically or manually, verified as appropriate, and displayed in current status form.

The printing of any of the subject forms and the recording of MAC Form 89 or AFTO Form 781 A/K data on the MDR, are controlled by operator command.

Also, forms data are retained in non-volatile mass memory to prevent loss due to power shut-down and are only cleared from the memory by operator command.

C-5 Forms Data/MADAR II

Forms Data Operations

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- All Subject Forms Data Displayed
- Interactive Forms Completion, Verification
- Any Subject Form Printed on Command
- MAC Form 89 and AFTO 781 A/K Recorded on Command
- Mass Memory Storage Cleared Only by Command

Forms Data Back-up Provisions

Back-up provisions will be required for occurrences of recorded data loss or equipment malfunction.

Print-outs of MAC Form 89 and AFTO 781 A/K data will be obtained and retained at the bases as back-up for the MDR data records in the case of data loss due to recorder problems, DFE transmission errors, or ground computer problems.

Manual forms will be available onboard for use in the event of printer or other MADAR component problems.

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C-5 Forms Data/MADAR II

Forms Data Back-Up Provisions

- Print-Outs for MAC Form 89 and AFTO 781 A/K
- Manual Forms Available Onboard for All Forms

Benefits of Forms Automation

The benefits of the subject forms data acquisition automation would be increased data accuracy and detail through a streamlined data flow to enhance analysis time and data accessibility while providing significant workload relief to the flight engineer during flight operations.

Additional benefits will be realized through enhanced usage and maintenance data visibility through the existing Information Management System (IMS).

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C-5 Forms Data / MADAR II

Benefits of Forms Automation

- Significant Relief to Flight Engineer Workload
- Increased Data Accuracy and Detail
- Streamlined Data Flow for Analysis
- Enhanced Management Visibility of Usage and Maintenance Data

Average Manual Entry Comparison

As shown, the reduction in average required manual entries by the flight engineer for the major forms data would be significant.

Under the proposed implementation design, a 75% reduction in manual entries could be realized in an average mission with simultaneously increased accuracy, data detail and more direct data access.

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C-5 Forms Data/MADAR II

Average Manual Entry Comparison

Form Type	Current	After Automation
• MAC Form 89	42	12
• MAC Form 95/ AF Form 796	216	29
• MAC Form 100	135	45
• AFTO Form 781 A/K	25	15

- 75% Reduction in Flight Engineer Entries

Proposed Implementation Plan/Schedule

Air Force approval of the proposed implementation plan, configuration selection and ECP request are pending.

The proposed implementation schedule reflects:

- ECP preparation/approval – 9 months after plan approval and configuration selection
- System development, testing and prototype evaluation – approx. 24 months ARO
- **Kit delivery – span of 24 to 36 months ARO**
- **Air Force kit installation completion – 38 months ARO**

C-5 Forms Data / MADAR II

Proposed Implementation Plan / Schedule

- Air Force Approval / Configuration Selection - Pending
- ECP Preparation / Approval - 9 Months After Approval / Selection
- System Development, Testing, Prototype Evaluation - Approximately 24 Month ARO
- Kit Delivery - Span of 24 Months to 36 Months ARO
- Air Force Kit Installation Complete - Approximately 38 Months ARO

Summary

In conclusion,

The acquisition of C-5 forms data is an essential but time consuming task required of the flight engineer in addition to his primary duties.

The majority of the forms data and the capability required to acquire the data already exists within MADAR II.

With moderate revisions, MADAR II can perform the majority of the Forms related work while enhancing data accuracy, detail, and flow and providing greater data visibility for analysis and force management.

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C-5 Forms Data / MADAR II

Summary

- C-5 Forms Data Are an Essential Element of Flight Operations
- Current Data Preparation Is a Significant Workload in Addition to Primary Flight Engineer Duties
- Majority of Forms Data and Capability Already Exists in MADAR II
- With Moderate Revisions, MADAR II Can Perform Majority of Forms Related Work
- Data Accuracy, Detail, Flow, and Visibility Would Be Greatly Enhanced

AUTOMATED ANALYSIS OF MXU-553 FLIGHT DATA

Kurt H. Schrader

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San Antonio, Texas

As part of the Aircraft Structural Integrity Program developed by the U.S. Air Force, different types of aircraft have been involved in a flight data recording program for many years. San Antonio Air Logistics Center/MMSA contracted with Alamo Technology, Inc., to develop software for analyzing the data recorded by the MXU-553 equipped aircraft, for generating spectra and profile information, and to display the resulting data. This software was designed to process not only the MXU-553 data but also the data from the micro-processor recorders being developed at this time. Once all software had been written, a final requirement of the contract with the Air Force was to process many hours of MXU-553 data and report the results.

The programs developed for SA-ALC/MMSA are summarized in the interaction overview. The first program consists of a compression effort where the raw MXU-553 data is converted to engineering units and compressed to retain only significant flight data. This process is conducted at OC-ALC and the compressed tapes are sent to SA-ALC for further analysis. (In the future, this effort will be accomplished by the on-board micro-processors.) The amount of compression is quite significant; the original MXU-553 data tape from an aircraft may contain from 10 to 15 flight hours where the compressed tape can contain approximately 1000 flight hours. This compressed flight data is processed by SA-ALC using the Edit/Pre-analysis program which allows for tabular and graphic display of data and editing of erroneous information. The final program permits the tabulation of spectra and profile information and is named S.O.U.P. (Spectra and Operational Usage Profile). This program will be the subject of this presentation.

All programs have been written to accept data from four MXU-553 equipped aircraft: T-38; F-5E/F; T-37; and OA-37. However, the programs are modular in nature so that aircraft can be added in the future.

The S.O.U.P. program was designed to run on SA-ALC's VAX 11/780 computer running the VMS operating system while using the Tektronix 4107 color terminal. The program has been used extensively on a microVAX computer system with a variety of Digital Equipment Corporation terminals so the program has demonstrated some portability. The program incorporates both interactive and batch processing modes and can be run as frequently as the user requires - daily, quarterly, semi-annually, etc. Typically, the ASIP manager would run the program quarterly to monitor the overall usage of a given fleet as compared to previous quarters or years.

The S.O.U.P. consists of three separate phases or sub-programs named Options, Count, and Disply. Options and Disply can be thought of as pre- and post-processors of the database generated during the middle phase. The Count program is the big number-cruncher which creates a very large database from the user input parameters. Each individual phase of the S.O.U.P. program will be discussed below.

Options incorporates an interactive menu system to allow the user to select from a wide variety of processing parameters. These parameters include such things as aircraft type, date (for use in restricting the data to be included in the analysis), base, command, and several statistical categories. These parameters will be presented in more detail later. The user receives a cursory review of all processing parameters prior to submittal of the batch job.

The first level in the flow of the Options program is to select a parameter file. This can be thought of as a template or style sheet which has pre-selected parameters. The user then proceeds to the first level where aircraft type, date range, command, base, and tail number are selected. These parameters restrict the amount of flight data that will be processed. For instance, a specific command selection such as Air Training Command would allow only data meeting that criteria to be considered for analysis.

The next level would be to select the types of statistics to gathered. These fall into four broad areas known as ASIP, ENSIP, O.U.P. (operational usage profiles), and Spectra data. There are additional selections in each of these categories which will be highlighted in the following charts.

The final level would be to review the parameter selections and start the data processing. The parameters selected during levels one and two have an impact on the processing time of the batch job. Level one selections have a small impact because the program will have to investigate each and every flight data file to determine if it meets the criteria for analysis. The selections on level two (statistical categories) have a much greater impact on processing time since the program will only sort data for those statistical categories that are requested. With this in mind, it is possible to make the appropriate parameter selections and ensure quick turnaround when the user requires specific results.

In each of the statistical categories, there a number individual selections that are also available and these are shown on the next four charts. In the ASIP area, a variety of cross tabulations can be selected. These include normal load factor (Nz) in airspeed and altitude blocks, Nz exceedences versus airspeed, altitude, gross weight, and mission segment, flight time for similar categories, and finally information about the aircraft that were used in the analysis.

There are only two selections in the ENSIP category, power lever angle versus time and engine speed versus time. The concept for this

Air Force program was conceived 4-5 years ago and the requirements for critical ENSIP parameter study was limited to these two areas. In addition, none of the four aircraft involved in this program are recording any ENSIP information on their MXU-553 recorder systems. As a result, this portion of the S.O.U.P. has never been validated.

The third statistical category contains operational usage profile (O.U.P.) tabulations. These include average information about mission profiles and phases (average time spent in a major mission phase, average gross weight and velocity in a phase, etc.) There is also information about discrete Nz occurrences.

The final category provides information about Nz exceedences and vertical tail bending moment (VTMX) or lateral acceleration (Ny) exceedences. An analytical equation for VTMX for both the T-38 and F-5E/F is used to calculate the bending moment and the S.O.U.P. program tabulates exceedences for this parameter. For the T-37 and OA-37, no such equation exists and the exceedences are tabulated for Ny.

Following all selections using the Options program, the batch processing can begin. The Count program develops a large database based on the selected parameters and informs the user that the job has terminated.

The last program, Disply, is used in an interactive environment to display the items in this large database. This program uses a series of menus in a hierarchical fashion to display the data in tabular, graphic, and histogram form. While viewing the many cross tabulations, the user can create hardcopies of the data either by screen copy commands associated with the Tektronix terminal or by routing output to the system line printer using Disply menu selections.

As an example of some of the output from the Disply program, the following charts have been chosen. From the ASIP category, the TIME PERCENTAGES BY SEGMENT was selected. The result is a screen which provides some background on the data included in this analysis. All commands and bases have been included as well as all dates (0-99999). The mission "high altitude combat" has been selected and represents 58.5% (545.83 hours) of the total hours available for analysis (932.57 hours). The percentage of time spent in each major mission segment is shown on the bottom of the chart. This data can also be displayed in histogram form as seen on the next chart.

The next selection is from the O.U.P. category and is DISCRETE NZ OCCURRENCES BY FLIGHT CONDITIONS. The next chart shows only a small portion of the data displayed for discrete occurrences for 1000 mission hours. Only the primary mission segment is shown and the Nz's are limited to 2.00 to 5.00 g's. The output would normally consist of all mission segments and shown occurrences out to 9.00 g's as well as negative g's. The Nz's are tabulated in bands for each representative flight condition. For example, for 1000 hours of high altitude combat mission in the primary phase, there were 1740 Nz occurrences between

2.00 and 2.50 g's for representative flight condition number 1. This flight condition has been previously defined as a Mach, airspeed, altitude, and flap setting condition for the F-5E. This particular output by the Disply program can be quite useful in developing cycle-by-cycle stress information for crack growth analysis in further DTA studies or comparisons.

From the Spectra category, NZ EXCEEDENCES was selected and the next two charts show the type of data that this category can provide. The Disply program cannot plot multiple lines per graph but it is a simple matter to take the tabular information from Disply and plot the data on a separate piece of log-linear graph paper. The first spectra plot shows the Nz exceedences (or cumulative occurrences) per 1000 mission hours for four different missions. The second spectra plot shows the Nz exceedences per 1000 phase hours for each phase of the high altitude mission.

In closing, I would like to highlight the benefits of the S.O.U.P. program. It is now possible to process more data in a shorter period of time. Spectra plots can be developed in about one hour and cycle-by-cycle stress information can be generated in 2-3 days. This process used to take 2-3 months. As an example, if the ASIP manager needed comparison spectra plots for the F-5E aircraft for all data during the year 1986 so that his commander could have the information for a meeting in one hour, he now has that capability. The ASIP manager would use the Options program to make very restrictive selections from all processing parameters so that the batch program, Count, could run in the shortest time possible. The spectra information could then be displayed and the data values transferred to a convenient form similar to the Nz exceedence plots shown in the previous charts.

It takes less manpower to use the S.O.U.P. programs. The ASIP manager can sit down and use the Options program near the end of the work day and start a batch process job that could run overnight. At the beginning of the next work day, the data would be available for him to develop charts and tables for a required quarterly update of aircraft usage.

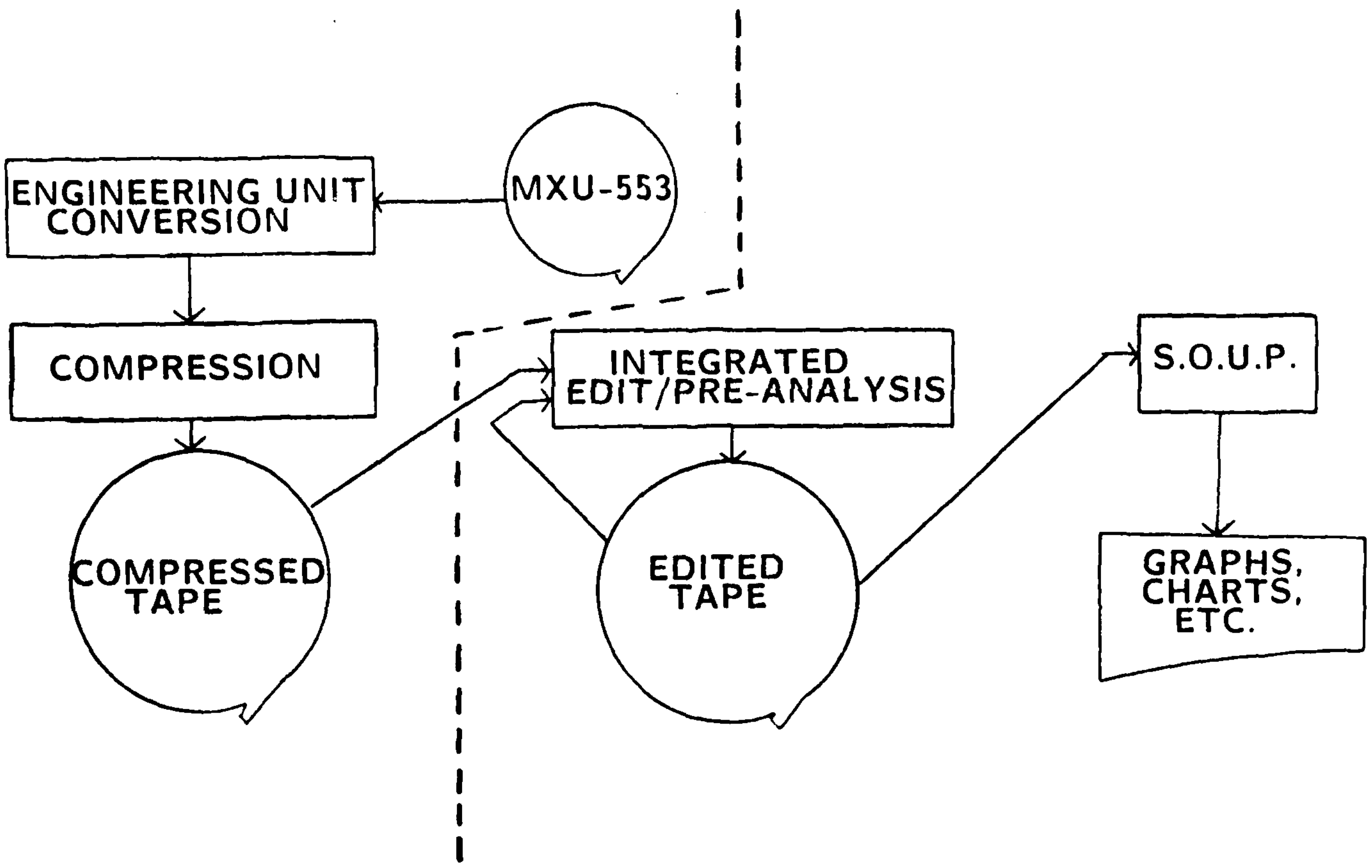
Using the program and associate template files, the analysis and comparison of aircraft usage from period to period would be identical. Successive ASIP managers for a given aircraft system would develop consistent results by using the tools provided by the S.O.U.P. program.

Finally, the S.O.U.P. is already being used on the current contract for SA-ALC. Some of the results for nearly 1000 hours of F-5E data have been shown during this presentation. Approximately 1300 hours of T-37 flight data has been analyzed and a report issued to the U.S. Air Force. By contract end, about 1100 hours of OA-37 data and over 3000 hours of T-38 MXU-553 data will have been processed using the S.O.U.P. program.

INTRODUCTION

- WORK PERFORMED FOR ALAMO TECHNOLOGY, INC.
- SA-ALC CONTRACT
- DEVELOP SOFTWARE FOR GENERATING SPECTRA AND PROFILE INFORMATION
- MXU-553 AND MICROPROCESSOR RECORDER SYSTEMS
- PROCESS MXU-553 DATA

PROGRAM INTERACTION OVERVIEW



DESIGNED FOR MXU-553 EQUIPPED A/C

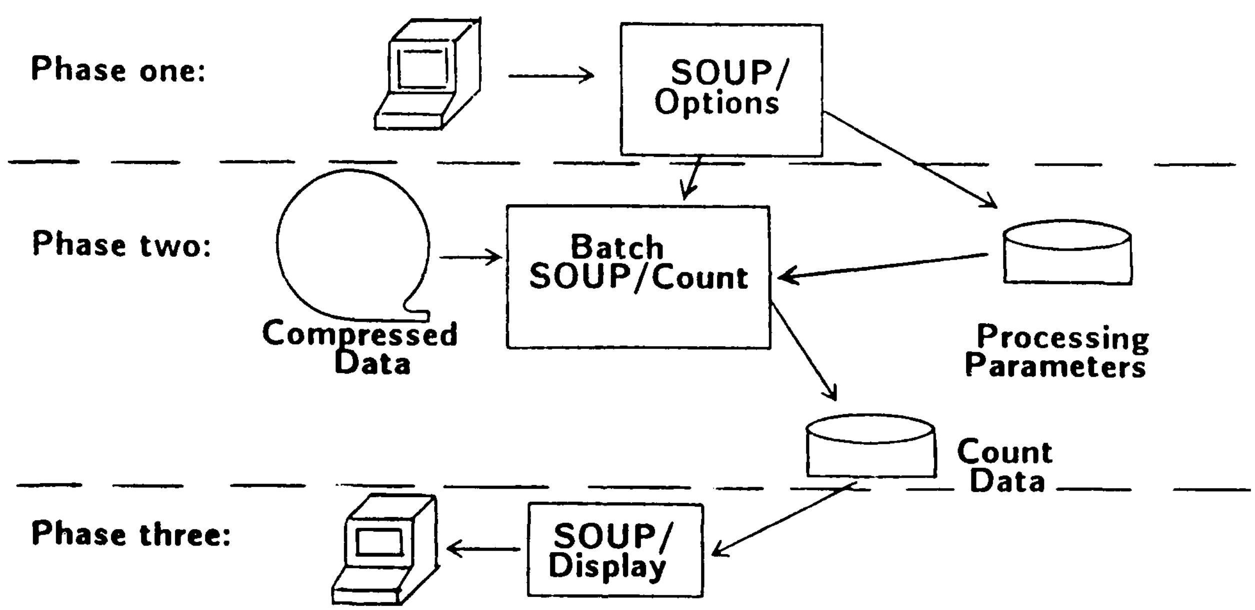
- T-38
- F-5E/F
- T-37
- OA-37
- FUTURE APPLICATIONS

OPERATIONAL CONSIDERATIONS

- VAX 11/780 WITH VMS OPERATING SYSTEM
- TEKTRONIX 4107 COLOR TERMINAL
- INTERACTIVE / BATCH PROCESSING
- DAILY / QUARTERLY / SEMI-ANNUALLY

S.O.U.P.

PROGRAM STRUCTURE



OPTIONS

- INTERACTIVE MENU SYSTEM TO DETERMINE PROCESSING PARAMETERS
- SELECT FROM A WIDE RANGE OF PARAMETERS
- CURSORY REVIEW OF PARAMETERS PRIOR TO BATCH JOB SUBMITTAL

OPTIONS FLOW

SELECT PARAMETER FILE

↓

SELECT
- A/C TYPE
- DATE WINDOW
- COMMAND
- BASE
- TAIL NUMBER

↓

SELECT STATISTICAL CATEGORIES
- ASIP
- ENSIP
- O.U.P.
- SPECTRA

↓

START PROCESSING

ASIP STATISTICAL CATEGORIES:

- NZ LOAD FACTOR DISTRIBUTION
- TIME PERCENTAGES BY SEGMENT
- NZ EXCEEDENCES BY AIRSPEED
- NZ EXCEEDENCES BY ALTITUDE
- NZ EXCEEDENCES BY GROSS WEIGHT
- NZ EXCEEDENCES BY SEGMENT
- FLIGHT TIME - SEGMENT VS. AIRSPEED
- FLIGHT TIME - SEGMENT VS. ALTITUDE
- FLIGHT TIME - SEGMENT VS. GROSS WEIGHT
- FLIGHT TIME - AIRSPEED VS. ALTITUDE
- FLIGHT TIME - MISSION VS. SEGMENT DURATION
- AIRCRAFT CHARACTERISTICS

ENSIP STATISTICAL CATEGORIES:

- POWER LEVEL ANGLE VS. TIME
- ENGINE SPEED VS. TIME

O.U.P. STATISTICAL CATEGORIES:

- MISSION PROFILE AVERAGES
- MISSION PHASE FLIGHT PARAMETERS
- DISCRETE NZ OCCURRENCES
- DISCRETE NZ OCCURRENCES BY FLIGHT CONDITIONS

SPECTRA STATISTICAL CATEGORIES:

- NZ EXCEEDENCES
- VTMX / NY EXCEEDENCES

COUNT

- BATCH PROCESSING
- DEVELOPS DATABASE BASED ON PREVIOUSLY SELECTED-PARAMETERS
- INFORMS USER OF JOB TERMINATION

DISPLY

- INTERACTIVE
- HIERARCHICAL / MENU ORIENTED
- DISPLAYS DATA IN TABULAR, GRAPHIC, HISTOGRAM FORM
- HARDCOPIES OF DATA
  - SCREEN COPIES
  - LINE PRINTER OUTPUT

ASIP STATISTICAL CATEGORIES:

- NZ LOAD FACTOR DISTRIBUTION
- TIME PER[illegible]
- NZ EXCEEDENCES BY AIRSPEED
- NZ EXCEEDENCES BY ALTITUDE
- NZ EXCEEDENCES BY GROSS WEIGHT
- NZ EXCEEDENCES BY SEGMENT
- FLIGHT TIME - SEGMENT VS. AIRSPEED
- FLIGHT TIME - SEGMENT VS. ALTITUDE
- FLIGHT TIME - SEGMENT VS. GROSS WEIGHT
- FLIGHT TIME - AIRSPEED VS. ALTITUDE
- FLIGHT TIME - MISSION VS. SEGMENT DURATION
- AIRCRAFT CHARACTERISTICS

F-5E PERCENTAGE OF MISSION TIME

COMMAND: ALL BASE: ALL DATE: 0 - 99999

USAGE: HIGH ALTITUDE COMBAT

TIME -- TOTAL: 932.57 MISSION: 545.83 % USAGE: 58.5

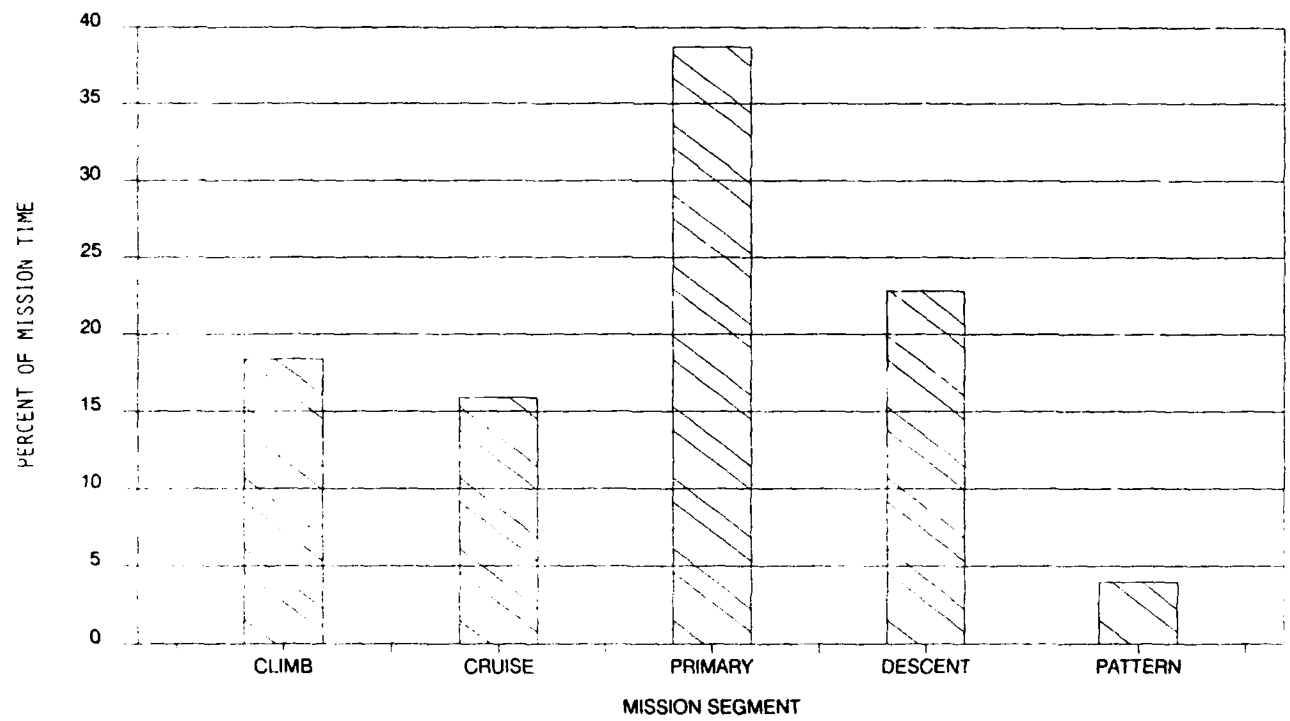
MISSION SEGMENT	% TIME
CLIMB	18.4
CRUISE	15.9
PRIMARY	38.7
DESCENT	22.8
PATTERN	4.0

F-5E PERCENTAGE OF MISSION TIME

COMMAND: ALL BASE: ALL DATE: 0 - 99999

USAGE: HIGH ALTITUDE COMBAT

TIME -- TOTAL: 932.57 MISSION: 545.83 % USAGE: 58.5



O.U.P. STATISTICAL CATEGORIES:

- MISSION PROFILE AVERAGES
- MISSION PHASE FLIGHT PARAMETERS
- DISCRETE NZ OCCURRENCES
- DISCRETE NZ OC[illegible]NS

F-5E DISCRETE OCCURRENCE DISTRIBUTION PER 1000 MISSION HOURS

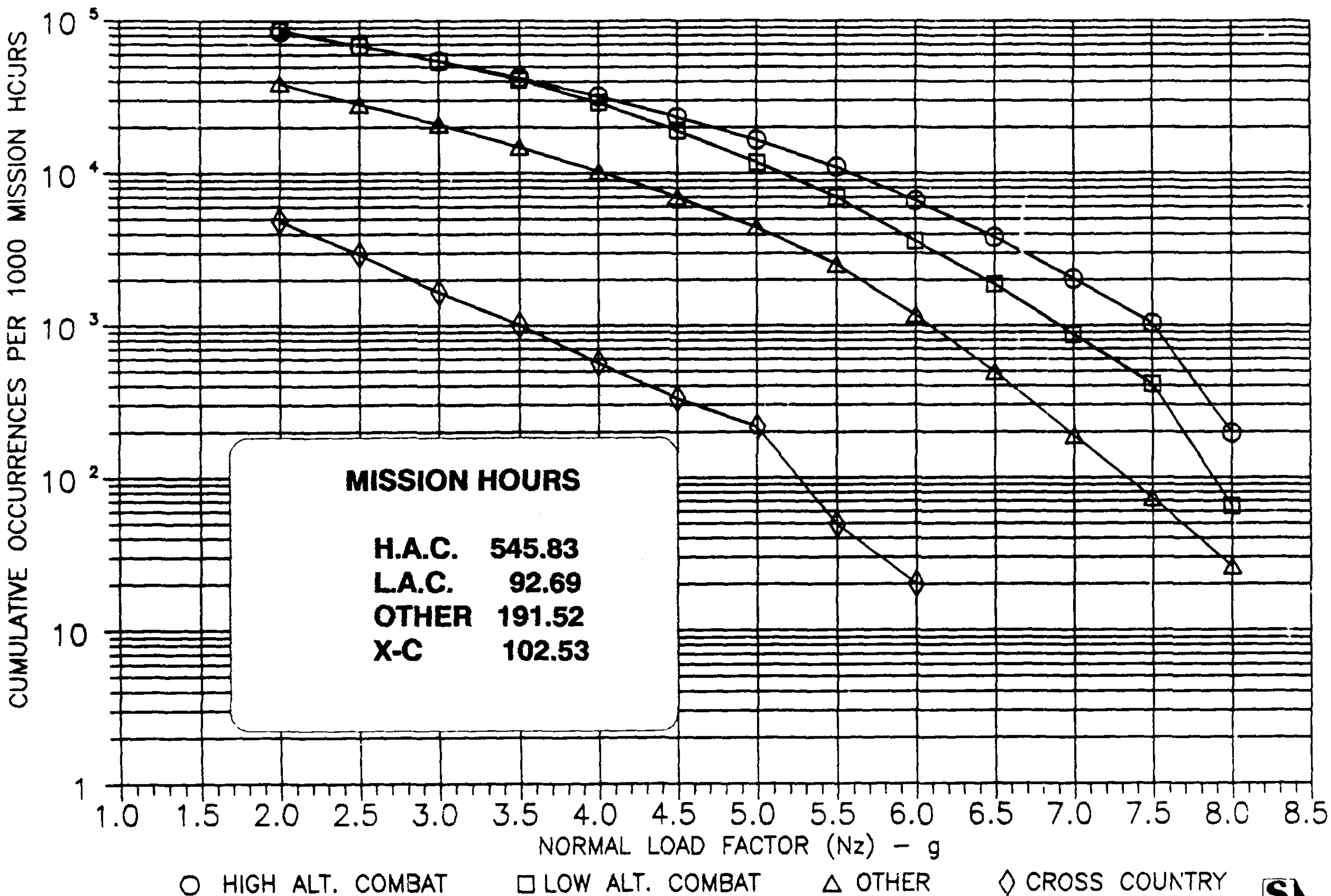
COMMAND: ALL BASE: ALL DATE: 0 - 99999
USAGE: HIGH ALTITUDE COMBAT
TIME -- TOTAL: 932.57 MISSION: 545.83 % USAGE: 58.5

MISSION SEGMENT	COND. NO.	W/W REF	GROSS WEIGHT	POSITIVE NZS - NZ AT LOWER BOUNDARY						
				2.00	2.50	3.00	3.50	4.00	4.50	5.00
PRIMARY	1	1.10	14092	1740	1570	1436	1277	572	436	445
	2	1.10	14062	0	0	0	0	0	0	513
	3	1.09	14011	799	1008	575	106	53	40	26
	8	0.95	14121	2867	2404	1830	1473	2704	1955	1286
	9	0.95	14075	431	451	632	753	2270	2089	1697
	10	0.95	14156	205	181	145	139	181	101	75
	12	1.10	14028	4186	3155	2395	1618	106	0	0
	13	1.18	14092	705	1110	1913	1951	134	0	0
	14	1.18	14079	0	0	0	0	429	390	29
	15	1.19	14159	0	0	0	0	425	491	48
	16	1.19	14127	1601	1484	1317	1218	867	720	856
	17	1.10	14119	137	176	266	266	126	97	137
	18	1.11	14241	991	826	751	581	330	293	229
	19	1.10	14069	7	7	15	15	9	11	7

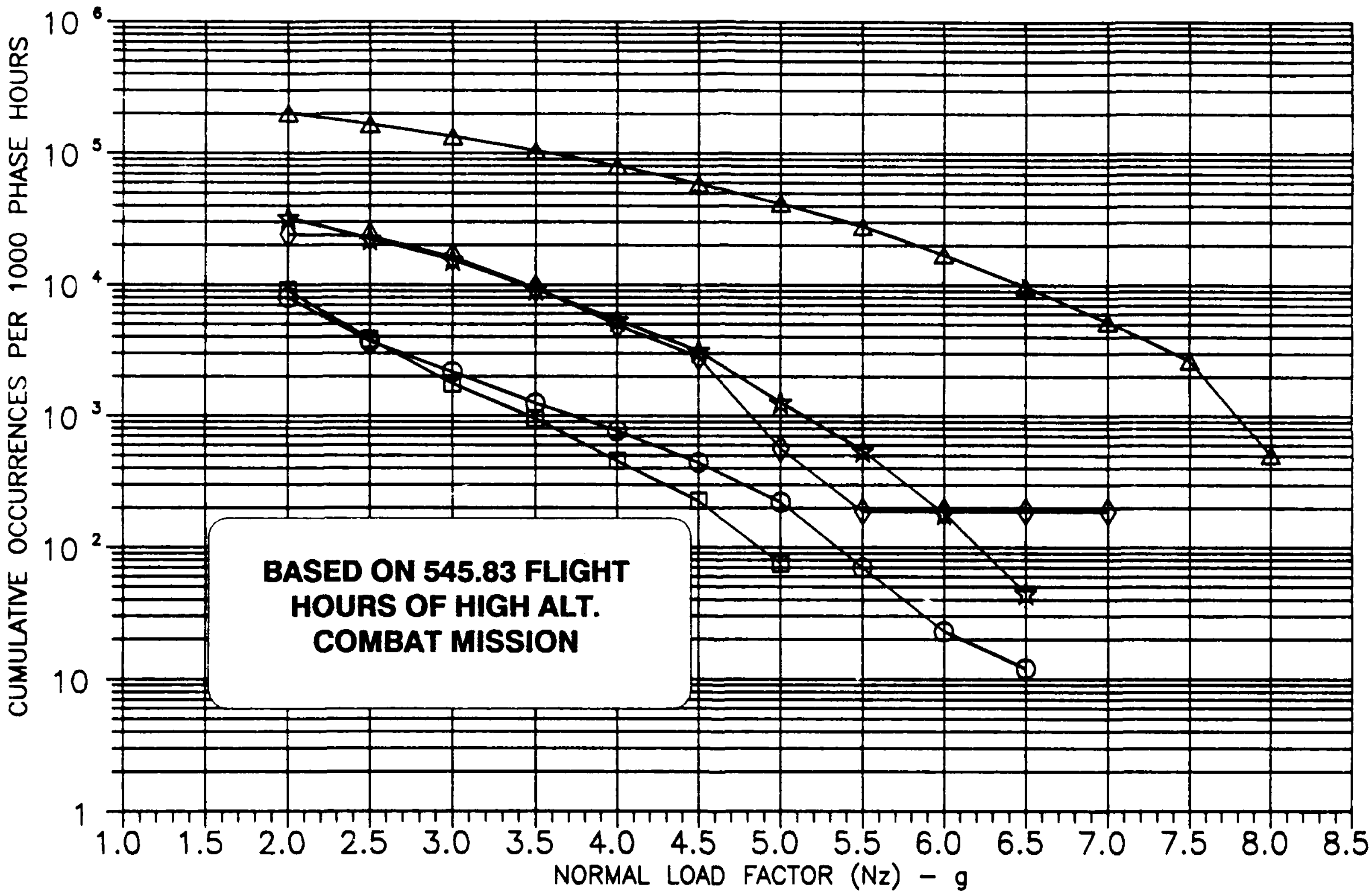
SPECTRA STATISTICAL CATEGORIES:

- NZ EXC
- VTMX / NY EXCEEDENCES

F−5E MISSION SPECTRA ANALYSIS PER 1000 MISSION HOURS
COMMAND: ALL BASE: ALL DATE: 0−99999



F–5E PHASE SPECTRA ANALYSIS PER 1000 PHASE HOURS
COMMAND: ALL BASE: ALL DATE: 0–99999



BENEFITS

- MORE DATA IN SHORTER TIME
- REDUCED MANPOWER
- CONSISTENT
- ALREADY BEING USED
  - 1000 HOURS - F-5E
  - 1300 HOURS - T-37
  - 1100 HOURS - OA-37
  - 3000 HOURS - T-38

PEAK IDENTIFICATION TECHNIQUES
(ABSTRACT)

Prepared for: 1987 USAF Aircraft/Engine (ASIP/ENSIP) Structural Integrity Program Conference

Peak identification is performed, primarily for flight measured data, to reduce the amount of information which must be considered for structural life analyses. The peak identification methodology applied defines those points in time, and therefore limits the measured data, for which further analysis will be performed. Ideally, the peak identification technique used will retain all stress cycles which contribute to structural damage while eliminating those times within an aircraft's life which are unimportant in a structural sense. In this context, that criteria is quite severe. To fulfill those requirements would mean that all aspects of the aircraft structure is well understood, which is seldom the case, and that all future concerns have been anticipated, which is never the case.

Peak identification methods can have a profound effect on the structural life analyses which are performed for the resulting spectra, and realizing these effects has become even more important with the advent of on-board, or black box, processing of fight recorded data. There is a current trend toward automated analyses, including spectral development. While this trend may be necessary due to the amount of work required for the limited staffs available, there is a danger which is inherent in using procedures which are not well understood or for which the limitations are not known.

This Peak Identification Techniques presentation is limited to a discussion of methods by which peak maneuver response times are established. The presentation does not discuss ordering techniques such as rang-pair or rain-flow. Additionally, the presentation primarily addresses fighter, attack, and fighter-trainer aircraft. Although examples and discussions are based on these limitations, the concepts presented have implications across a broad range of applications.

The Peak Identification Techniques presentation concentrates on three established methods of peak identification. These methods are known as "Conventional Count" or "50% Rise-Fall", "Range Pair" or "Fixed Increment", and "Counting Accelerometer" or "Fixed Return". It should be noted that the method known as "Range Pair" does not refer to the

occurrence ordering technique known as "Range Pair Counting" but is a name assigned to a peak identification technique which was originally established to be used in conjunction with "Range Pair Counting". Each of these peak identification techniques is described in terms of criteria and examples.

In order to compare the effect of using these three peak identification methods, crack growth analysis results are given for actual flight data obtained using each method. Also, crack growth analysis results are given for the resulting data as processed by three cycle-by-cycle development techniques. These cycle-by-cycle development techniques are:

1. direct cycle-by-cycle, where the actual saved maxima and minima values are used.
2. max occurrence spectrum with one g return, where each occurrence within a maxima range is assigned a range midpoint value and is coupled with a normal load factor of one g. The result is randomly sequenced.

and

3. max-min matrix where a two dimensional matrix of occurrences for maxima and minima ranges is created and the result is randomly sequenced.

Implications of the choice of peak identification techniques are discussed. This discussion is given not only in an absolute sense, but with respect to the manner in which the peak identified data is to be used. The comparisons given above are used to describe the dangers in using spectral data established in a manner which is not consistent with the analytical methods utilizing that data. The presentation is concluded with recommendations for peak identification techniques to be used for various circumstances and suggestions for further investigations.

PEAK IDENTIFICATION TECHNIQUES

I BACKGROUND

II CONCERN

III INVESTIGATION

IV ANALYSIS RESULTS

V CONCLUSIONS

VI RECOMMENDATIONS

PEAK IDENTIFICATION TECHNIQUES

BACKGROUND

PROFOUND IMPACT ON STRUCTURAL LIFE ANALYSES

BASED ON DATA AVAILABILITY

- COUNTING ACCELEROMETER - BUILT-IN CRITERIA
- LOADS/ENVIRONMENT SPECTRA SURVEY (L/ESS) - VGH, MXU
  - TIME HISTORY FLIGHT DATA
  - PEAK IDENTIFICATION - ANALYST'S CHOICE

LIMITED STUDIES TO ESTABLISH BEST METHODS

PEAK IDENTIFICATION TECHNIQUES

CONCERN

EFFECT OF PEAK IDENTIFICATION TECHNIQUES UNKNOWN

- ARBITRARY CHOICES
- MISLEADING ANALYTICAL RESULTS

ON-BOARD MICRO-PROCESSOR SYSTEMS

- PRE-PROGRAMMED PEAK I. D. METHODS
- POSSIBLY LIMITED DATA AVAILABILITY

PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

PURPOSE - EVALUATE VARIOUS PEAK I.D. METHODS

- APPLICATION VIEWPOINT
  - PROCESSING OF FLIGHT RECORDED DATA
  - COMPARISON OF STRUCTURAL LIFE ANALYSIS RESULTS
- DATA PROCESSING LIMITATIONS
  - COMPUTER TIME REQUIREMENTS
  - DATA STORAGE REQUIREMENTS

PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

LIMITATIONS

- F-5E MXU-553 FLIGHT RECORDED DATA
  - DISSIMILAR AIR COMBAT TRAINING (DACT) ENVIRONMENT
  - 932.57 FLIGHT HOURS OF DATA
  - NORMAL LOAD FACTOR (Nz) ONLY
- ONE STRUCTURAL LOCATION - DORSAL LONGERON
- SINGLE "POINT-IN-THE-SKY" LOAD
  - MACH = 0.80
  - ALTITUDE = 15000 FT.
  - FLAP DEFLECTION = 12/8 (LEADING/TRAILING EDGE)
  - GROSS WEIGHT = 50% FUEL, NO EXTERNAL STORES

PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

LIMITATIONS

- DTA ESTABLISHED RELATIONSHIPS
  - STRESS/LOAD RATIO
  - MATERIAL PROPERTIES
  - CRACK GROWTH MODEL
- NO RETARDATION

PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

DEFINITION OF TERMS

- CYCLE-BY-CYCLE SEQUENCE - SEQUENCE OF NORMAL LOAD FACTOR (Nz) VALUES (MAX_1, MIN_1, MAX_2, MIN_2, ... MAX_i, MIN_i)
- MEASURED FLIGHT SEQUENCE - CYCLE-BY-CYCLE SE-QUENCE TAKEN DIRECTLY FROM FLIGHT MEASURED DATA
- MAXIMUM OCCURRENCE SPECTRUM - NUMBER OF NORMAL LOAD FACTOR PEAK VALUES FOR VARIOUS Nz BANDS
- OCCURRENCE MATRIX - NUMBER OF Nz VALUES FOR VARIOUS Nz MAX/MIN BANDS

PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

PEAK IDENTIFICATION METHODS

- RANGE PAIR
  - A MAXIMUM IS THE LARGEST VALUE BETWEEN TWO MINIMA AND IS AT LEAST 0.50g GREATER THAN EITHER THE PRECEDING OR FOLLOWING MINIMUM
  - A MINIMUM IS THE SMALLEST VALUE BETWEEN TWO MAXIMA

PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

PEAK IDENTIFICATION METHODS

- CONVENTIONAL
  - A MAXIMUM
    - IS THE LARGEST VALUE BETWEEN TWO MINIMA
    - MUST BE PRECEDED AND FOLLOWED BY A MINIMUM WHICH IS NO MORE THAN 50% OF THE MAXIMUM - 1.0g AND AT LEAST 1.0g SMALLER THAN THE MAXIMUM
    - WHICH IS LESS THAN 2.0g MUST BE FOLLOWED BY A MINIMUM WHICH IS LESS THAN OR EQUAL TO 0.0g
  - A MINIMUM
    - A MINIMUM IS THE SMALLEST VALUE BETWEEN TWO MAXIMA
    - MUST BE NO MORE THAN 50% OF THE MAXIMUM - 1.0g
    - MUST BE AT LEAST 1.0g LESS THAN THE PRECEDING AND FOLLOWING MAXIMA

PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

PEAK IDENTIFICATION METHODS

- COUNTING ACCELEROMETER
  - FOR Nz EXCEEDING 2.5g, THE MAX Nz IS COUNTED WHEN Nz RETURNS TO 1.3g
  - FOR Nz LESS THAN 0.30g, THE MIN Nz IS COUNTED WHEN Nz RETURNS TO 0.7g

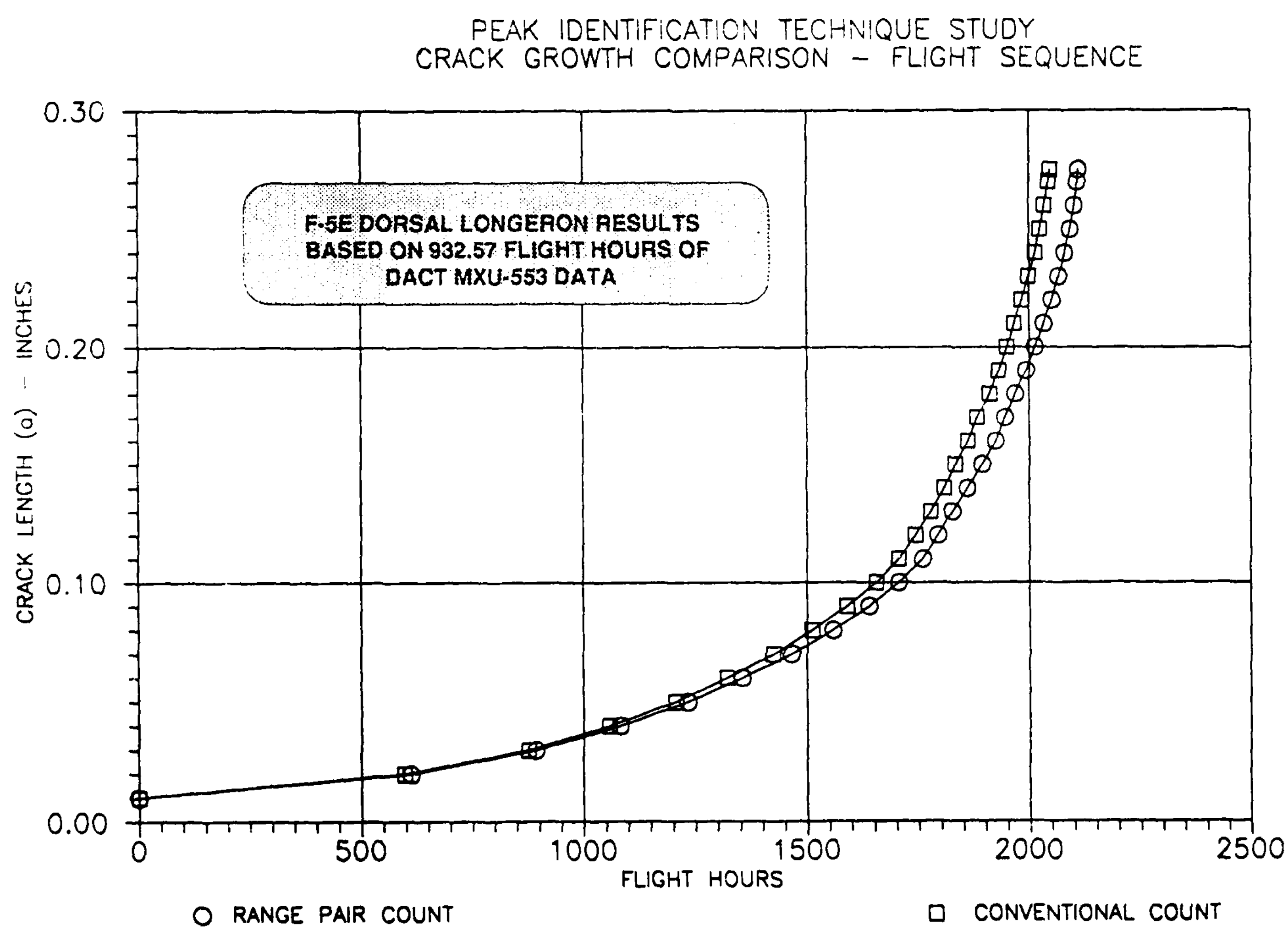
PEAK IDENTIFICATION TECHNIQUES

INVESTIGATION

ANALYSES

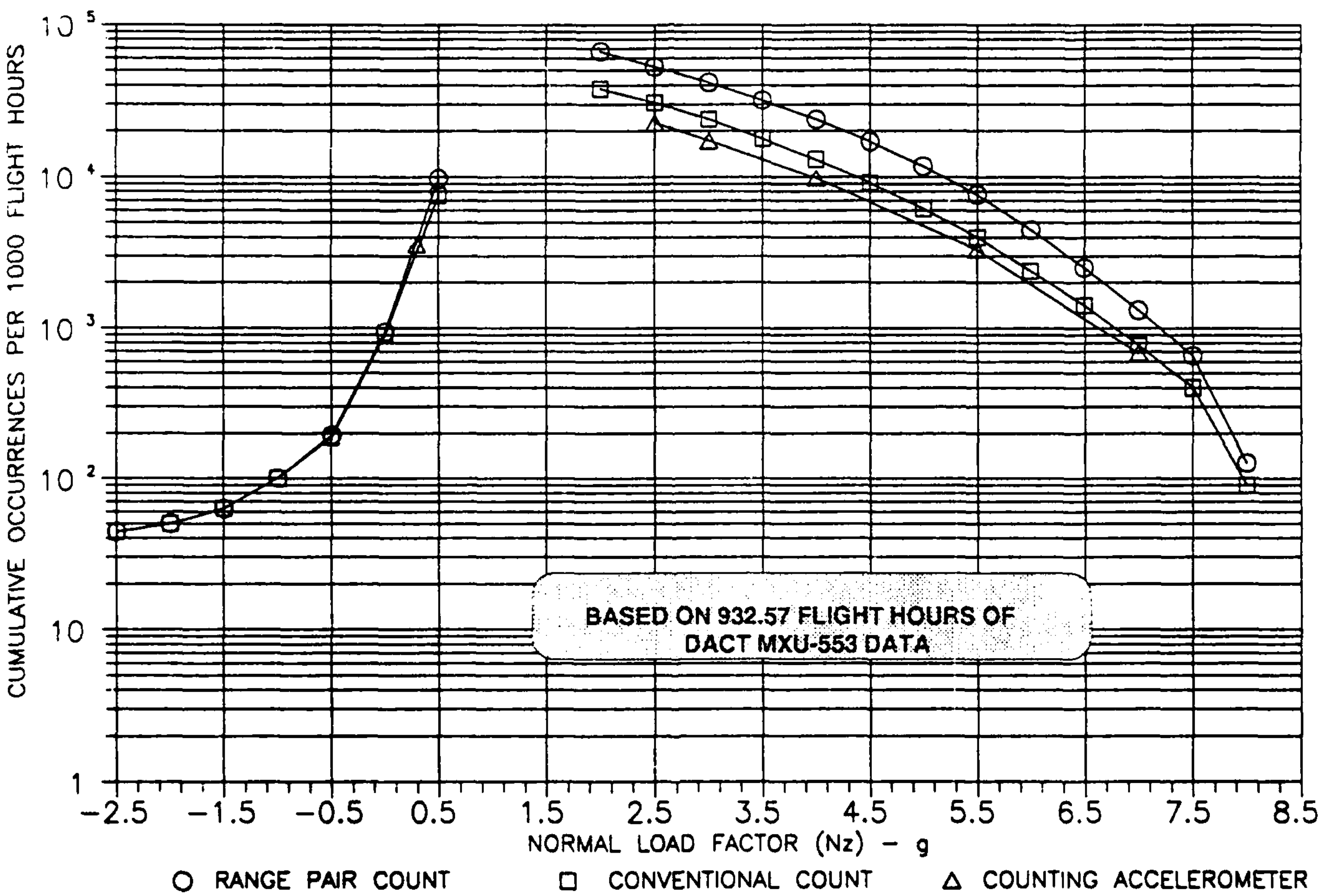
- CYCLE-BY-CYCLE FLIGHT SEQUENCE
  - DERIVED USING RANGE PAIR PEAK I.D.
  - DERIVED USING CONVENTIONAL PEAK I.D.
- RANDOM CYCLE-BY-CYCLE SEQUENCE FROM MAXIMUM OCCURRENCE SPECTRA
  - DERIVED USING RANGE PAIR PEAK I.D.
  - DERIVED USING CONVENTIONAL PEAK I.D.
  - DERIVED USING COUNTING ACCELEROMETER PEAK I.D.
- RANDOM CYCLE-BY-CYCLE SEQUENCE FROM OCCURRENCE MATRICES
  - DERIVED USING RANGE PAIR PEAK I.D.
  - DERIVED USING CONVENTIONAL PEAK I.D.

PEAK IDENTIFICATION TECHNIQUES

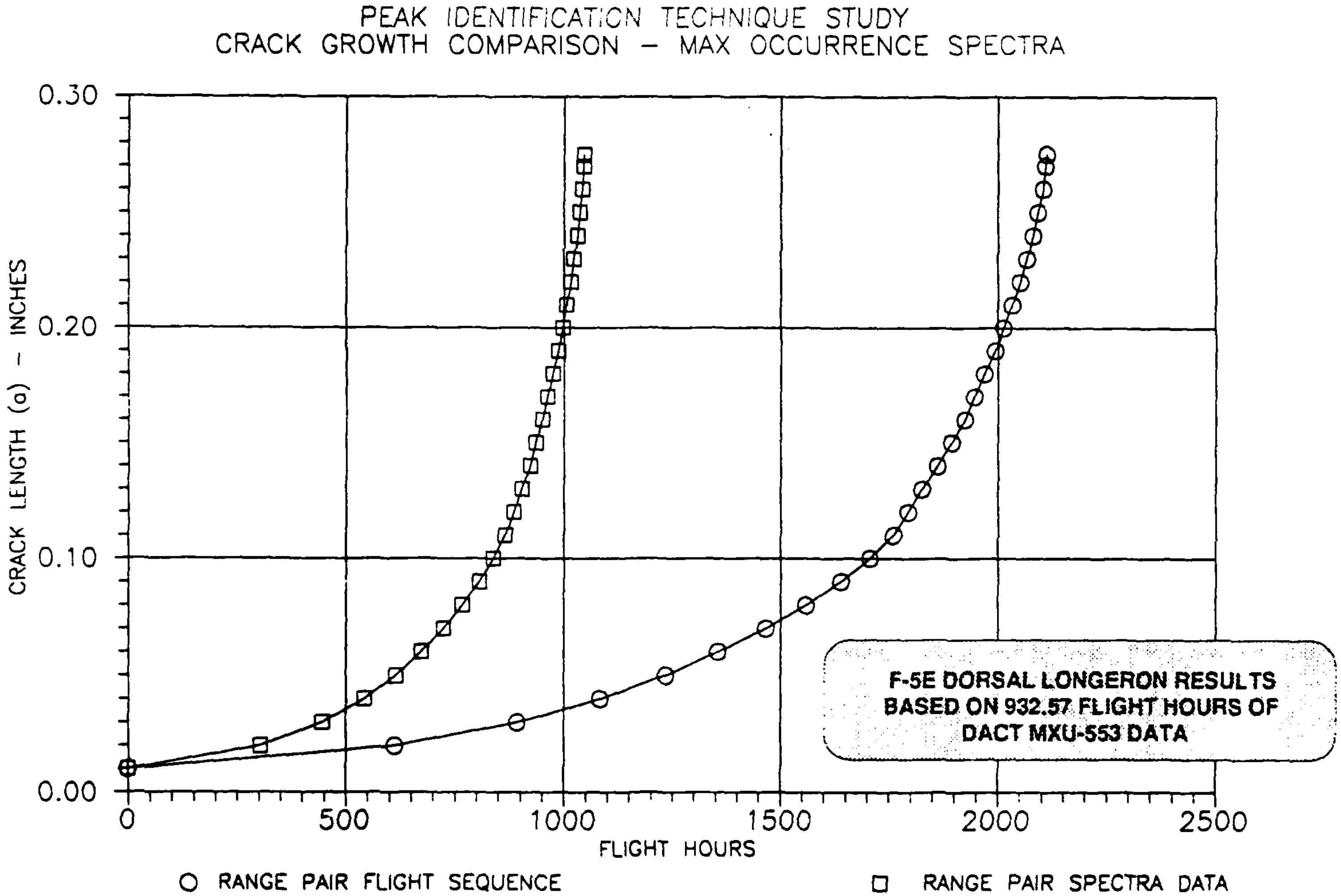


PEAK IDENTIFICATION TECHNIQUES

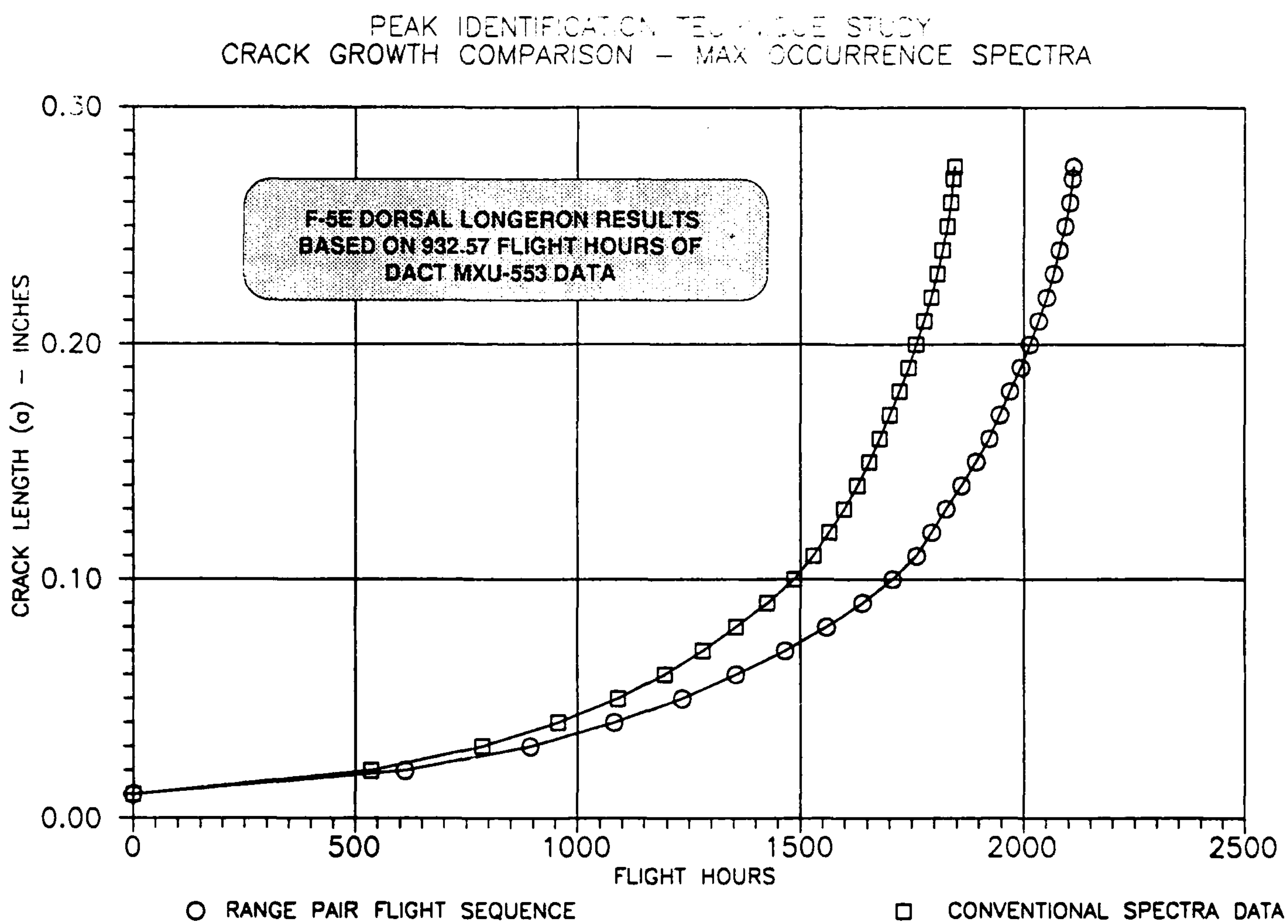
PEAK IDENTIFICATION TECHNIQUE STUDY
NORMAL LOAD FACTOR (Nz) SPECTRA COMPARISON



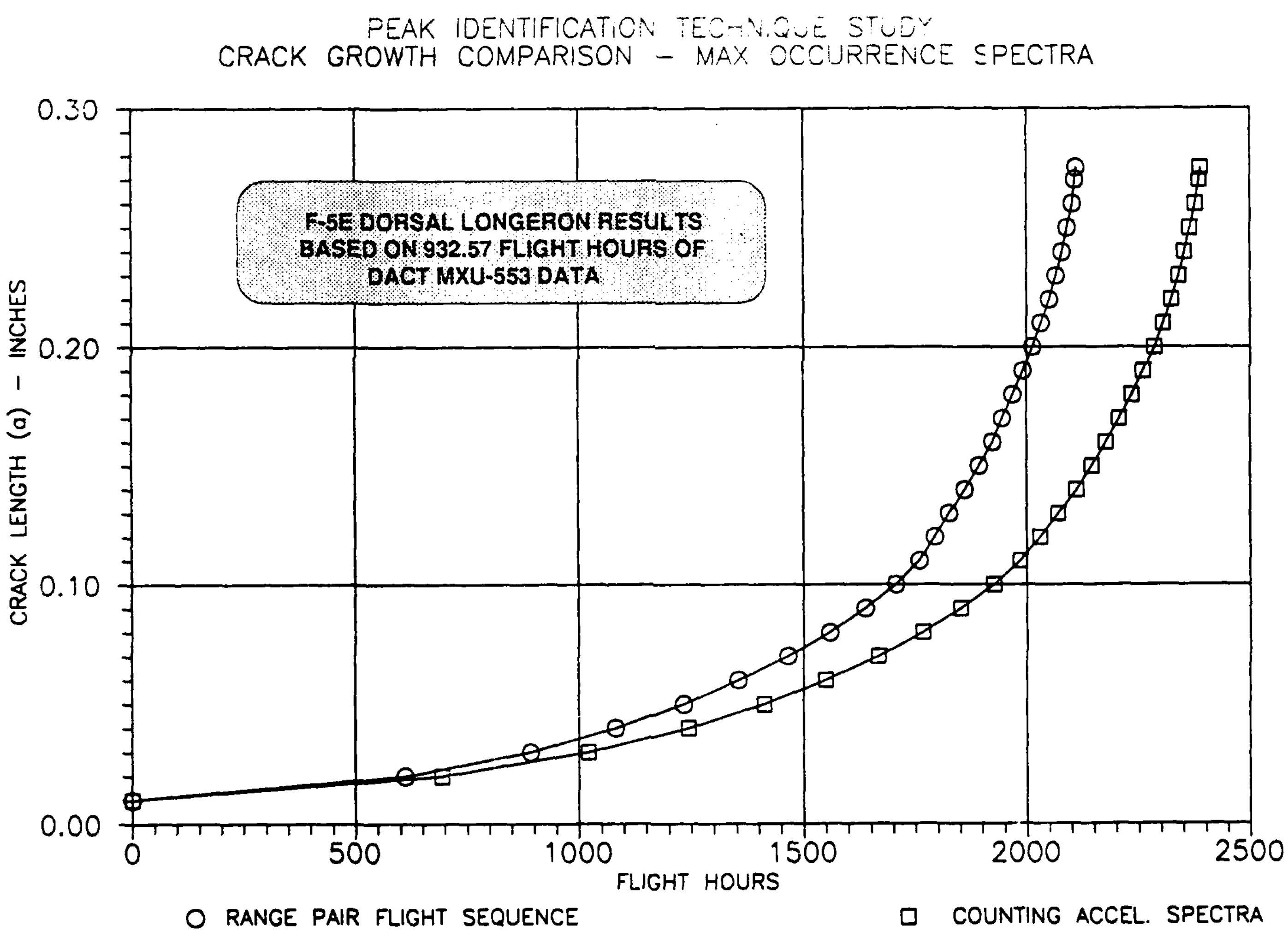
PEAK IDENTIFICATION TECHNIQUES



PEAK IDENTIFICATION TECHNIQUES



PEAK IDENTIFICATION TECHNIQUES



PEAK IDENTIFICATION TECHNIQUES

PEAK Nz OCCURRENCES AT LOWER BOUND (Nz X 1000)

VALLEY (UPPER BOUND) (NzX1000)	-2500	-2000	-1500	-1000	-500	0	500	1000	1500	2000	2500	3000
-2500	0	0	0	0	0	0	0	10	28	3	0	0
-2000		0	0	0	0	0	0	2	2	0	0	2
-1500			0	0	0	0	0	1	4	2	2	2
-1000				0	1	1	0	3	4	3	2	4
-500					1	4	0	11	10	6	13	12
0						10	51	86	103	68	62	73
500							103	2070	1844	1119	824	588
1000								12973	15631	6087	3559	2612
1500									3477	4307	2496	1674
2000										1444	2253	1386
2500											1086	1880
3000												940
3500												
4000												
4500												
5000												
5500												
6000												
6500												
7000												
7500												
8000												

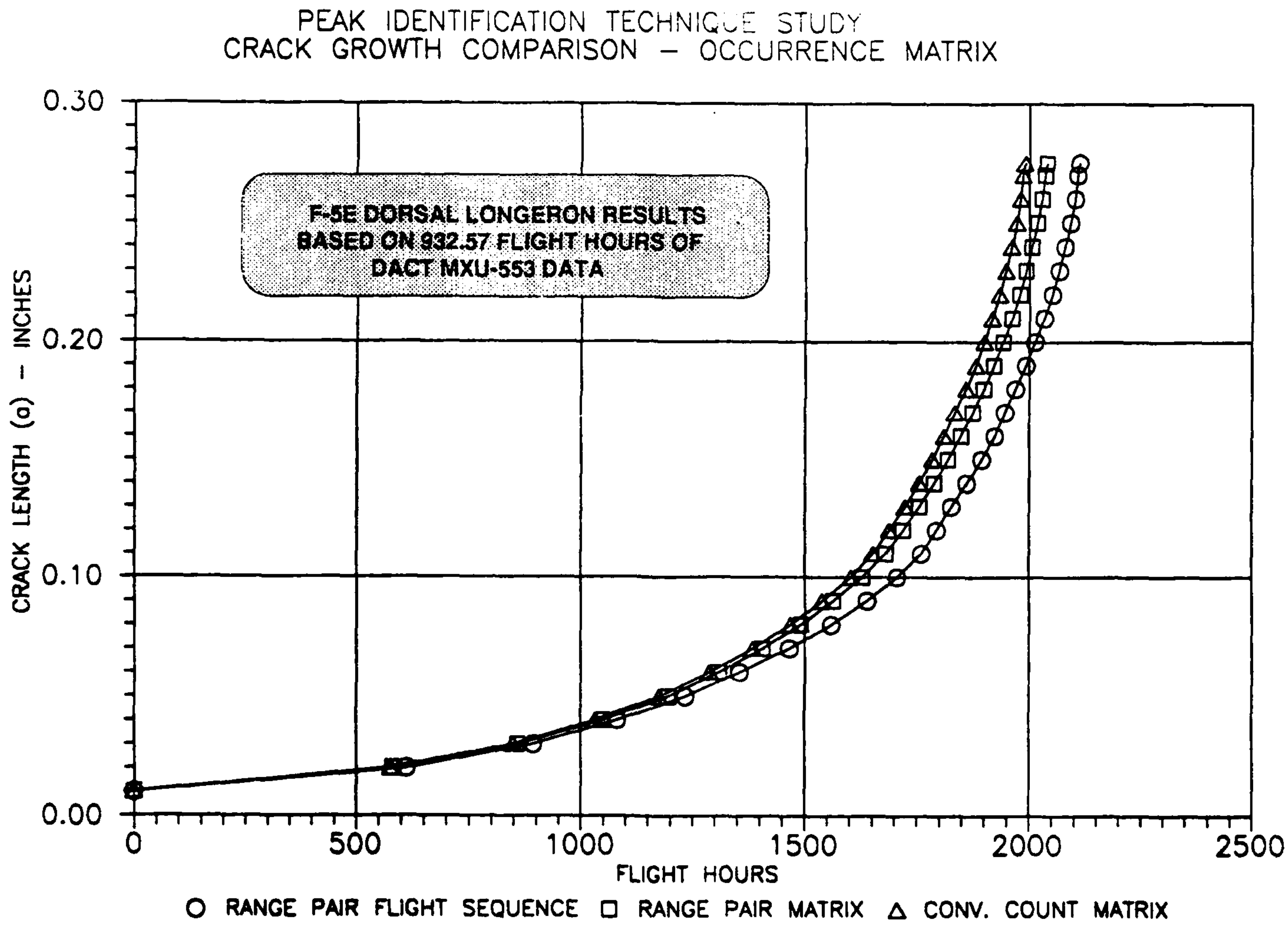
BASED ON 932.57 FLIGHT HOURS OF DACT MXU-553 DATA

PEAK IDENTIFICATION TECHNIQUES

VALLEY (UPPER BOUND) (NzX1000)	PEAK Nz OCCURRENCES AT LOWER BOUND (Nz X 1000) 3500	4000	4500	5000	5500	6000	6500	7000	7500	8000
-2500	0	0	0	0	0	0	0	0	0	0
-2000	0	0	0	0	0	0	0	0	0	0
-1500	0	0	1	0	0	0	0	0	0	0
-1000	1	5	4	2	2	1	1	0	0	0
-500	4	4	4	6	3	4	4	1	0	0
0	56	57	38	50	14	15	7	6	0	0
500	504	395	257	218	150	81	48	19	6	6
1000	1808	1338	860	502	269	166	69	32	21	6
1500	1094	804	508	363	248	95	51	33	17	7
2000	876	577	414	291	184	97	33	29	12	4
2500	1036	660	413	232	164	104	57	18	18	2
3000	1415	841	472	278	178	79	68	10	10	0
3500	697	1153	633	347	199	109	49	23	9	4
4000		495	885	521	310	114	48	30	21	5
4500			460	698	410	179	95	42	28	9
5000				338	595	293	125	49	62	8
5500					225	360	186	83	57	6
6000						132	213	89	73	7
6500							66	116	77	17
7000								34	58	24
7500									27	11
8000										2

BASED ON 932.57 FLIGHT HOURS OF DACT MXU-553 DATA

PEAK IDENTIFICATION TECHNIQUES



PEAK IDENTIFICATION TECHNIQUES

CONCLUSIONS

WHEN USING FLIGHT SEQUENCES

- EITHER RANGE PAIR OR CONVENTIONAL PEAK IDENTIFICATION TECHNIQUES PROVIDE ADEQUATE RESULTS
- CONVENTIONAL PEAK IDENTIFICATION RESULTS IN HALF AS MANY MAX/MIN VALUES WHICH ARE RETAINED

STRUCTURAL LIFE ANALYSIS RESULTS ARE HIGHLY DEPENDENT ON PEAK IDENTIFICATION AND SUBSEQUENT PROCESSING/EXPRESSION OF RECORDED RESULTS

PEAK IDENTIFICATION TECHNIQUES

CONCLUSIONS

OCCURRENCE MATRICIES PROVIDE THE BEST METHOD FOR THE EXPRESSION OF OCCURRENCE SPECTRA

MAXIMUM OCCURRENCE SPECTRA USING THE CONVENTIONAL COUNT METHOD RESULTS IN CONSERVATIVE BUT ACCEPTABLE ANALYSIS RESULTS

UNACCEPTABLE RESULTS ARE OBTAINED FOR MAXIMUM OCCURRENCE SPECTRA USING RANGE PAIR (EXCESSIVELY CONSERVATIVE) OR COUNTING ACCELEROMETER (UNCONSERVATIVE) PEAK IDENTIFICATION METHODS

PEAK IDENTIFICATION TECHNIQUES

CONCLUSIONS

COMPARISON OF PLOTTED SPECTRA RESULTS ARE MISLEADING WHEN SPECTRA ARE OBTAINED BY DIFFERENT PEAK IDENTIFICATION METHODS

PEAK IDENTIFICATION TECHNIQUES

RECOMMENDATIONS

USE OF PEAK IDENTIFICATION METHODS

- USE IDENTIFIED MAX AND MIN VALUES WHEREVER POSSIBLE
- WHEN MAXIMUM OCCURRENCE SPECTRA MUST BE USED, BE AWARE OF THE IMPLICATIONS AND CHOOSE METHODS ACCORDINGLY

PEAK IDENTIFICATION TECHNIQUES

RECOMMENDATIONS

ADDITIONAL INVESTIGATIONS

- OTHER F-5E STRUCTURAL LOCATIONS
  - WING - NON-LINEAR STRESS VARIATION WITH N_z
  - FORWARD FUSELAGE - COCKPIT PRESSURE/N_z SENSITIVE
  - VERTICAL STABILIZER - PRIMARILY LATERAL LOAD FACTOR (N_y) DEPENDENT
- OTHER USAGE ENVIRONMENTS
  - ATTACK AIRCRAFT (A-10, A-37)
  - TRAINER AIRCRAFT (T-38 ATC, T-37)
  - SUPPORT AIRCRAFT (OV-10)

The Edit/Pre-Analysis Program
ASIP/ENSIP Conference
December 3, 1987

Alamo Technology, Inc.
Devin Butts, Software Engineer
Computer Resources & Software
Development Department
Engineering Division
San Antonio, Texas
(512) 270-4513

Maneuver Spectra: The Edit/Pre-Analysis Program

DESCRIPTION OF SERVICES: ATI will provide Engineering Services to develop the required effort. Computer software will be generated to provide data compression, maneuver spectra and operational usage profiles utilizing MXU-553 or microprocessor data from F-5, T-38, T-37 and OA-37 flight loads data (FLD) recorder programs. The software will minimize man-machine interface when processing the FLD data. The software will provide graphic displays of compressed data, allow for mission and mission segment identification from data displayed, distinguish between maneuver load factors and gust load factors, assure that FLD parameters recorded are within acceptable ranges and account for flight data parameters as specified. F-5E - DACT Usage Assessment Task will be composed of the work described below.

- An analysis of F-5E MXU-553 data from the DACT environment will be performed. This analysis will include validation, mission identification, and data compression of the given MXU-553 data.
- Develop F-5E wing stress spectra at two wing locations. These stress spectra will be based on counting accelerometer normal load factor data, Nellis Air Force Base survey information, and MXU-553 data analyzed.
- Develop F-5E vertical stabilizer root bending moment spectra. These spectra will be determined from MXU-553 data analyzed.

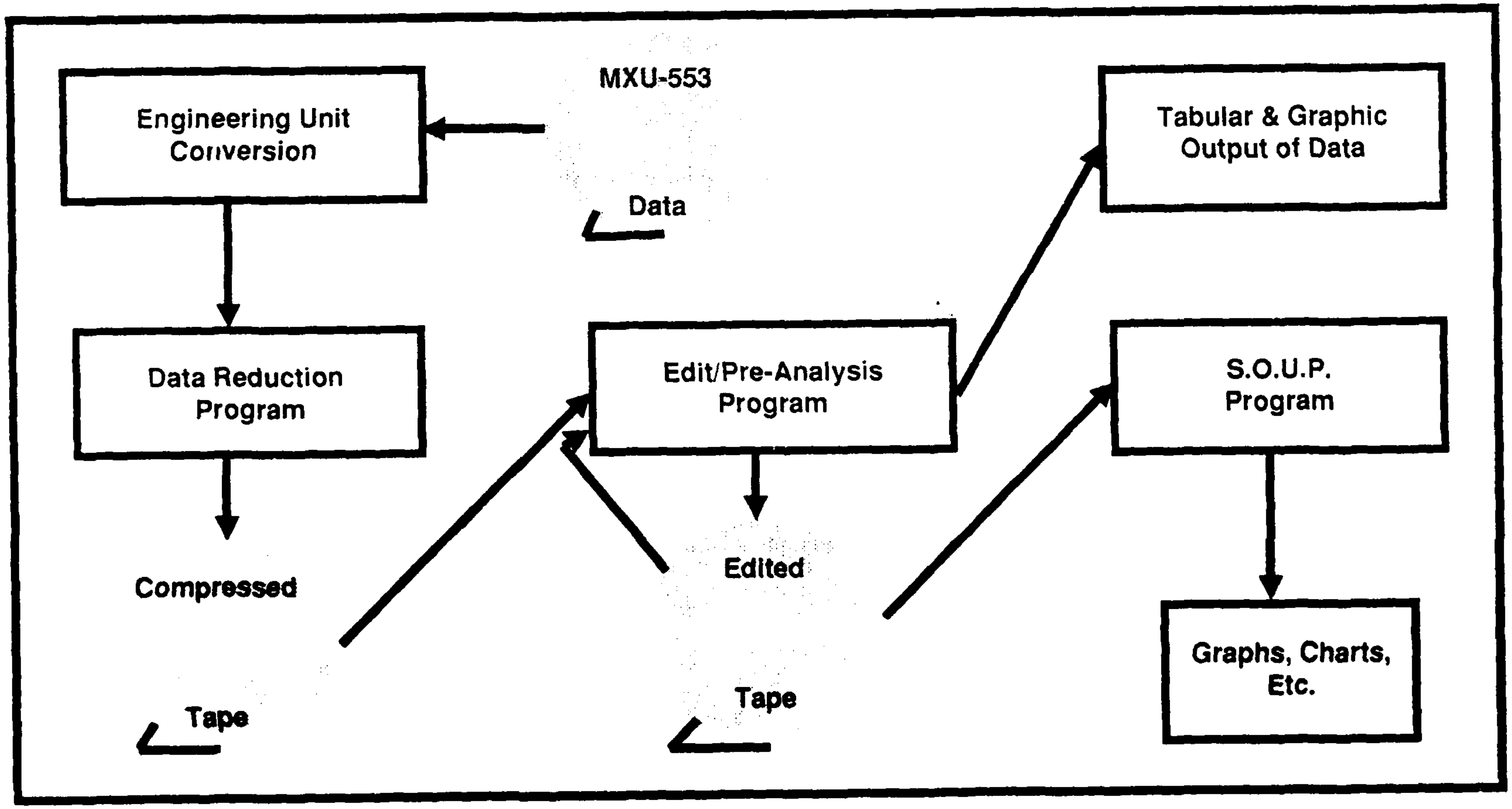
Maneuver Spectra: The Edit/Pre-Analysis Program

Introduction

- **Alamo Technology, Inc.**
  - The Maneuver Spectra Program (MSP)
    - Data Reduction Program
    - Edit/Pre-Analysis Program
    - Automated Spectra Operational Usage Profiles Program

Maneuver Spectra: The Edit/Pre-Analysis Program

Program Interaction



Maneuver Spectra: The Edit/Pre-Analysis Program

Maneuver Spectra Data Base

MXU-553 Data

- F-5E 932.57 hours
- T-37 825.18 hours
- OA-37 678.21 hours
- T-38 503.18 hours
- Header Record
- Flight Records
  - Nz Peaks & Valleys
  - Ny Peaks & Valleys
  - Periodic (30 Second)

Maneuver Spectra: The Edit/Pre-Analysis Program

Edit/Pre-Analysis Program

- Edit
- Pre-Analysis

Edit Portion

Data in Tabular Format

Edit/Pre-Analysis

?5E 2-DEC-87

FLT#	A/C-#	GWTO LB/10	FWTO LBS	FWLD LBS	MISS ID	BASE ID	CONF	DATE MMDDY	DUR SEC	VTMX SW
5	1551	1558	4410	1000	1	3	500	4265	2221	1

Edit/Pre-Analysis Tape: AC275A 5 Flight: flt005 2-DEC-87 Page: 2

TIME SEC	EVT 1	EVT 2	ALT FT /10	MACH *100	GW LBS	NZ G's *1000	NY G's *10000	Q D/S *100	P D/S *10	R D/S *100	Q-DOT D/S2 *100	P-DOT D/S2 *10	R-DOT D/S2 *100	DEL-H DEG *100	SGS PSI /100	DEL-R DEG *100	DEL-F CODE	VTMX IN-LB /100	RET COD	ERR COD	MS ID
30	0	0	234	52	15533	1285	-156	137	-32	62	-315	23	105	2366	-304	-2	0	66	5	0	1
52	0	0	346	59	15500	2102	-313	-244	46	-11	787	23	0	2366	-304	-52	0	96	1	0	1
56	0	0	369	61	15494	1530	-392	-15	-111	62	-105	116	0	2325	-304	23	0	231	3	0	1
60	0	0	436	59	15487	1653	-78	-15	77	62	157	-163	0	2366	-304	-27	0	29	5	0	1
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91	0	0	774	56	15440	1163	78	137	-1	173	-157	46	0	2366	-304	-27	0	-9	2	0	1
94	0	0	796	56	15435	1489	1411	23	-457	247	-315	-1539	-525	2305	-304	48	0	-457	4	0	1
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[illegible]	[illegible]	[illegible]	1137	67	15349	1530	0	23	30	136	0	0	0	2366	-304	23	0	0	5	0	1
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[illegible]	[illegible]	[illegible]	1263	66	15335	2265	-1176	-15	329	99	682	1119	420	2305	-304	-52	0	320	3	0	1
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[illegible]	[illegible]	[illegible]	1353	64	15331	1448	862	-283	30	-270	-945	-653	-420	2263	-304	98	0	-382	2	0	1
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[illegible]	[illegible]	[illegible]	1427	62	15325	1571	-705	-206	-237	802	105	536	525	2325	-304	-27	0	471	3	0	1
[illegible]	[illegible]	[illegible]	1447	62	15320	1122	156	-91	109	395	0	0	0	2387	-304	48	0	-71	2	0	1
[illegible]	[illegible]	[illegible]	1449	62	15318	1326	548	-321	109	358	-787	-466	0	2243	-304	23	0	-227	4	0	1

Edit/Pre-Analysis

[illegible] 2-DEC-87

FLT#	A/C-#	GWTO LB/10	FWTO LBS	FWLD LBS	MISS ID	BASE ID	CONF	DATE MMDDY	DUR SEC	VTMX SW
5	[illegible]	1551	[illegible]	1000	1	3	500	4265	[illegible]	1

Edit/Pre-Analysis Tape: AC275A 5 Flight: flt005 2-DEC-87 Page:

TIME SEC	EVT 1	EVT 2	ALT FT /10	MACH *100	GW LBS	NZ G's *1000	NY G's *10000	Q D/S *100	P D/S *10	R D/S *100	Q-DOT D/S2 *100	P-DOT D/S2 *10	R-DOT D/S2 *100	DEL-H DEG *100	SGS PSI /100	DEL-R DEG *100	DEL-F CODE	VTMX IN-LB /100	RET COD	ERR COD
30	0	0	234	52	15533	1285	-156	137	-32	62	-315	23	105	2366	-304	-2	0	66	5	0
52	0	0	346	59	15500	2102	-313	-244	46	-11	787	23	0	2366	-304	-52	0	96	1	0
56	0	0	369	61	15494	1530	-392	-15	-111	62	-105	116	0	2325	-304	23	0	231	3	0
60	0	0	436	59	15487	1653	-78	-15	77	62	157	-163	0	2366	-304	-27	0	29	5	0
90	0	0	751	55	15441	1530	0	-53	-48	173	105	116	0	2305	-304	23	0	26	5	0
91	0	0	774	56	15440	1163	78	137	-1	173	-157	46	0	2366	-304	-27	0	-9	2	0
94	0	0	796	56	15435	1489	1411	23	-457	247	-315	-1539	-525	2305	-304	48	0	-457	4	0
120	0	0	954	59	15395	1326	0	99	-1	136	-157	0	0	2366	-304	-2	0	16	5	0
140	0	0	1044	66	15365	1938	78	-129	-16	99	0	0	0	2325	-304	-2	0	-13	1	0
144	0	0	1089	66	15358	1285	-78	99	14	173	0	0	105	2387	-304	-2	0	45	2	0
151	0	0	1197	67	15349	1530	0	23	30	136	0	0	0	2366	-304	23	0	0	5	0
158	0	0	1247	67	15337	2510	235	-321	-16	247	105	0	105	2243	-304	23	0	-47	1	0
159	0	0	1269	66	15335	2265	-1176	-15	329	99	682	1119	420	2305	-304	-52	0	320	3	0
161	0	0	1337	64	15332	1530	627	-244	250	-344	0	-816	-262	2366	-304	48	0	-396	4	0
162	0	0	1359	64	15331	1448	862	-283	30	-270	-945	-653	-420	2263	-304	98	0	-382	2	0
163	0	0	1359	64	15329	2428	1333	-665	-410	210	1207	-163	-682	2305	-304	223	0	-323	1	0
166	0	0	1427	62	15325	1571	-705	-206	-237	802	105	536	525	2325	-304	-27	0	471	3	0
169	0	0	1449	62	15320	1122	156	-91	109	395	0	0	0	2387	-304	48	0	-71	2	0
170	0	0	1449	62	15318	1326	548	-321	109	358	-787	-466	0	2243	-304	23	0	-227	4	0
177	0	0	1449	64	15308	2795	-78	-397	235	-233	682	-186	157	2222	-304	-52	0	-87	1	0
180	0	0	1449	64	15303	2632	313	-359	-1	-48	157	-23	-105	2202	-304	73	0	-118	5	0
210	0	0	1719	64	15257	1489	0	61	-16	99	-157	69	0	2346	-304	48	0	20	5	0
240	0	0	1922	68	15211	1530	156	61	-32	136	105	-256	-105	2366	-304	48	0	-39	5	0
270	0	0	2147	70	15165	1448	0	61	-16	173	0	0	105	2366	-304	73	0	32	5	0
300	0	0	2304	75	15119	1489	78	61	-16	173	0	0	0	2387	-304	73	0	-1	5	0
320	0	0	2327	82	15073	1448	0	61	-16	136	-157	0	0	2387	-304	73	0	24	5	0
336	0	0	2327	83	15064	1204	78	137	-16	173	0	0	0	2428	-304	73	0	-4	2	0
[illegible]	[illegible]	0	2304	87	15027	1571	156	23	14	136	157	-46	0	2510	-304	73	0	-46	5	0
[illegible]	[illegible]	0	2304	86	15007	1816	-470	-53	-64	358	0	396	1207	2469	-304	348	0	351	3	0
[illegible]	[illegible]	0	2327	85	15005	2020	2195	-53	124	25	105	256	157	2551	-304	921	0	-630	1	0
[illegible]	[illegible]	0	2327	82	15001	1897	2588	23	235	-159	-262	-746	-420	2510	-304	696	0	-923	4	0
[illegible]	[illegible]	0	2342	76	14991	1122	1019	99	-268	99	-157	-466	-157	2387	-304	447	0	-165	2	0

Maneuver Spectra: The Edit/Pre-Analysis Program

Edit Portion

Data Review

- Movement
  - Begin
  - End
  - Page +/-
  - Second
- Data Search
  - Given Value
  - Largest Value
  - Smallest Value
- Help

Maneuver Spectra: The Edit/Pre-Analysis Program

Edit Portion

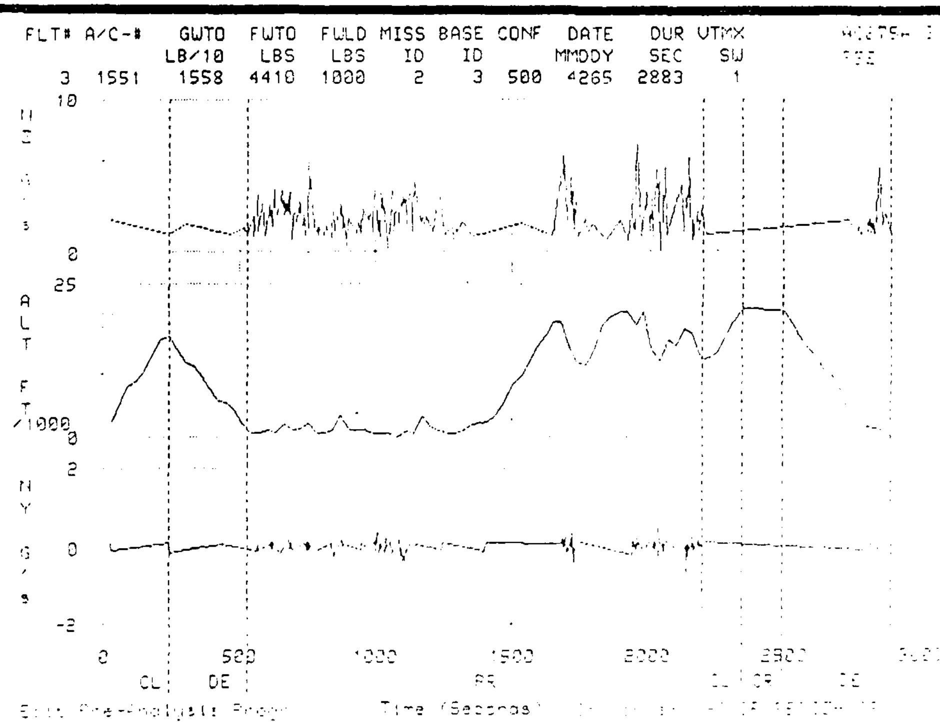
Data Correction

- Header Record
- Flight Records
  - Recalculation
    - Vertical Tail Bending Moment (VTMx)
    - Gross Weight
- Single Values
- Multiple Values
  - Column
  - Old Value
- Data Search
  - Given Value
  - Largest Value
  - Smallest Value
- Help

Maneuver Spectra: The Edit/Pre-Analysis Program

Pre-Analysis Portion

Graphic Data Display



FLT#	A/C-#	GWTO LB/10	FWTO LBS	FWLD LBS	MISS ID	BASE ID	CONF	DATE MMDDY	DUR SEC	VTMX SW
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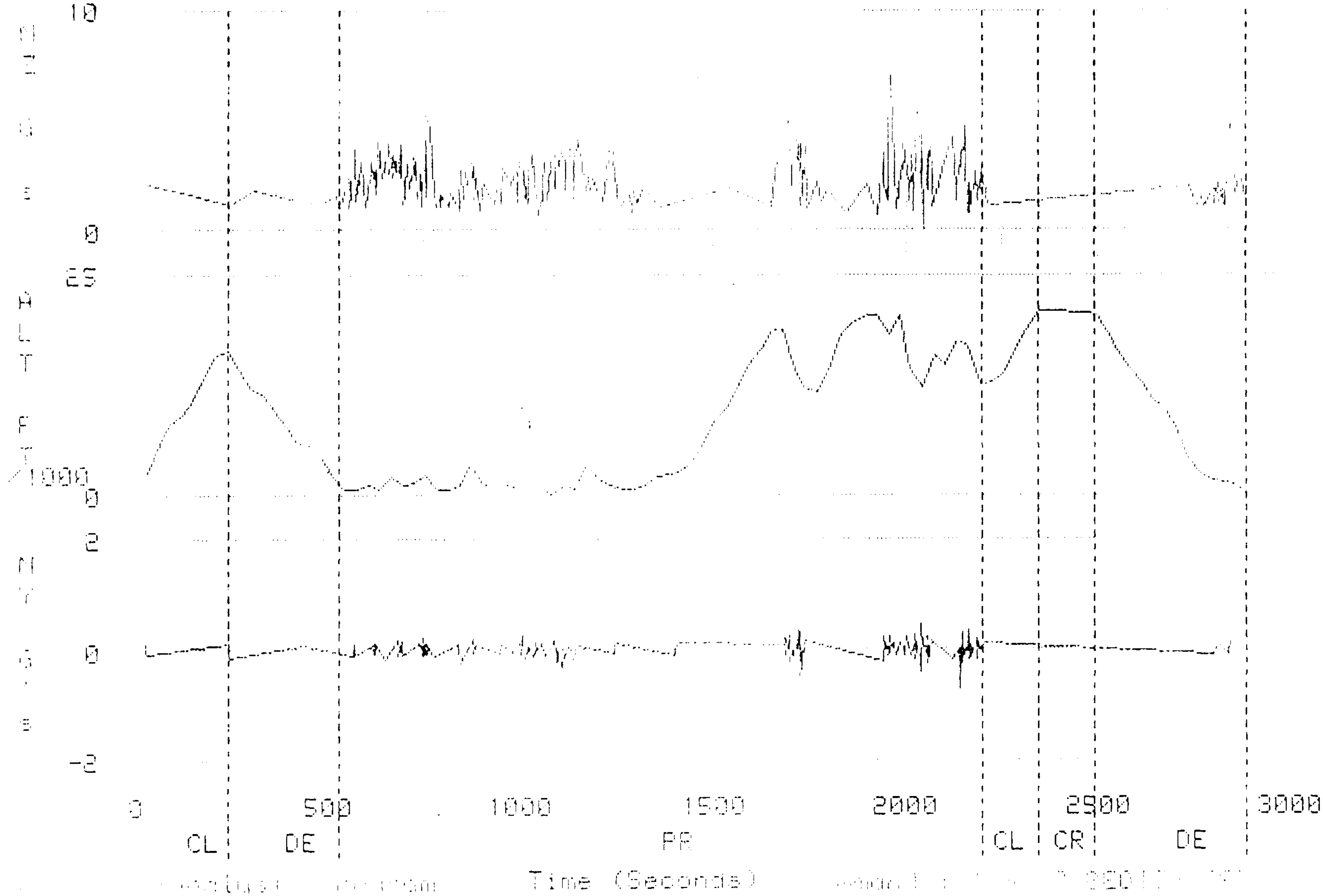


ALAMO TECHNOLOGY, INC.
Defense and Space Systems Engineering

Maneuver Spectra: The Edit/Pre-Analysis Program

Pre-Analysis Portion

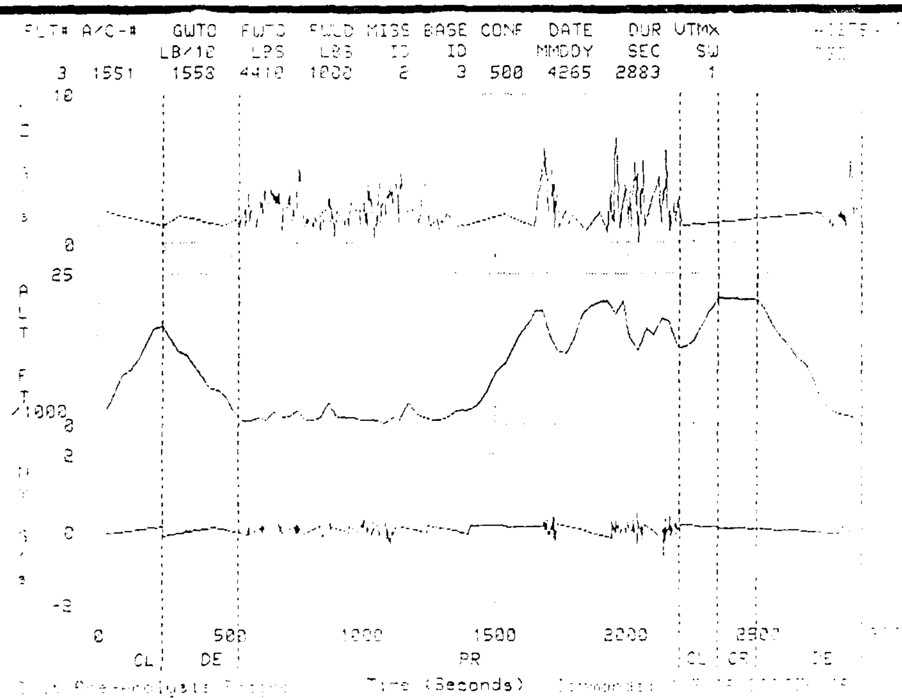
Plot Set-Up

- Number of Plots Per Screen
- Number of Plot Items Per Plot
- Plot Items
- Record Types Included
- Header Record
- Grid & Tick Marks
- Help

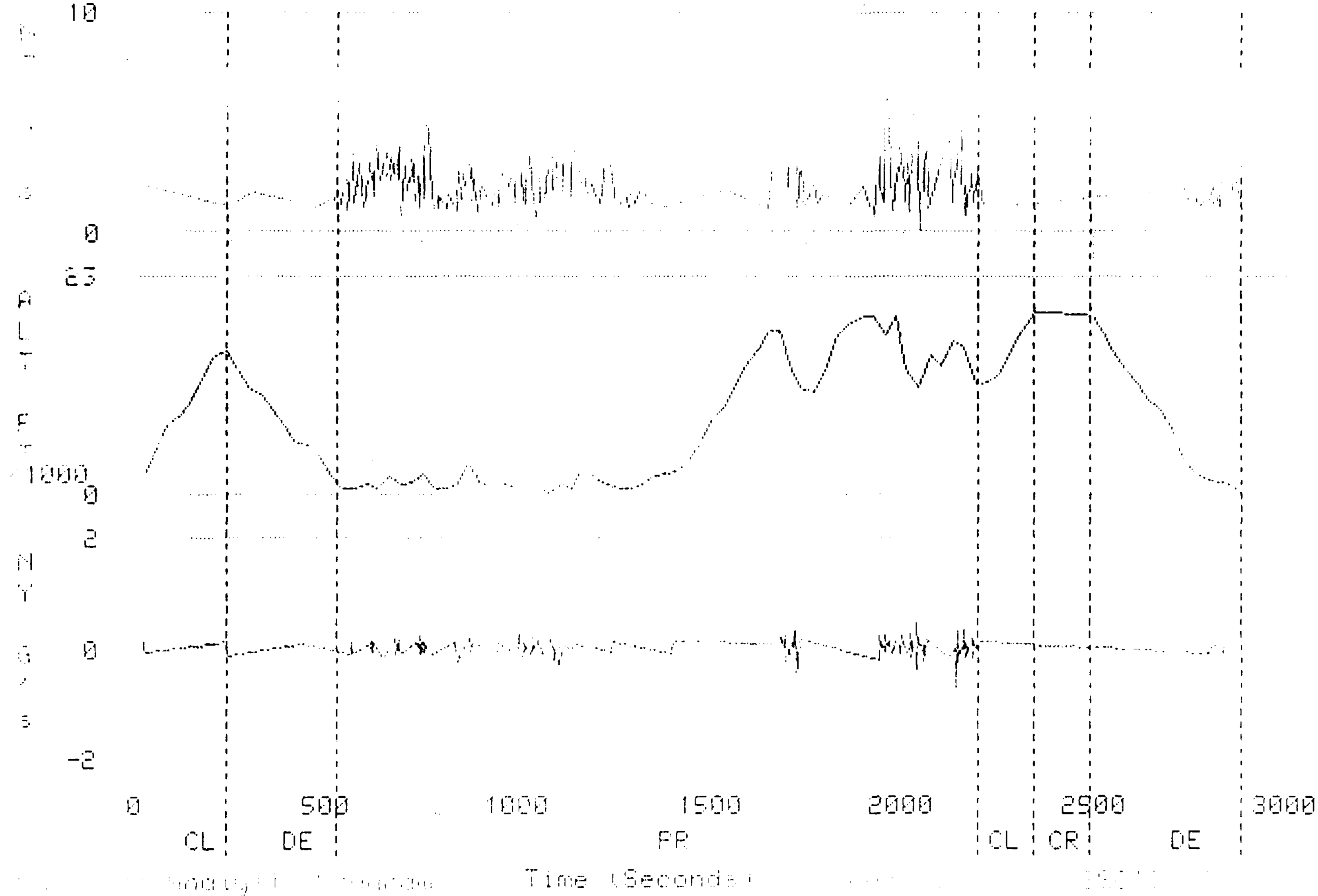
Maneuver Spectra: The Edit/Pre-Analysis Program

Pre-Analysis Portion

Major Mission Segments



FLT#	A/C-#	GWTO	FWTO	FWLD	MISS	BASE	CONF	DATE	DUR	VTMX
		LB/10	LBS	LBS	ID	ID		MMDDY	SEC	SW
3	1551	1558	4410	1000	2	3	500	4265	2883	1



ALAMO TECHNOLOGY, INC.

Maneuver Spectra: The Edit/Pre-Analysis Program

Pre-Analysis Portion

Cross-Hair Cursor

- Data Return Location

Mission Segments

- Climb
- Cruise
- Primary
- Descent
- Pattern

Maneuver Spectra: The Edit/Pre-Analysis Program

Pre-Analysis Portion

Commands

- Identify Mission
- Beginning
- End
- 1,2,3,4,& 5 Mission Segments
- Delete Segment Forward
- Delete Segment Backwards
- Help

Maneuver Spectra: The Edit/Pre-Analysis Program

Edit & Pre-Analysis Portions

File Operations

- Flight Data Files
  - Load
  - Save
  - Print
- Configuration Files
  - Load
  - Save
  - Print
- Command Line
- Help

Maneuver Spectra: The Edit/Pre-Analysis Program

Help Screens

```
Edit_Pre-Analysis

   The EDIT/PRE-ANALYSIS program is used to revise com-
   pressed data from the Data Compression Program (REDUCT).
   It allows the user to edit and display aircraft data in
   both tabular and graphic forms.

 Additional information available:

 EPRE       Features

Edit_Pre-Analysis Subtopic?
```

Maneuver Spectra: The Edit/Pre-Analysis Program

Conclusion

Examine

- Tabular
- Graphic

Correct

Mark Major Mission Segments

Potential Applications

- Standardized Flight Data Recorder

Presentation Abstract for
Maneuver Spectra: THE EDIT/PRE-ANALYSIS PROGRAM

The topic of this presentation, "Maneuver Spectra: The EDIT/PRE-ANALYSIS Program" deals with the computer software program called the EDIT/PRE-ANALYSIS Program. This program is the second of three programs developed for the San Antonio Air Logistics Center by Alamo Technology, Inc. (ATI).

The EDIT/PRE-ANALYSIS (EPRE) Program is an interactive utility program which allows the user to make multiple revisions to compressed MXU-553 data. It has also been tailored to accept microprocessor data which will extend its usefulness. The program also provides a convenient method of marking the major mission segments of the flight data.

The pre-processing program for EPRE is the DATA COMPRESSION Program. It reduces the amount of flight data by retaining only the records associated with structural stress peaks and periodic times necessary for mission identification.

The post-processing program to EPRE is the AUTOMATED SPECTRA AND OPERATIONAL USAGE PROFILES Program. It computes and presents maneuver spectra data.

EPRE uses the Aircraft Flight Database developed by ATI and currently accepts data from several different aircraft. The aircraft include the F-5E, T-37, OA-37, and T-38. The program was designed so that new aircraft types can be easily added.

EPRE is divided into two major parts: Edit and Pre-Analysis. The Edit portion displays the flight data in tabular form, much like the popular spreadsheet programs for personal computers. It allows the user to change any value in the flight data or in the header record that accompanies each flight file. If the user changes a parameter value that is used in the calculation of the Vertical Tail Bending Moment (VTMx), VTMx is automatically recalculated. Also, since the Gross Weight is calculated linearly, if the user changes a Gross Weight Value, Gross Weight for the remainder of the flight is recalculated. The same is true if the user changes the Gross Weight at Take-Off, Fuel Weight at Take-Off, or the Fuel Weight at Landing parameters in the header record.

The Pre-Analysis portion of the program allows the user to see the data displayed graphically. Through the use of screen prompts, the user can define which parameters he wants to see graphed. The program will display up to three graphs at once on the screen and up to three parameters per graph. It also allows the user to choose to have the header record information displayed or to use that space to increase the resolution of the plots. The user can save the configuration of plots and even develop a library of configurations.

Another important part of the Pre-Analysis section of EPRE is its capability to easily mark the major mission segments. The user can move a vertical line back and forth across the screen and mark the Climb, Cruise, Primary, Descent, Pattern, and Unknown flight segments. The data is automatically updated as the user marks each segment.

Through EPRE the user can examine and change aircraft flight data in both its numeric and graphic forms. The EDIT/PRE-ANALYSIS program facilitates the evaluation of aircraft flight data in a way never before realized by the Air Force. Data Channels which appear questionable, have been easily verified using EPRE. Perhaps the most impressive quality of the EPRE Program is the interactive color graphic user interface that is easy to learn and thorough in its application; at the same time, it maintains the users interest in a tedious task. The EDIT/PRE-ANALYSIS program has a bright and useful future, not only just for SA-ALC, but other ALC's as well.

CRASH SURVIVABLE FLIGHT DATA RECORDER
(CSFDR) SYSTEM
FOR
F-16 ASIP FORCE MANAGEMENT

PRESENTED BY

JANET WEISS

F-16 PROGRAM MANAGER/PROJECT ENCINEER
CFSDR SYSTEM/FORCE MANAGEMENT

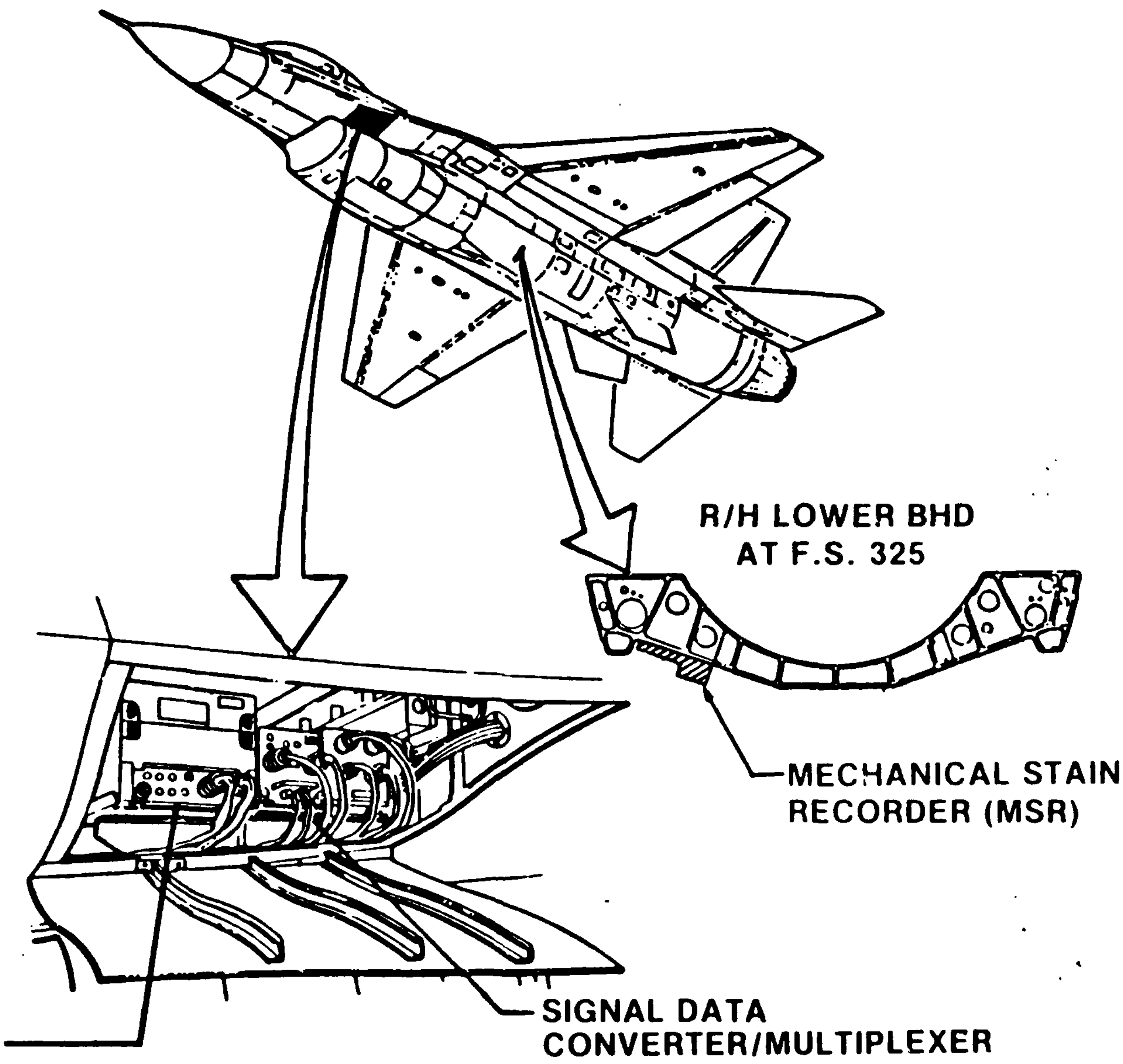
ASD/YPEF
WRIGHT-PATTERSON AIR FORCE BASE, OH

1-3 DECEMBER 1987

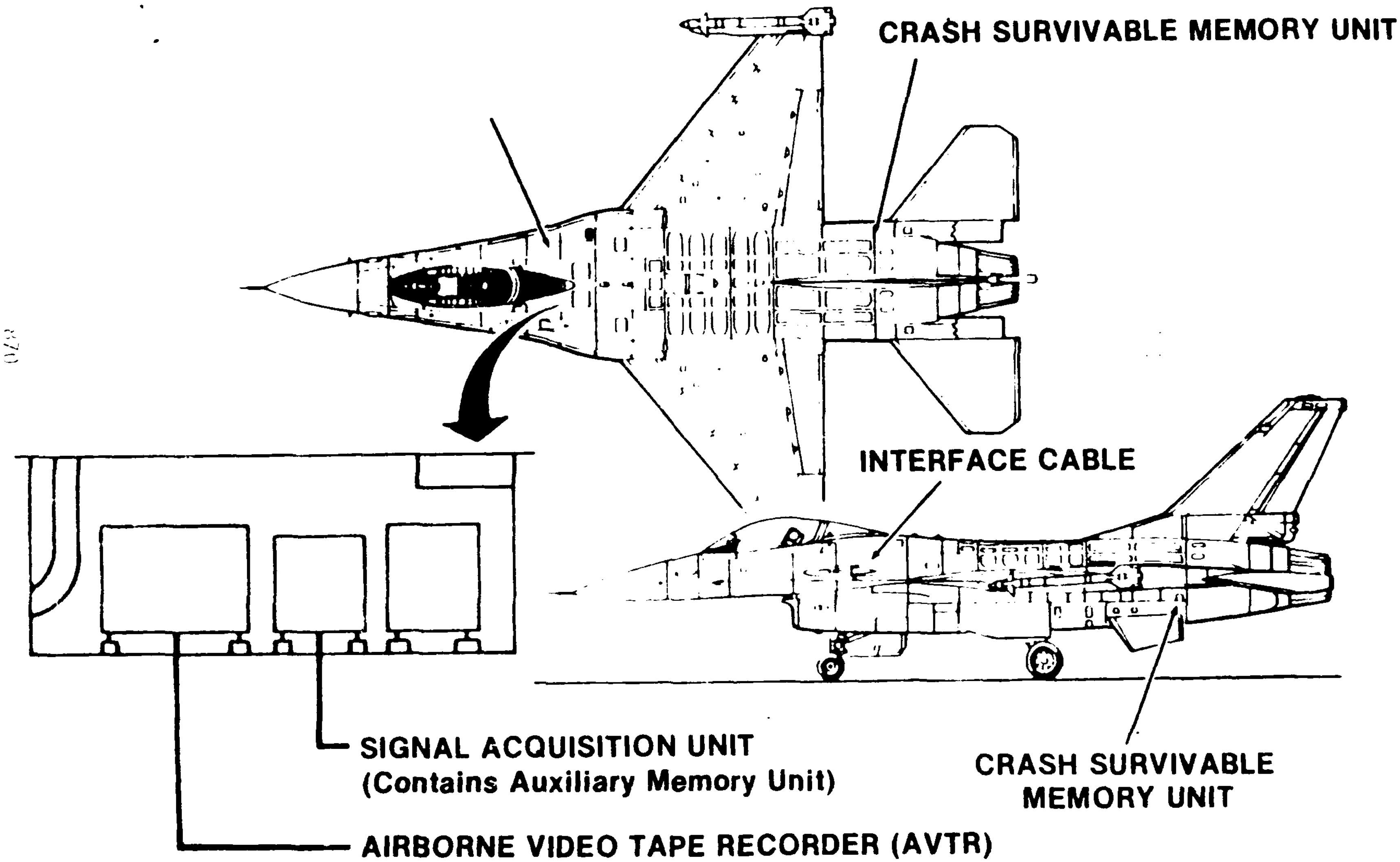
OUTLINE

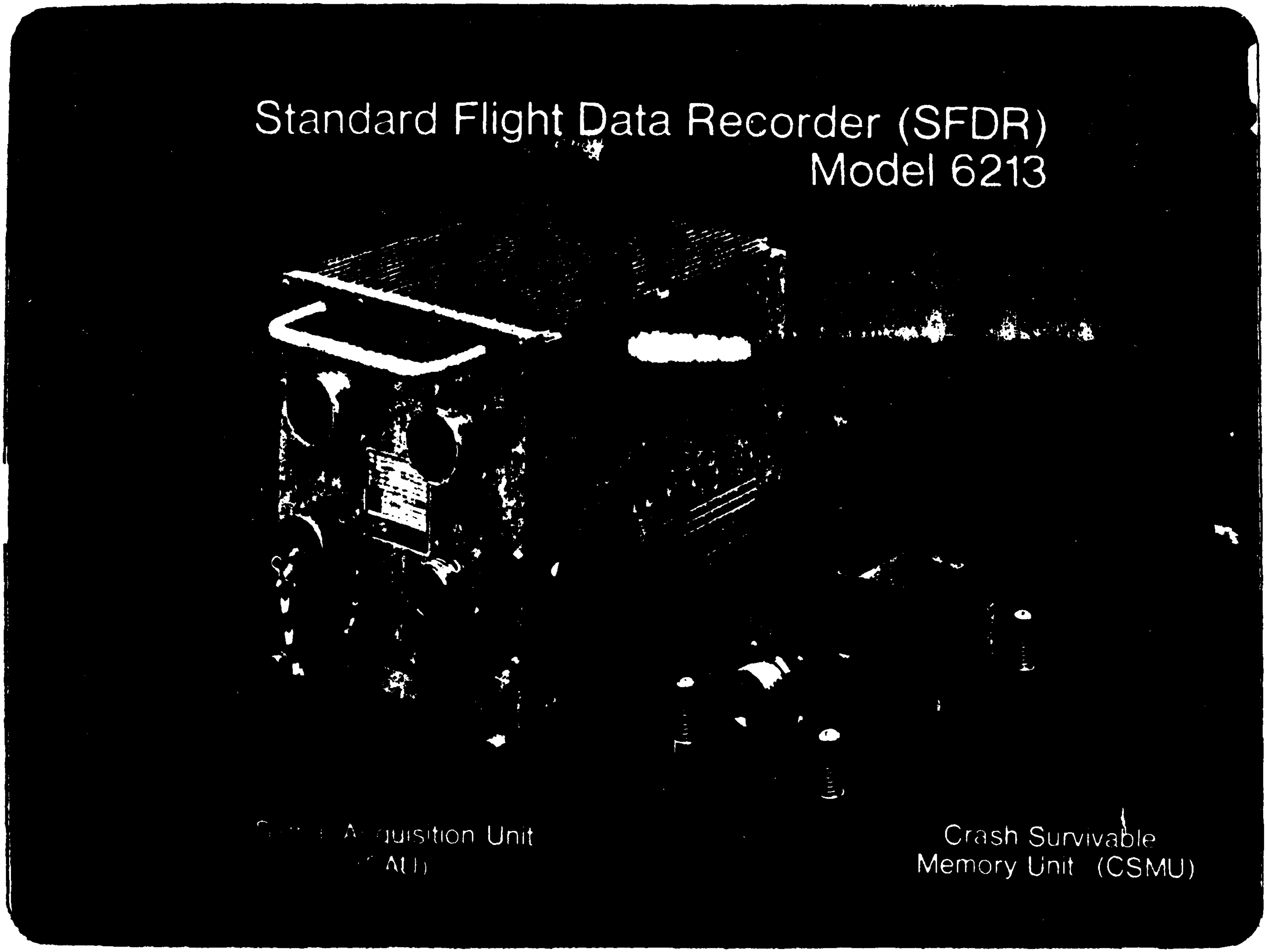
- F-16 RECORDING SYSTEMS
  - HARDWARE
  - PARAMETERS
- F-16C/D DATA FLOW
- F-16C/D DATA PROCESSING (ENGINEERING WORKSTATION)
- CONCLUSIONS

FLR AND MSR LOCATIONS



F-16 FLIGHT DATA RECORDER SYSTEM INSTALLATION
USAF F-16C/D AIRCRAFT

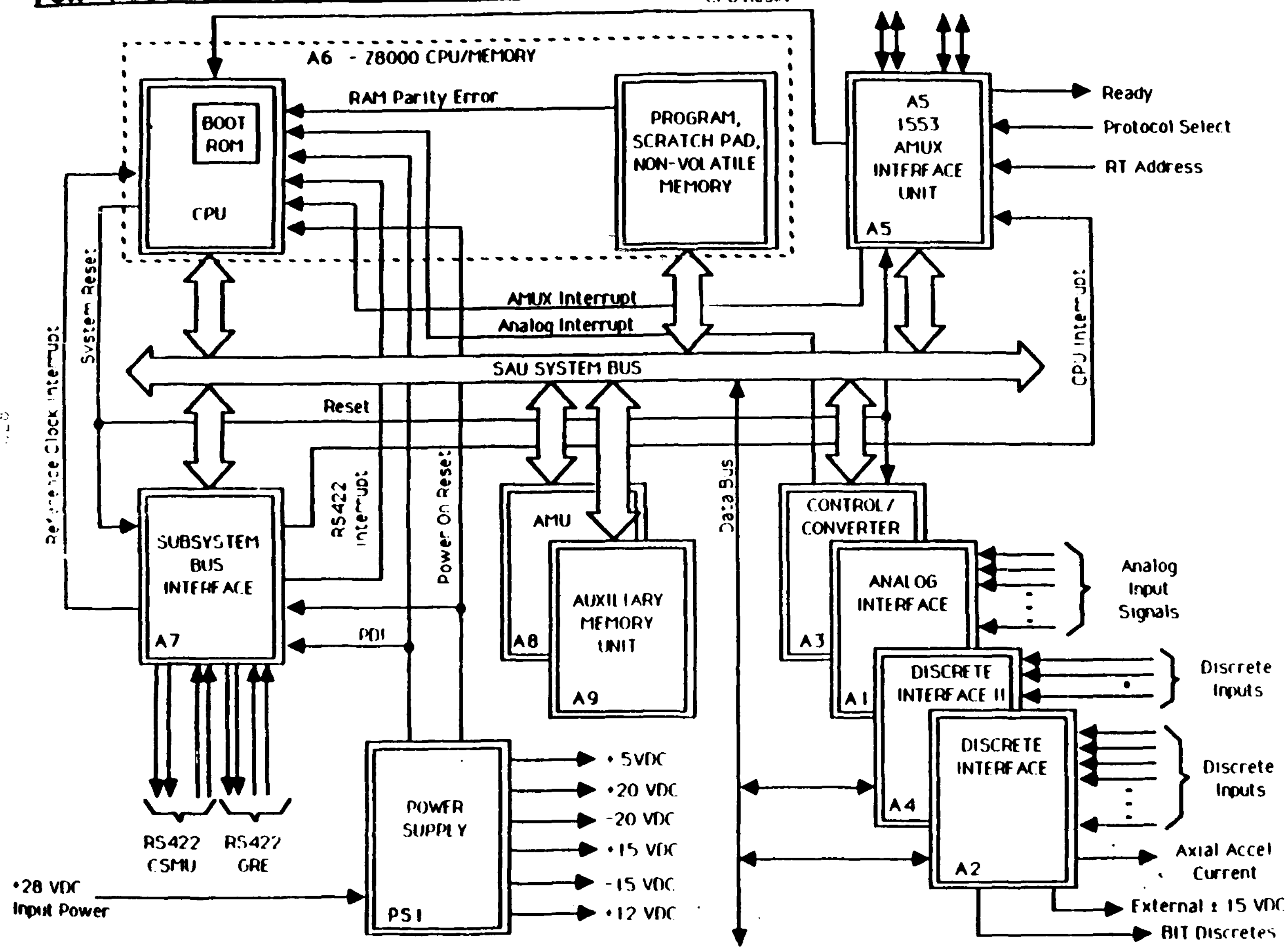



Standard Flight Data Recorder (SFDR)
Model 6213
Crash Survivable
Memory Unit (CSMU)

CSMU SURVIVABILITY

- **IMPACT SHOCK** - 200G PEAK, 115 MSEC DURATION LOW LEVEL SHOCK
  - 1700G PEAK, 5-8 MSEC DURATION HIGH LEVEL SHOCK
- **PENTRATION** - WITHSTAND IMPACT FORCE EQUAL TO A 500LB STEEL PIN DROPPED FROM 10 FEET ON TO EACH SIDE
- **STATIC CRUSH** - 5000 LBS APPLIED TO EACH AXIS FOR 5 MINUTES
- **FIRE** - 1100°C MIN AT A THERMAL FLUX OF 50,000 $BTU/FT^2/HR$ FOR 30 MINUTES MIN
- **FLUID IMMERSION** - SEAWATER FOR 48 HRS
  - JP1 FOR 24 HRS
  - JP4 FOR 24 HRS
  - LUBRICATION OIL FOR 24 HRS
  - HYDRAULIC FLUID FOR 24 HRS
  - WATERGLYCOL FIRE EXTINGUISHING FLUID FOR 24 HRS
  - CO_2 FOAM FIRE FIGHTING MATERIAL FOR 24 HRS
  - FIRE EXTINGUISHING FOAM CONCENTRATE FOR 24 HRS

SAU BLOCK DIAGRAM FOR POST BLOCK 40 AIRCRAFT



RECORDING SYSTEMS

- o CRASH SURVIVABLE FLIGHT DATA RECORDER (CSFDR)
  - o SIGNAL ACQUISITION UNIT (SAU) ON EVERY AIRPLANE
  - o AUXILIARY MEMORY UNIT (AMU) WITHIN THE SAU ON EVERY AIRPLANE
  - o CRASH SURVIVABLE MEMORY UNIT (CSMU) AN EVERY AIRPLANE
  - o PERFORMS ONBOARD DATA COMPRESSION AND DATA STORAGE

    - TYPE 1 - DATA FOR MISHAP INVESTIGATION ANALYSIS
      - LAST 15 MINUTES OR MORE OF FLIGHT DATA
      - UP TO 5 SPECIAL EVENTS (30 SEC SNAPSHOTS EACH)
      - BASELINE DATA AT LIFTOFF
    - TYPE 2 - DATA FOR INDIVIDUAL AIRPLANE STRUCTURAL TRACKING (NVM IN SAU)
      - NzW EXCEEDANCES AND OTHER USAGE INFO
      - DATA EXTRACTED AT 150-HOUR PHASED INSPECTIONS
    - TYPE 3 - DATA FOR STRUCTURAL LOADS & SERVICE LIFE ANALYSIS (AMU)
      - RETAINS DATA FOR LAST 8 TO 10 FLIGHTS
      - DATA EXTRACTED AT 150-HOUR PHASED INSPECTIONS
    - TYPE 4 - DATA FOR ENGINE USAGE ANALYSIS (AMU)
      - DATA EXTRACTED AT 150-HOUR PHASED INSPECTIONS

TABLE C.4.2-1

F-16C/D CSFDR System Parameters

RECORDED PARAMETERS	SAMPLING RATE/SEC	DATA TYPE 1	2	3	4	SOURCE	DISCRETE SIGNALS	SAMPLING RATE/SEC	DATA TYPE 1	2	3	4
PRESSURE ALTITUDE	1			3	4	MUX	SEC ENGINE CONTROL	1	1			
CORRECTED ALTITUDE	1	1				MUX	ELEC ENGINE CONTROL	1	1			
CALCULATED AIRSPEED	1	1		3		MUX	JFS "START 2"	1	1			
TRUE FREESTREAM AIR TEMP	1	1		3	4	MUX	AIR LIGHT (EPU)	1	1			
ANGLE OF ATTACK	8	1		3		MUX	GEN - MAIN FAIL	1	1			
RADAR ALTITUDE	1	1				MUX	GEN - STANDBY FAIL	1	1			
RADAR ALT LOW SET	1	1				MUX	FLCS POWER TEST SWITCH	1	1			
TRUE HEADING	4	1				MUX	FLCS BRANCH A INV "GOOD"	1	1			
HSI COURSE DEV	1	1				MUX	FLCS BRANCH B INV "GOOD"	1	1			
PITCH ATTITUDE	4	1				MUX	FLCS BRANCH C INV "GOOD"	1	1			
ROLL ATTITUDE	4	1				MUX	FLCS BRANCH D INV "GOOD"	1	1			
GROSS WEIGHT	1		2	3		MUX	FLCS DISCH LT BRANCH A	1	1			
MACH NUMBER	1			3	4	MUX	FLCS DISCH LT BRANCH B	1	1			
TIME	16	1	2	3		SAU CALC	FLCS DISCH LT BRANCH C	1	1			
PITCH RATE	16	1		3		FCC ANALOG	FLCS DISCH LT BRANCH D	1	1			
PITCH ACCELERATION	16			3		SAU CALC	FLCS CAUTION RESET	1	1			
ROLL RATE	16	1		3		FCC ANALOG	DUAL FLCS FAIL	1	1			
ROLL ACCELERATION	16			3		SAU CALC	FWD FUEL RESERVE LOW	1	1			
YAW RATE	16	1		3		FCC ANALOG	AFT FUEL RESERVE LOW	1	1			
YAW ACCELERATION	16			3		SAU CALC	HYD PRESSURE "A" LOW	1	1			
LONGITUDINAL ACCELERATION	4	1		3		FCC ANALOG	HYD PRESSURE "B" LOW	1	1			
LATERAL ACCELERATION	8	1		3		FCC ANALOG	FSC CAUTION "RESET"	1	1			
VERTICAL ACCELERATION	16	1	2	3		FCC ANALOG	AUTO PILOT - ROLL/HDG	1	1			
DYNAMIC PRESSURE	1			3		SAU CALC	AUTO PILOT - ROLL/ATT	1	1			
RUDDER POSITION	8	1		3		LVDT	AUTO PILOT - PITCH/HDG	1	1			
LEFT HT POSITION	16	1		3		LVDT	AUTO PILOT - PITCH/ATT	1	1			
RIGHT HT POSITION	16	1		3		LVDT	FWD PADDLE DEPRESS	1	1			
LEFT FLAPERON POSITION	16	1		3		LVDT	AFT PADDLE DEPRESS	1	1			
RIGHT FLAPERON POSITION	16	1		3		LVDT	CONTROL STICK POS	1	1			
LEF POSITION	4	1				LEF FDBK	CAT I - CAT III POS	1	1		3	
STICK FORCE - LONG.	4	1				FCC	DUAL FC FAILURE	1	1			
STICK FORCE - LATERAL	4	1				FCC	AUTO TF SELECT	1	1		3	
ENGINE POWER LEVER ANGLE	2	1			4	HARD WIRED	MANUAL TF SELECT	1	1		3	
CORE RPM (N2)	2	1			4	INDICATOR	TF FAILURE LIGHT	1	1			
FAN RPM (N1)	2	1			4	N1 SIGNAL	OBS WARN LIGHT	1	1			
FAN TURBINE INLET TEMP	2	1			4	INDICATOR	AUTO TF FAILURE LIGHT	1	1			
NOZZLE POSITION	1	1				INDICATOR	VALID WEAPON RELEASE	1	1		3	
TOTAL FUEL QUANTITY	1	1		3		MUX	SIMULATED WEAPON RELEASE	1	1			
FWD FUEL QUANTITY	1/5	1				FLC	MLG WEIGHT-ON-WHEELS	1	1	2	3	
AFT FUEL QUANTITY	1/5	1				FLC	LG DOWN COMMAND	16	1	2	3	
FUEL FLOW RATE	1	1				FF METER	COMM XMIT (UHF/VHF)	1	1			
EXTERNAL STORES WEIGHT	1			3		MUX	OVERHEAT	1	1			
PYROMETER (GE Only)	1				4	FCC ANALOG	LEF ASYMETRY LOCKUP	1	1			
5 KVA PMG FREQ	2	1				FCC ANALOG						

AC5343

SPECIAL EVENTS PRESERVED

MAIN GENERATOR FAIL	NZ > +9.5G
LEADING EDGE FLAP LOCKUP	NZ < −3.0G
DUAL FLT CONTROL FAIL	5KVA PMG FREQ > 109%
N_2 RPM > 97%	FTIT > 1015°C
N_1 RPM > 115%	N_2 < 55%, PLA > 14°
N_1 > 110% + D1/DT > 30%/SEC	PLA < 14° W/WOW
AOA > 29°	FLCS CAUTION RESET
AOA < −5°	OVERHEAT DISCRETE

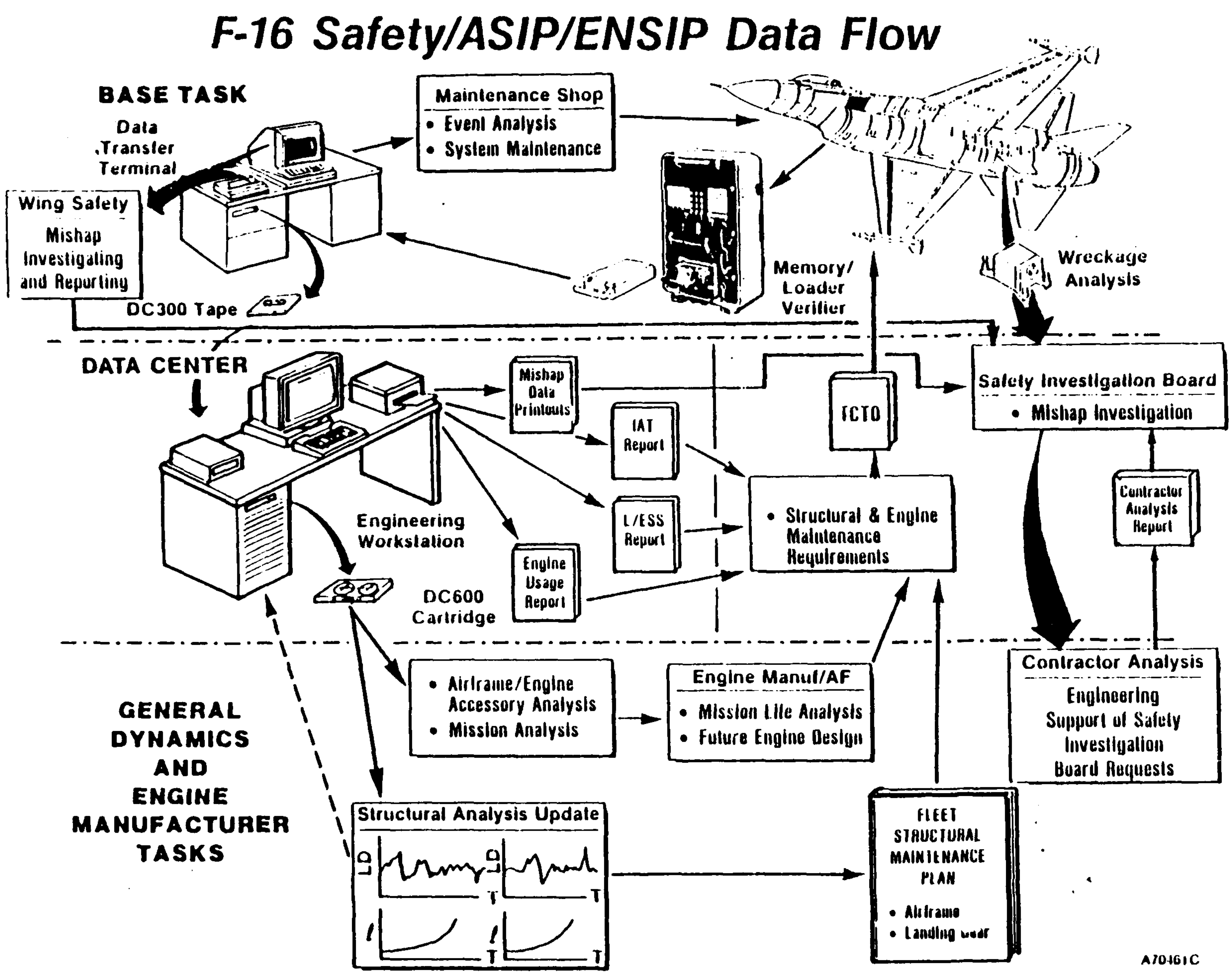
- THIRTY (30) SECONDS OF DATA PRESERVED PER EVENT
- EVENT OCCURRENCE REPORTED
- DATA PRESERVED UNTIL START OF NEXT FLIGHT

CSFDR A/B PARAMETER LIST

PARAMETER	SIGNAL TYPE	DATA TYPE			
REF CLOCK TIME	-	1	2	3	4
CORRECTED ALTITUDE (HC)	MUX	1		3	4
CALIBRATED AIRSPEED (VCAL)	MUX	1		3	
TRUE FREESTREAM AIR TEMP	MUX	1		3	4
MACH NUMBER	MUX			3	4
RADAR ALTITUDE	MUX	1			
RADAR ALT LOW SET	MUX	1			
EXTERNAL STORES WEIGHT	MUX			3	
TOTAL FUEL QUANTITY (FQ)	MUX	1		3	
GROSS WEIGHT (GW)	MUX		2	3	
ANGLE OF ATTACK (ALPHA)	MUX	1		3	
PITCH ATTITUDE	MUX	1			
ROLL ATTITUDE	MUX	1			
MAG. HEADING	MUX	1			
GREAT CIRCLE STEERING ERROR	MUX	1			
BREAKAWAY BARS ON	MUX	1			
RADAR LOCK-ON	MUX	1			
DELIVERY MODES ON THE SCP	MUX	1			
MAINTENANCE FAULT PRESENT	MUX	1	2	3	4
MAINTENANCE FAULT LIST	MUX	1	2	3	4

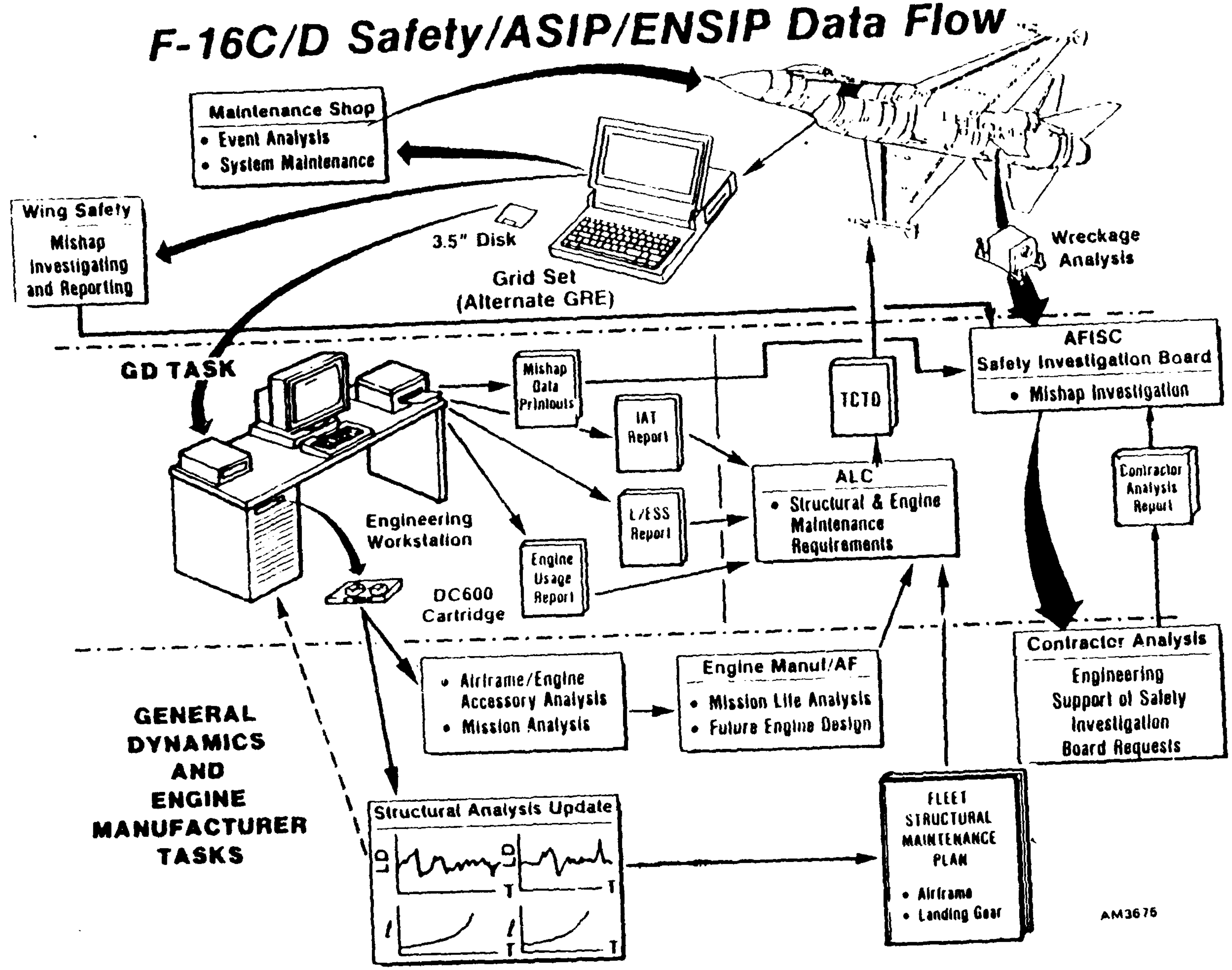
CSFDR A/B PARAMETER LIST

PARAMETER	SIGNAL TYPE	DATA TYPE			
ROLL RATE (P)	ANALOG	1	2	3	
PITCH RATE (Q)	ANALOG	1	2	3	
YAW RATE (R)	ANALOG	1	2	3	
LONG. ACCELERATION	ANALOG	1		3	4
LATERAL ACCELERATION (NYMEAS)	ANALOG	1		3	
NORMAL ACCELERATION (NZMEAS)	ANALOG	1	2	3	4
RUDDER POSITION (DR)	ANALOG	1		3	
LEFT HT POSITION (DHL)	ANALOG	1		3	
RIGHT HT POSITION (DHR)	ANALOG	1		3	
LEFT FLAPERON POSITION (DFL)	ANALOG	1		3	
RIGHT FLAPERON POSITION (DFR)	ANALOG	1		3	
POWER LEVER ANGLE (PLA)	ANALOG	1			4
CORE RPM (N2)	ANALOG	1			4
VALID WEAPON RELEASE	DISCRETE	1		3	
MLG WEIGHT-ON-WHEELS (RIGHT)	DISCRETE	1	2	3	4
LANDING GEAR DOWN COMMAND	DISCRETE	1	2	3	
CANOPY OPEN	DISCRETE	1			
ROLL ACCELERATION	COMPUTED		2	3	
PITCH ACCELERATION	COMPUTED		2	3	
YAW ACCELERATION	COMPUTED		2	3	
NORMAL ACCEL (NZMEAS)*GROSS WGT	COMPUTED		2		



F-16 Safety/ASIP/ENSIP Data Flow
BASE TASK
Data Transfer Terminal
Wing Safety
Mishap Investigating and Reporting
DC300 Tape
Maintenance Shop
• Event Analysis
• System Maintenance
Memory/ Loader Verifier
Wreckage Analysis
DATA CENTER
Mishap Data Printouts
IAT Report
L/ESS Report
Engine Usage Report
FCTO
• Structural & Engine Maintenance Requirements
Safety Investigation Board
• Mishap Investigation
Contractor Analysis Report
Engineering Workstation
DC600 Cartridge
GENERAL DYNAMICS AND ENGINE MANUFACTURER TASKS
• Airframe/Engine Accessory Analysis
• Mission Analysis
Engine Manuf/AF
• Mission Life Analysis
• Future Engine Design
Contractor Analysis
Engineering Support of Safety Investigation Board Requests
Structural Analysis Update
FLEET STRUCTURAL MAINTENANCE PLAN
• Airframe
A70461C

Interim USAF
F-16C/D Safety/ASIP/ENSIP Data Flow



CSFDR ENGINEERING WORKSTATION

CSFDR EWS SYSTEM TAILORED TO:

- o EFFICIENTLY PROCESS LARGE QUANTITIES OF COMPRESSED RAW CSFDR DATA
- o PROVIDE FIRST LINE DATA VALIDATION AND EDITING BY TECHNICIANS
- o CONDUCT MISHAP/INCIDENT ANALYSIS AND REPORTING
- o CONDUCT INDIVIDUAL AIRCRAFT TRACKING ANALYSIS AND REPORTING
- o CONDUCT LOADS/ENVIRONMENT SPECTRA SURVEY ANALYSIS AND REPORTING
- o CONDUCT ENGINE USAGE ANALYSIS AND REPORTING
- o PROVIDE GREATLY IMPROVED DOCUMENT PREPARATION CAPABILITY

O INITIAL CSFDR DATA PROCESSING SOFTWARE DEVELOPMENT BY GD UNDER 2ND MULTI-YEAR BUY

O ACSN FOR USAF EWS HARDWARE SUBMITTED TO GD SEP 87

O USAF EWS NEED DATE - MID 90

CSFDR ENGINEERING WORKSTATION

SYSTEM HARDWARE REQUIREMENTS

USAF SYSTEM(S) SHOULD BE COMPARABLE TO GD SYSTEM IN:

- o PROCESSING POWER (32BIT MICROPROCESSOR WITH FPA)
- o COMMUNICATION AND INTERFACE CAPABILITIES (MODEM AND ETHERNET NETWORK)
- o INPUT/OUTPUT DEVICES (DC300 TAPE DRIVE, 3.5 FLOPPY, LASER PRINTER)
- o MASS STORAGE DEVICED (500+ MBYTE DISK(S) AND 60+ MBYTE BACKUP TAPE)
- o INTELLIGENT HIGH RESOLUTION GRAPHICS DISPLAYS (PREFERABLY CAPABLE OF DISK-LESS OPERATION IN THE NETWORK ENVIRONMENT)

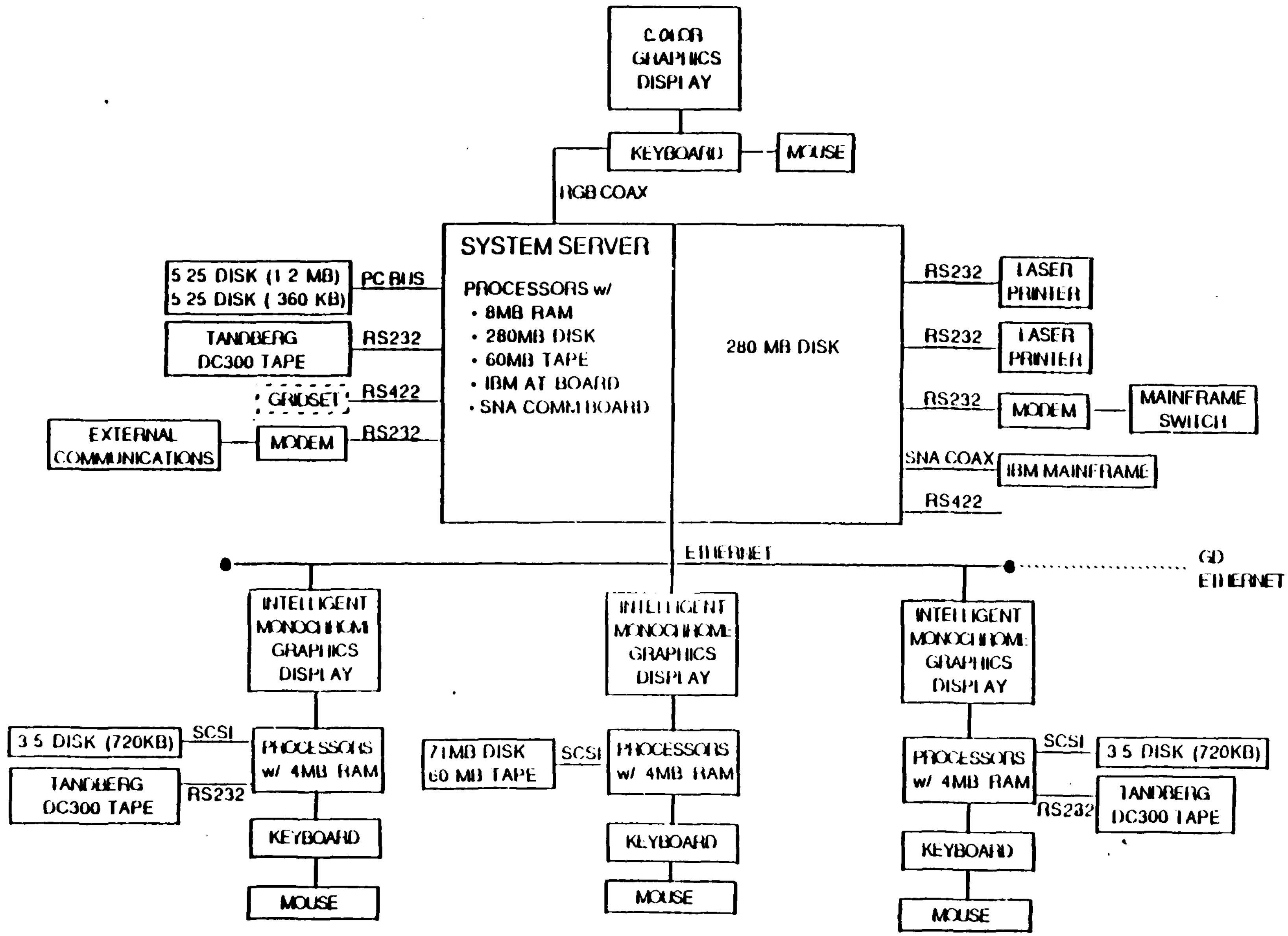
MAINFRAME COMMUNICATION AND PROCESSING IS NOT REQUIRED

SYSTEM SOFTWARE REQUIREMENTS

USAF SYSTEM(S) SHOULD HAVE THE SAME SOFTWARE REQUIREMENTS AS THE GD SYSTEM PARTICULARLY THE FOLLOWING:

- o UNIX OPERATING SYSTEM (EITHER UNIX 4.2 OR V)
- o C AND FORTRAN 77 COMPILERS, LINKER AND DEBUGGER
- o CORE AND/OR CGI GRAPHICS LIBRARIES
- o DOCUMENT PREPARATION ENVIRONMENT

CSFDR ENGINEERING WORKSTATION HARDWARE



CSFDR ENGINEERING WORKSTATION

MANPOWER REQUIREMENTS

PROCESSING CAPABILITY AND MANPOWER LOADING CALCULATIONS BASED ON THE FOLLOWING ASSUMPTIONS:

- 25 HOURS PER MONTH PER AIRCRAFT
- 150 HOURS (PHASED INSPECTION)PER DOWNLOADING PER AIRCRAFT
- TWO (2) DOWNLOADINGS PER YEAR PER AIRCRAFT
- 2000 DOWNLOADINGS PER YEAR PER 1000 AIRCRAFT
- TWO(2) HOURS FOR DATA PROCESSING (EXCLUDING DOCUMENT PREPARATION) PER DOWNLOAD (ONE DC300 TAPE)
- ONE (1) HOUR PER REPORT FOR IAT, L/ESS SUMMARY AND ENGINE USAGE SUMMARY
- EIGHT (8) HOURS PER REPORT FOR L/ESS ANALYSIS AND ENGINE USAGE ANALYSIS
- 40 TO 80 HOURS PER REPORT FOR DOCUMENT ASSEMBLY AND MASTER COPY
- 2000 MANHOURS PER YEAR PER COMPUTER OPERATOR

- TOTAL MANPOWER LOADING - APPROX. 5000 MANHOURS PER YEAR PER 1000 AIRCRAFT PLUS PROGRAM MAINTENANCE AND ADMINISTRATIVE SUPPORT

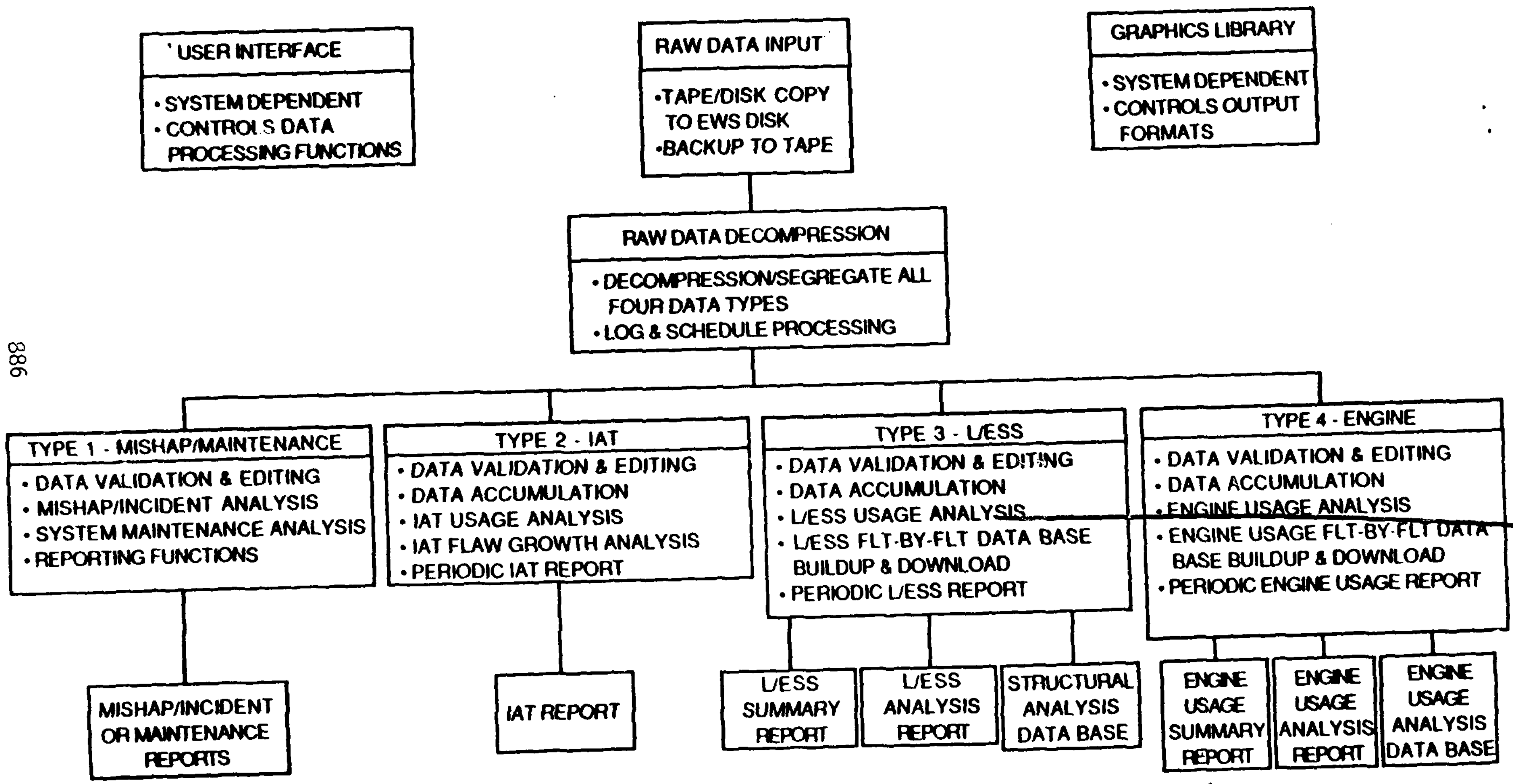
CSFDR ENGINEERING WORKSTATION

MANPOWER REQUIREMENTS (CONT'D)

BASED ON THE PREVIOUS ASSUMPTIONS GD RECOMMENDS THE FOLLOWING EQUIPMENT AND MANPOWER AVAILABLE AND DEDICATED TO THE USAF CSFDR DATA PROCESSING TASK:

- ONE (1) EWS SYSTEM SERVER WITH 500+ MBYTES MASS STORAGE PER 1000 AIRCRAFT
- FOUR (4) GRAPHICS WORKSTATIONS PER SYSTEM SERVER (1000 AIRCRAFT)
- ONE (1) COMPUTER OPERATOR PER GRAPHICS WORKSTATION (8 HOURS PER DAY)
- ONE (1) COMPUTER PROGRAMMER PER SYSTEM (WORKSTATIONS AND SERVER)
- ONE (1) ADMINISTRATIVE SUPPORT PERSON (LEADMAN)

CSFDR DATA PROCESSING SOFTWARE REQUIREMENTS (GD DEVELOPED)



F-16 CSFDR INCORPORATIONS

C/D BLOCK 25/30	Z8002	C/D OFP	PRODUCTION JUN 87
		30/40 OFP	PRODUCTION FEB 88
		HARDWARE	RETROFIT JUN 88
		30/40 OFP	RETROFIT MAY 88
C/D BLOCK 40	Z8002	30/40 OFP	PRODUCTION JAN 89
(DOWNWARD COMPATIBLE - CAN BE USED ON BLK 25/30 A/C)			
THUNDERBIRDS	Z8002	A/B OFP	AUG 88
AIR DEFENSE FIGHTER	1750	A/B OFP	SEP 89
A/B AIRCRAFT	1750	A/B OFP	APPROX. 91

CONCLUSIONS

- AVTR AND CSFDR ON ALL F-16C/D
- SAFETY PARAMETERS CRASH SURVIVABLE
- DATA DOWNLOAD AT PHASED INSPECTION REDUCES DATA LOSS
- DATA PROCESSING CAN BE DONE ANYWHERE WITH EWS

CSFDR MISHAP / MAINTENANCE ANALYSIS REQUIREMENTS

PROVIDE CSFDR SYSTEM MAINTENANCE ANALYSIS FOR EACH DOWNLOAD

- DATA SOURCES
  - Type 1 Baseline Event
  - Type 1 Special Events
  - Type 1 15 Minute Time History
  - Type 2 BIT Status
  - Type 3/4 Flight by Flight Time Histories

- ANALYSIS METHOD
  - Evaluation of Baseline & Special Events
  - Evaluation of System BIT
  - Evaluation of Flight by Flight Time Histories
  - Summarize Content
  - Summarize Invalid Parameters
  - Summarize BIT Failures

- REPORT METHOD
  - Display Maintenance Summary
  - Develop and Printout Maintenance Report
  - Record Maintenance Summary in Individual Aircraft Usage Database for IAT / L/ESS / Engine Reports

FLR PARAMETERS

ITEM	SYMBOL	SAMPLE RATE
PRESSURE ALTITUDE	H_P	1
CALIBRATED AIRSPEED	V_C	1
PITCH RATE	Q	15
YAW RATE	R	15
ROLL RATE	P	30
ROLL ACCELERATION	P	30
NORMAL ACCELERATION	N_Z	15
LATERAL ACCELERATION	N_Y	15
LONGITUDINAL ACCEL (1)	N_X	5
FUEL QUANTITY	F_Q	1
ENGINE ROTOR SPEED	N_2	1
RUDDER POSITION (1)	Δ_R	15
L/H H.T. POSITION (1)	Δ_{HL}	15
R/H H.T. POSITION (1)	Δ_{HR}	15
LEFT FLAPERON POSITION (1)	Δ_{FL}	15
RIGHT FLAPERON POSITION (1)	Δ_{FR}	15
ENG POWER LEVER ANGLE (2)	PLA	5
STRUCTURAL STRAIN (1)	σ	15
WEIGHT ON WHEELS EVENT	E_1	1
LDG GEAR POSITION EVENT	E_2	1
WEAPONS RELEASE EVENT	E_3	1
DOCUMENTARY DATA	DD	1
TIMING WORD/GAP	T	1

(1) FLR PECULIAR GROUP B SENSORS (2) BLOCK 11 & ON AIRPLANES

TYPE 1 SOFTWARE

TOP LEVEL DESIGN

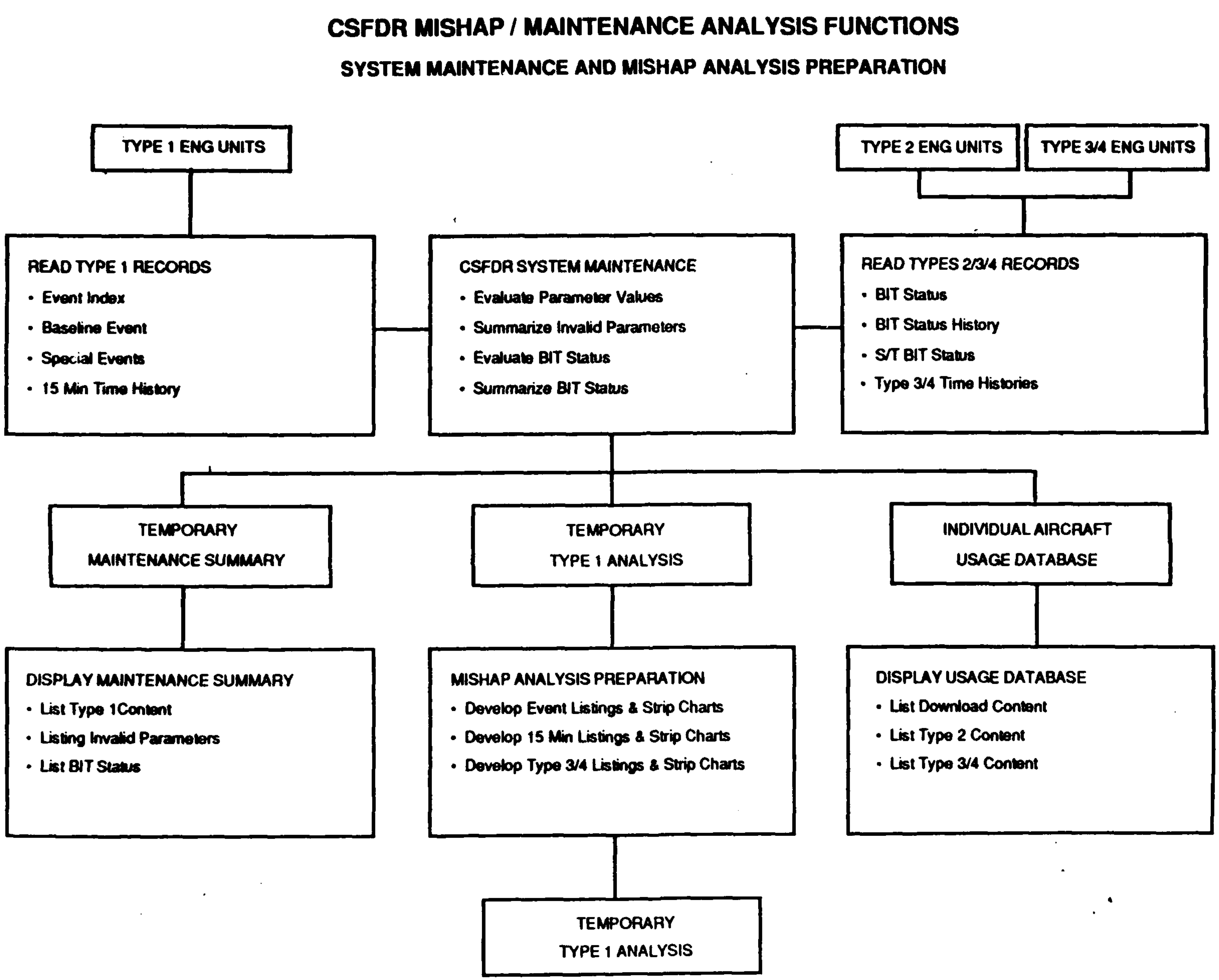
CSFDR MISHAP / MAINTENANCE ANALYSIS REQUIREMENTS

PROVIDE TIME HISTORY DATA FOR MISHAP ANALYSIS AS NEEDED

- DATA SOURCES
  - Type 1 Baseline Event
  - Type 1 Special Events
  - Type 1 15 Minute Time History
  - Type 3/4 Flight by Flight Time Histories

- ANALYSIS METHODS
  - Develop Listings & Strip Charts
  - User Selection of Events, Parameters and Time Spans
  - Display of Selected Data
  - Develop Time History Snapshots
  - User Selection of Events or Time Span and Views
  - Display of Selected Snapshots

- REPORT METHOD
  - Develop Mishap Analysis Report from Formatted Listings, Strip Charts and Snapshots
  - Integrate and Printout Mishap Report


CSFDR MISHAP / MAINTENANCE ANALYSIS FUNCTIONS
SYSTEM MAINTENANCE AND MISHAP ANALYSIS PREPARATION
TYPE 1 ENG UNITS
TYPE 2 ENG UNITS
TYPE 3/4 ENG UNITS
READ TYPE 1 RECORDS
• Event Index
• Baseline Event
• Special Events
• 15 Min Time History
CSFDR SYSTEM MAINTENANCE
• Evaluate Parameter Values
• Summarize Invalid Parameters
• Evaluate BIT Status
• Summarize BIT Status
READ TYPES 2/3/4 RECORDS
• BIT Status
• BIT Status History
• S/T BIT Status
• Type 3/4 Time Histories
TEMPORARY MAINTENANCE SUMMARY
TEMPORARY TYPE 1 ANALYSIS
INDIVIDUAL AIRCRAFT USAGE DATABASE
DISPLAY MAINTENANCE SUMMARY
• List Type 1Content
• Listing Invalid Parameters
• List BIT Status
MISHAP ANALYSIS PREPARATION
• Develop Event Listings & Strip Charts
• Develop 15 Min Listings & Strip Charts
• Develop Type 3/4 Listings & Strip Charts
DISPLAY USAGE DATABASE
• List Download Content
• List Type 2 Content
• List Type 3/4 Content
TEMPORARY TYPE 1 ANALYSIS

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(ASIP/ENSIP)

1 - 3 DECEMBER 1987

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